IJCRT.ORG ISSN : 2320-2882

INTERNATIONAL JOURNAL OF CREATIVE RESEARCH THOUGHTS (IJCRT)

An International Open Access, Peer-reviewed, Refereed Journal

DEVELOPMENT OF SOLID ROCKET MOTOR FOR ASCENT AND SOFT LANDING IN VERTICAL FLIGHT

[1]Anutha M, [2]Aishwarya K, [3]Kavya Patil, [4]Prajwal Nayak, [5]Prajwal P.

[1]Asst. professor, [2]Student, [3]Student, [4]Student, [5]Student.
[1]Aeronautical department,

[1]Dayananda Sagar College of Engineering, Bangalore, India.

Abstract: The high launch cost of rockets has led to a great interest in the development of reusable rockets. These missions were achieved with the use of liquid engine thrust vectoring. This paper discusses design and flight simulation of solid propellant rocket equipped with thrust vector control. Solid rocket motor has been designed with a goal to reach 10 km altitude. It is aimed that the solid rocket motor is to be recovered so that it can be reused for subsequent launches. The design and simulation are carried out assuming that the motion takes place in vertical direction, considering flat earth approximations. This paper deals with selection of solid propellant, grain design, calculation of performance parameters, design and optimization of the rocket motor and nozzle for desired performance, computational analysis and flight simulation with structural constraint. The flight simulation has been carried out using MATLAB® and Simulink® to prove the effectiveness of the concept. For simplicity, rocket is assumed as a point mass in flight simulation. The simulation results testify that the ascent motor is meeting the design goal of 10 km altitude and descent motor works perfectly to bring the rocket back to ground with closer to zero velocity.

***Index Terms* – thrust vector control, solid rocket motor, design and optimization of rocket motor, flight simulation.**

I. INTRODUCTION

Implementation of thrust vector control for solid rocket motor gives, stability and guidance which is the key to propulsive landings. Vectoring the thrust, empowers control over the direction of rocket's engine during flight so as to change its trajectory and stabilize it. This is related to SpaceX's Falcon series of self-landing rockets where the first and second stage is automated to return to earth and land so that it can be reused. This is a recent field which was successful in 2015 and very little information is available for rockets at a smaller scale for development. This project is to explore the possibility of application of thrust vector control for a solid propellant rocket motor in design and point mass simulation aspects.

Solid rocket engine ends up being a reliable and financially savvy propulsion system for wide scope of rocket-based applications beginning from little strategic weapons to current huge space supporters and furthermore generally simple in usage. Propulsive landing missions were achieved with the integration of thrust vectoring for liquid engines where it is relatively easy to vary the total thrust, this project will explore to see whether it is possible to achieve landing for solid propellant motor whose thrust remains almost constant.

Solid rocket motor has been designed with a goal to reach 10 km altitude using OPENMOTOR®. The theoretical calculations of grain configuration and nozzle is done using MATLAB. Then the rocket body and nozzle are designed such that they meet the desired properties. The MATLAB simulations are carried out for a point mass having cross sectional area of the motor bulkhead. The bulkhead has all the aerodynamic properties that of the actual rocket that has to be soft landed. Flow analysis over the solid rocket has been carried out in ANSYS®. The thrust vector control system is a flexible laminated bearing nozzle configuration which is similar to gimbal nozzle used in liquid engines to control and stabilize the rocket. The flight simulation is done using Simulink® in order to show the effectiveness of the concept.

Abbreviations and Acronyms

APCP	Ammonium perchlorate composite propellant
MIT	Massachusetts Institute of Technology
R_c	Burn rate co-efficient
N	Burn rate exponent
ISP	Specific impulse
PDMS	Poly-di-methyl-siloxane
HTPB	Hydroxyl-terminated polybutadiene
Al	Aluminium
AP	Ammonium perchlorate
CCF	Chopped carbon fibre
K_n	Ratio of burn area to throat area
V	Velocity
A	Area
Cd	Co-efficient of drag
ρ	Density
P	Pressure
M	Mach number
k	Specific heat ratio
Fi	Inertial force
F_d	Drag force
T	Thrust
$\dot{m}$	Mass flow rate
m	Mass
a	Acceleration
COG	Centre of gravity
D	Diameter of rocket body
V_{bo}	Burnout velocity
m_r	Rocket dead mass
N	Drag influence number
t	Thickness
σ	Yield strength of Aluminium 6061 t6
FOS	Factor of safety

II. DESIGN AND BURN SIMULATION

2.1 Analytical calculations

2.1.1 Prediction of propellant mass required to reach an altitude of 10km

The spreadsheet designed by Richard. A. Nakka is intended as an easy-to-use design aid for model rockets. Unlike most simulation software, that estimates peak altitude for a given rocket and motor combination, this program instead considers peak altitude as a design goal and based on this goal, the program computes propellant mass required to reach this altitude.

The various parameters that describe the proposed rocket vehicle, as well as basic motor sizing parameters are given as inputs for subsonic flights. These inputs are as follows:

- Altitude Goal
- Rocket basic data: Body diameter, Empty mass and Drag coefficient
- Motor sizing data: Chamber Diameter, Propellant loading, Propellant type and Thrust time
- Propellant data: Density and Specific Impulse

This program is meant only for low-powered rockets propelled by propellant that has a specific impulse in the range of 100 to 150. For high-powered rockets that utilizes a propellant of specific impulse of >180 (used in our case), this spread-sheet is taken as a reference and has been modified. This resulted in the mass of 26.348 kg for APCP as propellant.

