Conceptual Design of Two Stage Rocket to cross Karman Line

A Journey Towards the Unknown and Infinite Possibilities

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Declaration

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This research internship work entitled "Conceptual Design of a Two Stage Rocket to cross the Karman Line" has been carried out by us under the guidance of **Mr Rakeshh Mohanarangan**, Propulsion Engineer, **Ms Rashika S N**, Space Engineering student at Politecnico Di Milano and Logistics Officer of the Science team at Mission- Asclepios, **Mr Vishnuvardhan Sakthibala**, Master of Science student at Politecnico Di Milano and **Mr Mahesh P**, Head of Innovation at Space Education Research and Development.

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Abstract

Humanity's interest in space exploration has grown by leaps and bounds since the moon landing on 20th July 1969. Over 50 years later, rocketry has become far more accessible than ever before, owing to technological advancements in materials, fabrication, controls and other rocket subsystems. Challenges to space exploration, such as affordability and efficiency, are being overcome with techniques such as the use of composites, 3D printing, and computer simulation.

This steady development in rocket technology is being met with an equally steady growth in the demand for space access. One of the biggest problems humankind is facing today, climate change, is causing us to look beyond earth for solutions. This requires extensive research and data collection from the atmosphere and beyond, which is why small scale experimentation rockets are essential.

This project is to demonstrate the conceptual design of a two stage sounding rocket with a solid propellant motor capable of surpassing the Karman Line. The rocket that we have designed can take a payload to the Karman Line and sustain it above it for few minutes to perform scientific experiments. Our project aims at making space travel more affordable by recovering and reusing the stages.

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1 Introduction

The endeavour to launch rockets to the Karman Line requires precision engineering where speed, power, and accuracy must all come together under high-stakes scenarios. There are four crucial areas of consideration when it comes to rocket engineering; Reliability, Cost, Safety, and Reusability. Lately, despite many launch mandates for scientific research's using sounding rockets, the prospects of launches are regulated because of high-cost of rocket launches.

Rockets that reach space and then return to Earth are classed as suborbital rockets. They are usually sounding rockets ('sounding' refers to taking measurements). Sounding rockets are one or two stage solid propellant rockets used for probing the upper atmospheric regions and for space research. They also serve as easily affordable platforms to test or prove prototypes of new components or subsystems intended for use in launch vehicles and satellites. Sounding rockets are generally used to do experiments in region of space between 50km to 700km where it is very difficult for traditional balloon to reach altitude. They are used to do investigation such as detection of the solar activity and anomalies, analysis of the constituents of the upper atmosphere, thermal analysis on new materials, and generally, measurements of the space surrounding the Earth. It has capability of achieving microgravity for experimental payload without man to perform experiments, and this reduces the cost of mission.

A vast number of sounding rocket has been designed such as American Black Brant and Terrier-Orion, European Maxus, Raxus and Taxus, and Swedish Maser. Current sounding rocket design has lots of disadvantages including high operational cost, extreme G-Loads placed on Payloads, toxic propellant combination and lack of reusability. These disadvantages have significantly curtailed the ability of students, researchers, and corporations to conduct experiments and research in the high—atmospheric and near—space region around Earth. Considering the emerging market for commercial spaceflight, enabling efficient access to the region in the vicinity of the Karman Line is of paramount importance.

1.1 | Motivation

On April 21, 2019, the University of Southern California's Rocket Propulsion Laboratory (USCRPL) became the first undergraduate group to launch a completely student-designed and student-built rocket that crossed the Karman Line, defined by the World Air Sports Federation as 100 kilometres above Earth's sea level. Their rocket, the Traveler IV, reached 103,632 meters, with a margin of error of +/- 5120.64 meters (340,000 feet +/- 16,800 feet), into the air, completing the organization's 15-year mission to be the first college group to reach outer space.

1.2 | Aims and Objectives

The main objective of this present work is to design a Two Stage Sounding Rocket with solid motor which can reach the Karman Line, 100Km above sea level, and to design a rocket such way that it can take payload of 1 kg above Karman line and sustain it there for few minutes to perform scientific experiments. The Aim of this project is to make rocket stages recoverable and reusable to reduce the cost of the flight.

2

Literature Survey

Gorge P. Sutton [1] has introduced to basic principles of rocket propulsion for guided missiles, space flight, or satellite flight in his book 'Rocket Propulsion Element'. It describes fundamentals of rocket propulsion, its essential technologies, its key design, systems design, propellants, applications, manufacturing technologies and solid propellant rocket motors. It gives detailed information about different propellant like PBAN, Lox, and FIOx etc. It also describes Final lift off mass and stage masses calculations, drag calculations and design procedure

Adam Okninski [2] has presented a methodology for finding close-to-optimal, in terms of launch mass minimization, design configurations for small unguided sounding rockets in his paper. A numerical, multidisciplinary approach is used. During the optimization process vehicle sizing and corresponding aerodynamics modelling is done.

Adde Y. Alemayehu et al [3] has presented design of solid rocket propulsion for a sounding rocket. It shows different materials for different parts of the propulsion system based on density, cost and availability. In this paper, the design results show that the mass of propellant is 960kg with volume of 0.546 m3 in the length of 1.944 m.