2.1.2 Best mass

Best mass is the mass that the rocket should have in order to reach maximum altitude propelled by certain mass of propellant. If the total mass is less than the best mass then, the rocket will not have enough momentum to travel further after burnout. If the total mass is greater than the best mass then, the rocket will have large momentum and will reach a lower altitude before burnout. Therefore, consideration of best mass is a crucial aspect.

The Simulink program is designed and simulated for propellant mass data from above and the best mass is tabulated as shown in Table 2.2.

Table 2.2 Variation of altitude with mass

Mass (kg)	Altitude (m)
65.711	10140
63.711	10330
61.711	10390
59.711	10600
57.711	10630
55.711	10680
53.711	10710
51.711	10690
49.711	10650
47.711	10610
45.711	10520
43.711	10450
41.711	10340

A series of varying wet mass of the rocket for peak altitude is calculated.

1.Peak altitude of 10710 m for propellant mass of 26.348 kg

2.With this the total best mass of the Rocket (wet mass) = 53.711 kg

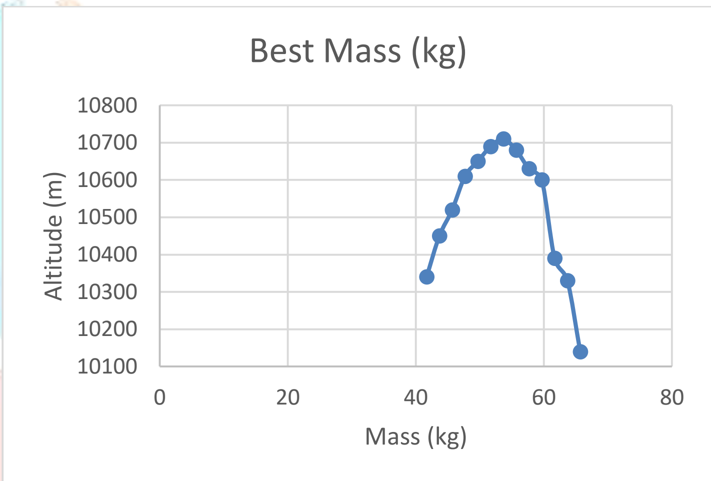

Fig.2.2 Variation of altitude with mass

The dry mass has to be around 27 kg and is designed and estimated in the later sections.

2.1.3 Structural constraint

As far as the cylindrical column is not extensively long (that is, L / D > 15) that may result in failure by buckling, thin-walled cylinders will fail in a collapsing, or crippling, mode. This is beneficial, as this means that the structure will remain mostly undeformed till collapse. Therefore, the rocket body has L/D ratio of 15, so that it is easier to control when a controller is implemented in further research. From ascend motor design, the outer diameter of the rocket is estimated to be 0.25 m and therefore the total length is 3.75m.

2.1.4 Motor case thickness

The thickness of the motor case is calculated using hoop stress for a maximum chamber pressure of 550 psi and factor of safety of 2.5. The material for the motor case is Aluminum T-6.

$$t = \frac{P \times D}{2 \times \sigma} \times FOS \quad (1)$$

$$t = \frac{550 \times 0.16}{2 \times 35000} \times 2.5$$

$$t = 3.2 \text{ mm}$$

2.2 Materials

2.2.1 Airframe materials

The conventional airframe materials for high-powered rockets are non - metallic, composite materials having high strength to weight ratio like fiberglass, carbon-fibre, phenolic and PVC (rules of National Association of Rocketry).

For this project the fiberglass is chosen as airframe material since it has corrosion resistance property, good structural strength, superior strength to weight ratio and it is cost effective- especially for complex shapes.

2.2.2 Motor case materials

1. Cardboard is used for small black powder model motors
2. Aluminum is used for larger composite-fuel hobby motors
3. Steel was used for the space shuttle boosters
4. Filament-wound graphite epoxy casings are used for high-performance motors

Aluminum T6 is preferred as motor case material. 6061-T6 aluminum has good structural strength and toughness and offers good finishing characteristics.

2.2.3 Thermal insulation by heat resistant polymers

The flame temperature of APCP is 2700 K. It is vital to include thermal insulation for the motor case for it to bear this high temperature. Thermal insulation is made of chopped carbon fibre (CCF) and aramid fibre (pulp form). 0.6 centimeters long CCFs and Kevlar pulp (KP) which are scattered in EPDM polymeric matrix is made for curing purpose. Six alternative laminates composed of these prepregs are shown to exhibit better thermal, mechanical, physical, and ablative properties when compared to non-laminated counterparts.

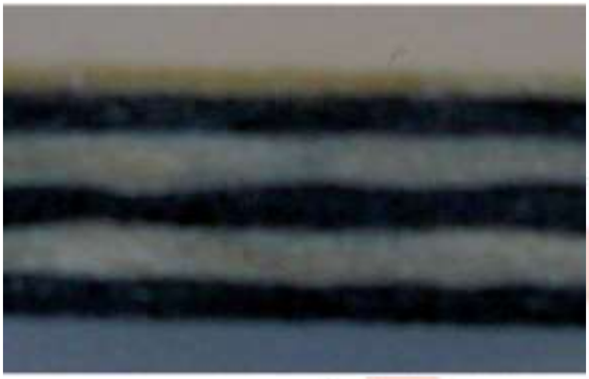

Fig.2.2.3 Photo of cross-section of a laminate consisting of six alternative layers of (CCF or KP) based prepregs.

Table 2.2.3 Properties of laminates made using six alternate layers of CCF and KP based EPDM prepregs

Property	Unit	Test value for laminate	Test value for hybrid (25 phr CCF + 25 phr KP)
Tensile strength	MPa	7.8 ± 0.5	11
Elongation	%	12.1 ± 1.2	7.4
Hardness	Shore A	88.9 ± 0.2	96.4
Density	gm/cm^3	1.239	1.256
Specific heat capacity	J/kg-K	1691 ± 4	1973
Thermal diffusivity	mm^2/s	0.085 ± 0.002	0.08
Thermal conductivity	W/m-K	0.178 ± 0.001	0.198
Ablation rate	mm/s	0.006 ± 0.0002	0.005
Outer case temperature (for 60 s)	°C	39.1 ± 0.5	82
TGA remaining weight	%	27 ± 0.5	24

2.2.4 Thermal insulation by heat resistant polymers

Now, we calculate the total heat transfer across the insulation to check if the motor casing can withstand this high temperature. From Fourier law of heat conduction, we have,

$$Q=\frac{K\times A\times \Delta T}{\Delta x}\times t \qquad (2)$$

Taking the burn time as 13 s,
And Area,

$$A=2\times\pi\times0.08\times0.96 \tag{3}$$

$$A=0.4825 \ m^2$$

$$Q=\frac{0.178\times0.4825\times(2700\text{-}300)}{0.006}\times13$$

$$Q=446.64 \text{ kJ}$$

Specific heat of material of case = 1000 J/kg
Total mass of motor casing = 6 kg
Hence, the raise in temperature of the case = $\Delta T = \frac{444.64\times10^3}{6000} = 74\ K$
which is well below the melting point of the material (855 – 925 K).

2.3 Motor design

Ammonium perchlorate composite propellant (APCP), a compound propellant that has both fuel and oxidizer mixed with a binder, usually of a rubbery nature, also known as 'Cherry Limeade', developed by MIT Rocketry team suits best and hence has been selected to power the rocket. The propellant contains ammonium perchlorate as the oxidant and hydroxyl-terminated polybutadiene (HTPB) as an elastomer binder. It includes aluminium, which along with the binder, which serves as the fuel. Usually, it is used in aerospace missions since it is easy to handle, store and has good propulsive characteristics.

Performance and Combustion Properties:
Density: 1688.4742 kg/m3
Rc: 0.0006346444 m/s
n: 0.327392
Typical ISP: 225 s

Table 2.3 Composition of APCP

Ingredient	Percentage
Binder	17.1%
Caster oil	0.3%
PDMS	0.05%
Triton X100	0.05%
Al	7.5%
200 AP	65.5%
90 AP	9.5%

The motor configuration that provides the optimum performance to reach this altitude is designed via burn simulation. The dimensions for maximum optimized rocket motor are calculated from MATLAB code. This iterative design is done such that the following constraints are satisfied:

- Port to throat area ratio > 2
- Peak mass flux < 0.907185 kg/s
- Maximum combustion chamber pressure = 3.792*106 pa
- Burn/Throat area ratio is between 180 to 250

Subjected to assumptions:

- No erosive burning
- Nozzle efficiency = 1 (Nozzle exit pressure = Atmospheric pressure)
- Throat erosion coefficient = 0
- Slag build-up coefficient = 0

2.3.1 Ascent motor design

The ascend motor has two different grain configurations are used that is; finocyl and bates. This grain configuration provides almost neutral burn, as the surface area remains fairly constant throughout the burn. To get greater efficiency in delivering total impulse from a rocket motor, neutral burning is usually preferred and also the nozzle operates efficiently when chamber pressure is constant. The ascend motor is designed in SOLIDWORKS®.

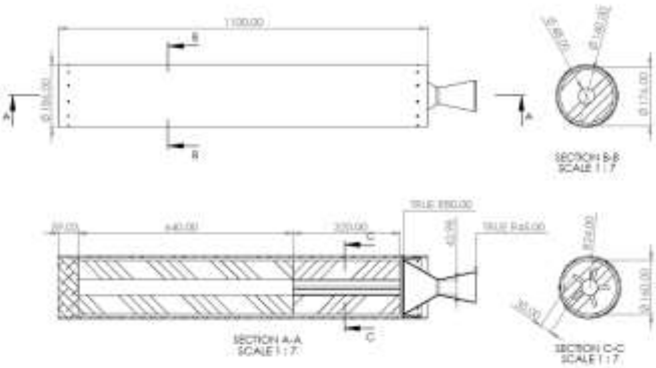

Fig.2.3.1 Ascent motor design

2.3.2 Descent motor design

For descend motor an end burning grain configuration best suits because for propulsive landing, the thrust level should be low and it should have long duration of burn. The descend motor is also designed in SOLIDWORKS®.

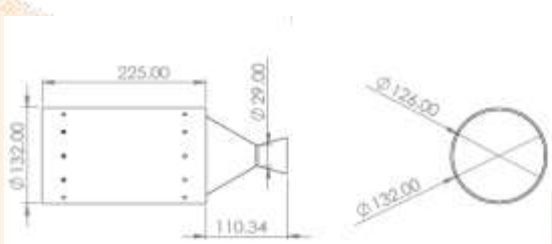

Fig.2.3.2 Descent motor grain configuration

2.4 Burn simulation

The combustion parameters are obtained from burn simulation in OPENMOTOR®. This software takes type of propellant, grain configuration and its dimensions as input and calculates the optimized nozzle dimensions that gives maximum specific impulse. By giving these inputs from the theoretical calculations from MATLAB®, burn simulation is carried out in OPENMOTOR® and following results were obtained.

2.4.1 Ascent motor

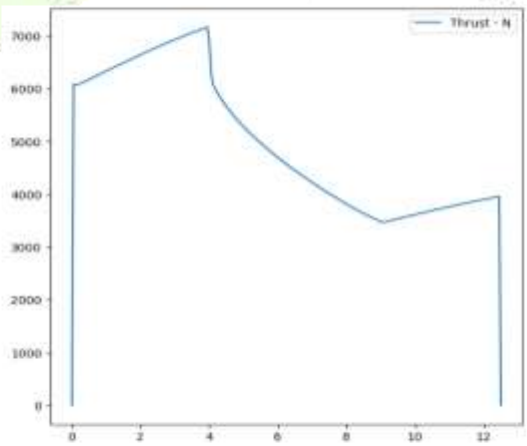

Fig.2.4.1 Thrust curve for ascend motor

Motor statistics:
Total Impulse =62031.636 Ns
Burn time = 12.5 s
Average Pressure = 2.564*106 Pa
Peak Pressure = 3.852*106 Pa
Peak Kn = 254.61
Propellant mass = 26.35 kg

Propellant length = 0.849 m
Port/Throat Ratio = 2.969

2.4.2 Descent motor

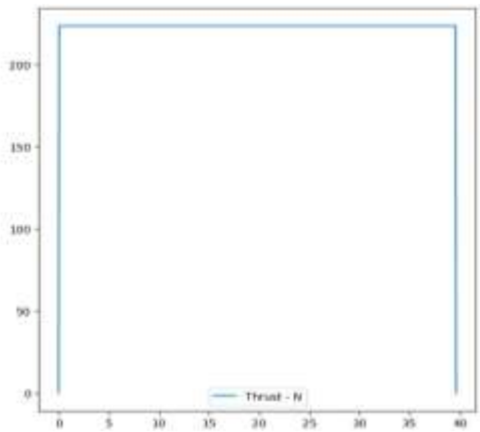

Fig.2.4.2 Thrust curve for descend motor

Motor statistics:
Total Impulse = 8848.67 Ns
Burn time = 39.65 s
Average Pressure = 2.806*106 Pa
Peak Pressure = 2.810*106 Pa
Peak Kn = 216.263
Propellant mass = 3.711 kg
Propellant length = 0.1799 m

2.5.1 3D motor design

For motor casing Aluminium T-6 alloy is chosen, as it has high strength to weight ratio and pressure handling capabilities. A layer of CCF material of 6mm thickness is used for thermal insulation. The motor case thickness required to sustain a maximum chamber pressure of 3.627*106 Pa was found to be 3.2mm as per the calculations shown above.


Fig.2.5.1 3D motor design

2.5.2 Nozzle design

A nozzle is a generally basic device, only an exceptionally molded cylinder through which hot gases flow. Since the stream is supersonic, the nozzle utilized is a convergent-divergent nozzle. Flexible laminated bearing nozzle is like that of gimbal nozzle and is by all accounts the best choice if there should be an occurrence of solid rocket engine. It is assumed that no erosion takes places during the burn. The dimensions for nozzle design are taken from burn simulation as follows:
Inlet diameter: 160mm
Throat diameter: 43mm
Exit diameter: 90mm
Divergence half angle: 15 degree
Convergence half angle: 30degree

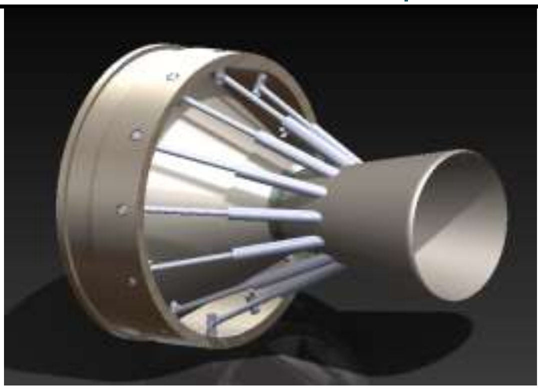

Fig.2.5.2 Convergent-Divergent nozzle design

2.5.3 Fin design

Although the rocket is equipped with TVC, in order to have at least neutral stability, presence of fins is essential. Therefore, fins are designed such that the rocket has neutral stability. The rocket has a total of 4 fins.

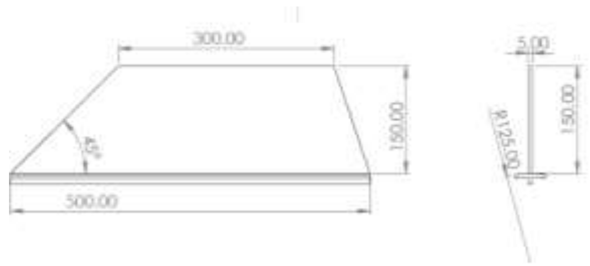

Fig.2.5.3 Fin design

2.5.4 Rocket body design

Considering the dimensions of the motor, the rocket body design is optimized in OPEN ROCKET®. Then, the rocket body is designed in SOLIDWORKS® and thickness of the rocket body is calculated using hoop stress. The total length of the rocket is 3.75 m. The fins are designed such that the centre of pressure is located just behind the centre of gravity for the rocket to be stable or coincide to have neutral stability. The payload and landing legs are placed at the tip and end of the nose cone respectively. This is done because the rocket is inverted during the descent flight after the ejection of nose cone. It becomes much easier to operate the descent motor without involving any complex mechanism.

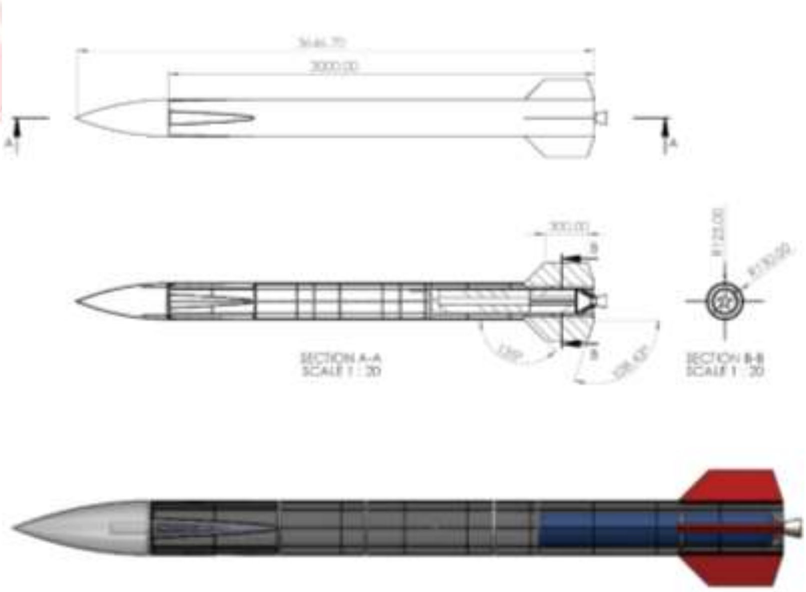

Fig.2.5.4 Rocket body design

2.6 Flow analysis

2.6.1 Nozzle flow analysis

The analysis of flow inside the nozzle has been carried out in Ansys 18.1® and following results were obtained. The hot fumes leave the ignition chamber and combines down to the base region, or throat of the nozzle. The throat diameter is selected such that the flow is choked at this section and set the mass flow rate through the system. The exit velocity, pressure, and mass flow through the nozzle decides the measure of thrust delivered by the nozzle.

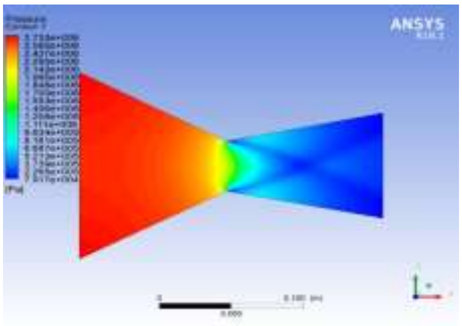

Fig.2.6.1(a) Pressure contour

The expansion of a supersonic flow causes the decrease in static pressure and temperature from the throat to the exit, so the exit pressure and temperature are determined by amount of the expansion. The speed of sound at exit that determines the exit velocity is governed by the exit temperature.

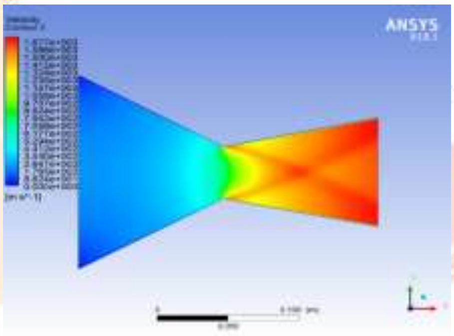

Fig.2.6.1(b) Velocity contour

The stream in the throat is sonic which implies the Mach number is equivalent to one at the throat. In the divergent section, the geometry strolls and the flow is isentropically expanded to Mach number greater than 1 that depends on the ratio of the exit to the throat. The stream leaves the nozzle exit at around Mach 5.

2.6.2 Flow analysis over the rocket body

The flow over the rocket body is analyzed in SOLIDWORKS® for Mach 1.5. The rocket body is treated as a no-slip wall, which means the fluid will have zero velocity relative to the surface boundary. Since the rocket is fixed in its reference frame, the velocity at the solid boundary is zero.

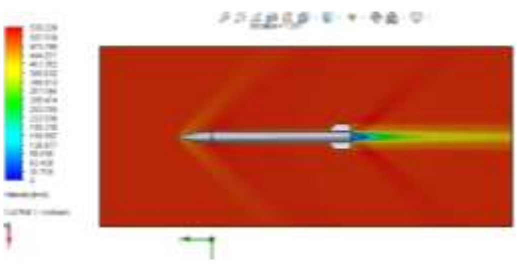

Fig.2.6.2(a) Velocity contour

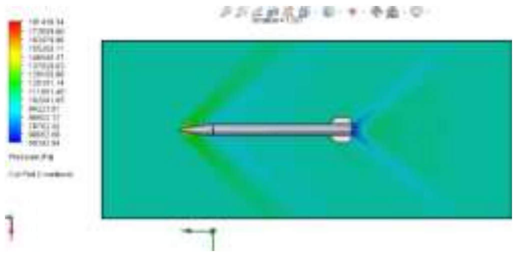

Fig.2.6.2(b) Pressure contour

III. SIMULATION

3.1 Flight simulation

The flight simulation is done such that the rocket reaches an altitude of 10 km and then perfectly land to ground. The flight simulation is done considering a point mass in Simulink. It includes two subsystems namely, Engine and Aerodynamic drag to reduce the complex design of Simulink block. With the initial inputs such as thrust curve, engine mass, propellant mass the sum of forces acting on the rocket body has been calculated. Then using newton's second law of motion the acceleration of the vehicle is calculated. Then by integration the velocity and altitude of the rocket body is determined. In the simulation, initially the drag is considered as 0 then from the calculated velocity the drag is determined as a closed loop. The results are discussed below.

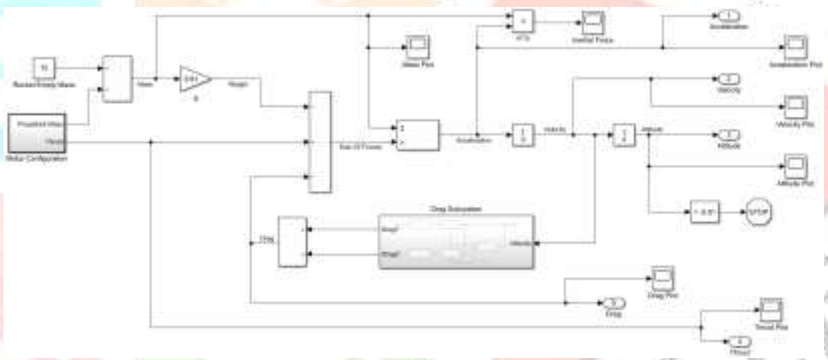

Fig.3.1 (a) Simulink block and connections for simulation of flight

The thrust variation with a time step of 0.05 s is imported to MATLAB. Then the fraction of the propellant burned is calculated. The Rocket empty weight is added from the estimation of best mass. Multiplying the total mass with acceleration due to gravity gives the force of gravity.

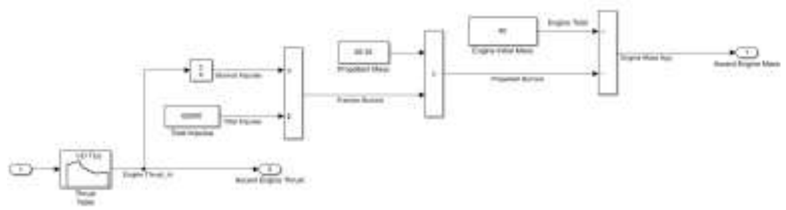

Fig.3.1 (b) Simulink blocks and connections to calculate engine thrust and weight

The variation of air density, pressure and speed of sound with altitude is also considered. For this, the international standard atmosphere model (ISA) block is taken. It is observed that the rocket is supersonic during its flight path and hence the variation of coefficient of drag with Mach number is implemented. This data is taken from RASAero 2®.

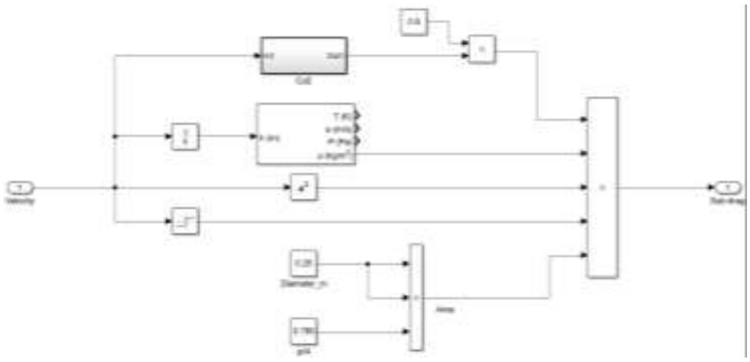

Fig.3.1 (c) Simulink block and connections to calculate subsonic drag

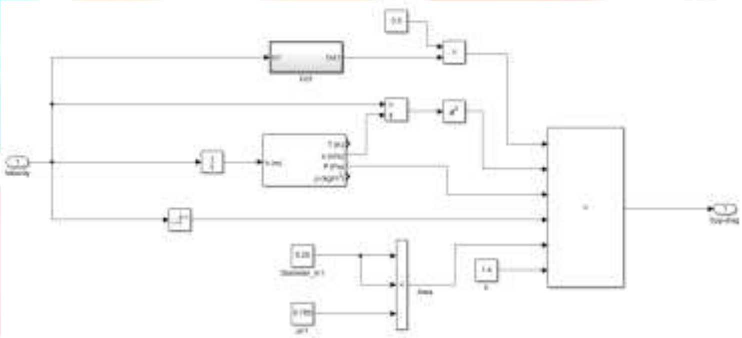

Fig.3.1 (d) Simulink block and connections to calculate supersonic drag

3.2 Results

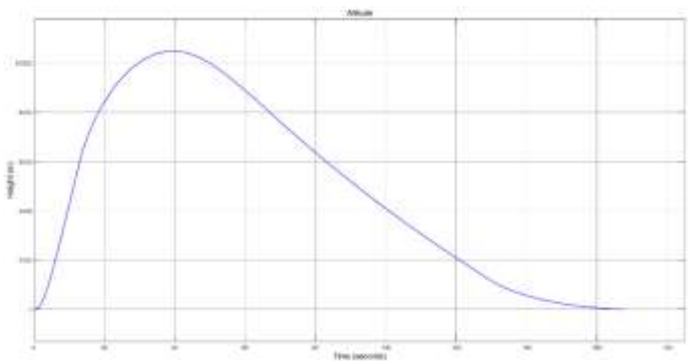

Fig.3.2(a) variation of altitude with time

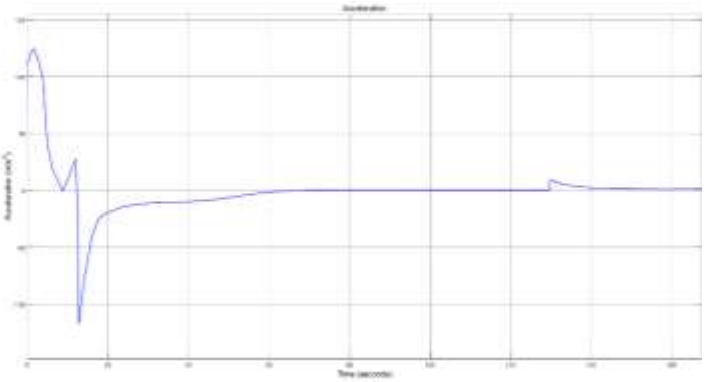

Fig.3.2(b) variation of acceleration with time

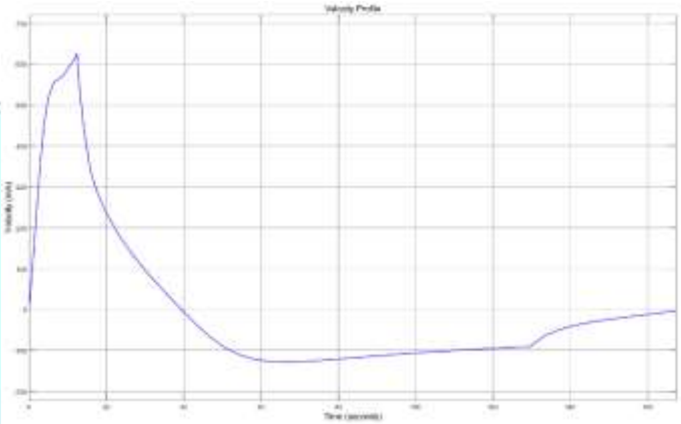

Fig.3.2(c) variation of velocity with time

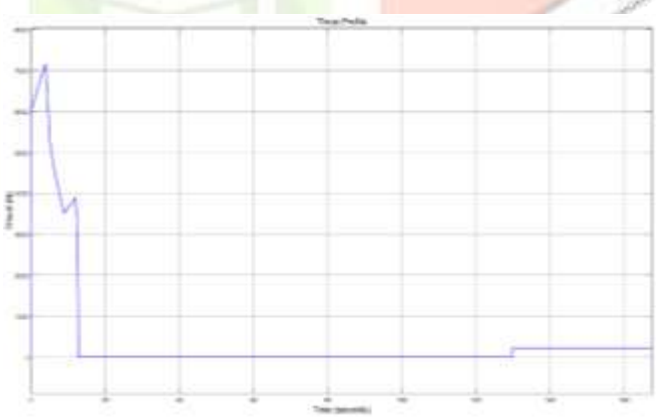

Fig.3.2(d) variation of thrust with time

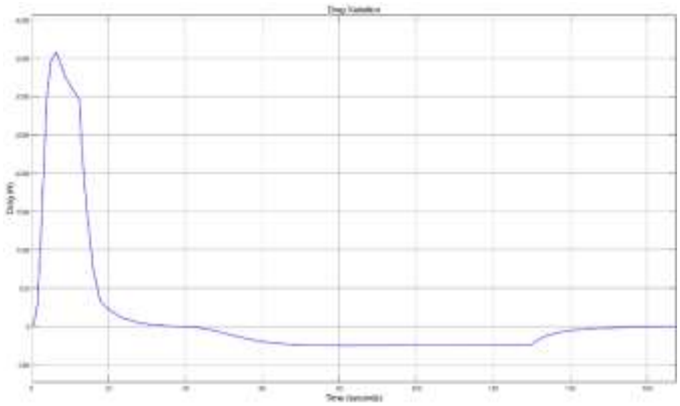

Fig.3.2(e) variation of drag with time

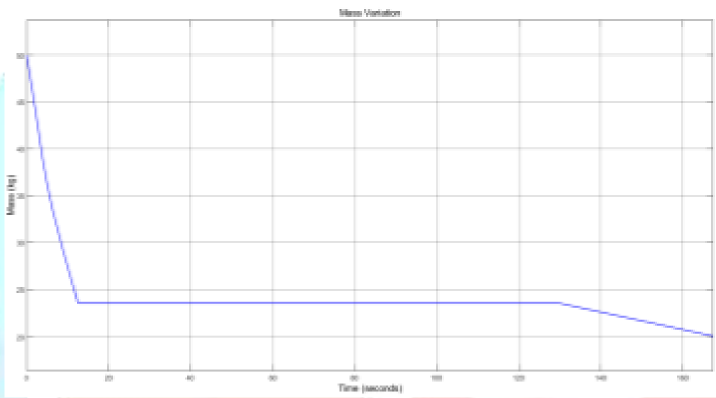

Fig.3.2(f) variation of mass with time

The Rocket reaches an altitude of 10km in 40 seconds and then begins to descend under gravity. The rocket is under free fall for 128 seconds post apogee. The slow decrease in altitude can be depicted from the graph during which the descent motor is operated. The total time of flight is 168 seconds. The maximum acceleration experienced is 120 m/s^2. It is seen that the acceleration is negative at the time of burnout and then increases gradually due to inertial force. The flow analysis is done for the rocket body from the velocity plot as shown above. It is clear that the rocket has zero velocity when the rocket touches ground. Thereby, the rocket is soft landed. The delay time (time at which the descent motor is ignited/operated) is 129.75 s. If the motor is ignited earlier, then the rocket will reach zero velocity at a higher altitude and if the motor is ignited late, then the rocket will hard land. Therefore, delay time is a crucial aspect to be considered while operating the descent motor. It is assumed that the motor can be ignited at any instant of time, although APCP propellants have difficulty in igniting at a desired instant.

3.3 Stability

The rocket is designed in OPENROCKET to calculate stability. This software makes use of Barrowman equations to calculate center of pressure and it is seen that the rocket is neutrally stable for this configuration. CG and CP coincide at a point 243 cm away from the tip of the nose giving 0 caliber stability.

Rocket Design

Rocket
Stages: 1
Mass (Empty): 53457 g
Stability: 0 cal
CG: 243 cm
CP: 243 cm

Parts Detail

Stage

Nose cone	Fiberglass (1.85 g/cm³)	Ogive	Len: 75 cm	Mass: 1098 g
Payload		Dia$_{out}$ 2.5 cm		Mass: 3600 g
Landing legs		Dia$_{out}$ 25 cm		Mass: 3300 g
Body tube	Fiberglass (1.85 g/cm³)	Dia$_{in}$ 24.7 cm Dia$_{out}$ 25 cm	Len: 300 cm	Mass: 6499 g
Fin set (4)	Acrylic (1.19 g/cm³)	Thick: 0.5 cm		Mass: 960 g
Ascent Motor		Dia$_{out}$ 17.6 cm		Mass: 32500 g
Descent Motor		Dia$_{out}$ 13.2 cm		Mass: 5500 g

Fig.3.3 Rocket stability and part details

IV. RESULTS AND DISCUSSION

Based on the requirement of reaching an altitude of 10km with a payload capacity of 3.6 kg and then to achieve soft landing, two solid rocket motor (ascend and descend) with Ammonium perchlorate composite propellant has been designed and their performance characteristics were calculated. It was found that the best total mass of the rocket to satisfy this need was 53.5 kg that includes 26.348 kg + 3.711 kg of fuel. The flight simulation has been carried out in MATLAB® and Simulink® which resulted in the rocket reaching to desired altitude and then landed safely (zero velocity when altitude is zero). These designs were developed considering the control aspects of the rocket to achieve thrust vectoring for solid rocket motor.

Future research includes development of a nonlinear dynamical model of the rocket and designing of controller so as to implement thrust vector control for the designed rocket.

IV. ACKNOWLEDGMENT

We thank our project guide Asst. Prof. Anutha M [Aeronautical Department, DSCE] who provided insight and comments that greatly improved the manuscript. We thank our co-guide, Dr. R Pandiyan for sharing his pearls of wisdom regarding Rocket stability and expertise that greatly assisted the project.

We would also like to show our gratitude to our HOD, Dr. Hareesha N G for his encouragement throughout the project.

REFERENCES

[1] H. Platzek., 1979, Preliminary solid rocket motor design techniques, NWC technical memorandum, 2953: 1-42.
[2] George P. Sutton., “Rocket Propulsion Elements,” John Wiley & Sons, Inc., New York, 1964.
[3] Siddanth Singh., 2013, Solid Rocket Motor for Experimental Sounding Rockets. AISA, 3(3): 199-208.
Eric Hardester and Philip Kinghorn., “Rocket Fin Design,” February 15, 2013.
[4] D. E. Coats., N. S. Cohen., J. N. Levine., D. P. Harry III., G. R. Nickerson., C. F. Price. And T. J. Tyson., 1975, A computer Program for the prediction Of Solid Propellent Rocket Motor Performance, National Technical Information Service., Vol. 1
[5] Anoop Thankachen and Santosh Kumar., “Design Optimization and Analysis of Rocket Structure for Aerospace Applications,” International Journal of Engineering Trends and Technology (IJETT), Volume-24, Number-6, June 2015.
[6] M. S. EL-Wazerya., M. I. EL-Elamya., and S. H. Zoalfakarb., “Mechanical Properties of Glass Fiber Reinforced Polyester Composites,” International Journal of Applied Science and Engineering, 2017. 14, 3: 121-131.
[7] Md Akhtar khan1., B.K Chaitanya., Eshwar Reddy Cholleti., “Conceptual Design and Structural Analysis of Solid Rocket Motor Casing,” IJCER, ISSN €: 2250 – 3005., Volume, 07., Issue, 02., February, 2017.