Merchant Coleman [4] has demonstrated the complete design and construction of two flight ready vehicles capable of surpassing the Karman line, to be flight tested at SpacePort America in New Mexico in May of 2019. It provides insight on the materials and structural design employed in the sounding rocket constructed. It gives detail about parts of sounding rocket like Aerodynamic design, Booster motor, Booster fincan, Interstage coupling, Airframe and Nosecone and Recovery system design.

XUE YU et al [5] have described a brief introduction to the principle, mechanism and method of separation between stages. It gives detailed study on hot separation and cold separation.

Gerald Hagemann et al [6] has presented data of nozzle design of rockets, types of nozzles like conventional nozzle dual-bell nozzle etc., method of calculation of nozzle pressure, area ratio and mathematical and graphical analysis methods. It also contains design parameters of different nozzle like toroidal plug nozzles.

W Gurkin [7] has described the specific design features and environmental factors to be taken into account while designing a high altitude sub-orbital rocket in his paper 'The NASA Sounding Rocket Program and Space Sciences'.

L. Pepermans et al [8] have presented Design of a Parachute Recovery System for the Stratos III Student Built Sounding Rocket; it contains the type of recovery system such as no drogue, drogue parachute and drag augmentation system.

Dr Darrell Guillaume et al [9] have demonstrated the design, fabrication, launch, and recovery of a supersonic experimental sounding rocket with a target apogee of 25,000 feet above ground level. The configuration of the rocket will consist of a booster stage with a second stage, designed to carry a ten pound payload

Bollerma M et al [10] have described about super Loki sounding rocket. The Super Loki Dart vehicle has been designed as a low cost replacement for the Cajun Dart for high altitude wind measurements. The goal of the Super Loki system was to achieve an altitude of a t least 95 km at less than one-half the cost of the Cajun Dart. A further goal was versatility of the Super Loki rocket motor so that it could be used to propel darts of various sizes containing various kinds of meteorological payloads

Marco Pallone et al [11] have presented a methodology in order to design, model, and evaluate the performance of new sounding rockets in paper. A general configuration composed of a rocket with four canards and four tail wings is sized and optimized, assuming different payload masses and microgravity durations. The aerodynamic forces are modeled with high fidelity using the interpolation of available data.

Lars Pepermans et al [12] have described the various methods for parachute deployment, an overview of the advantages and disadvantage of the various systems. The article provides a recommendation of when to use which deployment system.

David Keeports [13] has presented numerical calculation of model rocket trajectories, also presented forces upon a model rocket, calculations for vertical launch, two dimensional trajectories and three dimensional trajectories with mass drag and lunch angle variation.

Blazej Marciniak et al [14] have described an overview of the development of the ILR-33 "Amber" sounding rocket designated for microgravity experiments. The proposed design enables performing experiments in microgravity for almost 150 seconds with an apogee over 100km, also described the rocket structure and the vehicle's capabilities. The proposed design utilizes a hybrid rocket motor with High Test Peroxide as an ox-

idizer along with two reusable solid rocket boosters. The early phase analysis of the rocket configuration and propellant considerations are also presented in the paper.
Eric T. Pillai [15] has presented a simulation—validated design for a sounding rocket that will enable collegiate teams to surpass the Karman Line for the first time in history while fulfilling three primary requirements critical to allowing collegiate teams to reach space. The design criteria include 1) the rocket must be capable of reaching an altitude of at least 100 km, 2) the rocket must be relatively easy to manufacture and free of toxic materials, and 3) the rocket must be able to return telemetry, video, and data from on board experiments.

3

Methodology

3.1 | Mission Design

3.1.1 | Material selection

The selection of the materials to be used in the rocket was done keeping the following things in mind - majority of the rocket had to be recoverable and reusable, and the mass budget requirement had to be met. For this reason, titanium alloy Ti-6Al-4V was chosen as the material for a majority of the rocket, i.e., the outer shell, nosecone, tank and nozzle, owing to its high strength-to-weight ratio [1].

The fins, aside from the strength requirement and mass budget constraints, needed to be stiff to minimise flutter and aeroelastic damage especially in the supersonic regime, so the material chosen was CFRP (Carbon-Fibre Reinforced Polymer) [2].

The interstage coupling is a heavily stressed region, which meant that its reusability would be low as compared to the other parts of the rocket, so to save on cost, aluminium alloy AA2024 was chosen [3].A CFRP shroud was chosen to cover this structure to minimise form drag.

3.1.2 | Delta V and Mass estimation

The rocket designed in this project has an objective to reach the Karman Line (100 kilometres vertically above sea level) with a payload of 1-kilogram mass and sustain flight beyond the KL for a minimum of 180 seconds and perform scientific experiments[4]. The initial factor that has to be considered to design such a rocket is the delta V requirement. Every mission has specific delta V requirement, which depends upon the apogee to be attained, gravity losses and drag losses.

FACTORS	DELTA V
Height	1400.7 m/s
Gravity Losses	2452.5 m/s

Table 3.1: delta V contribution

The final obtained delta V is 3853.2 m/s.
Note: The drag loss is not considered and it is remunerated by the increasing acceleration (whose computation is beyond the scope of this paper.[5])

The mass of rocket such as Lift-off mass, stage mass and propellant mass can be estimated by using Rocket equation.The Tsiolkovsky rocket equation,

$$\Delta v = v_e \ln \frac{m_0}{m_f} = I_{sp} g_0 \ln \frac{m_0}{m_f}$$

Using this equation, selected specific Impulse and Dry mass of rocket, Mass of propellant for both stages and total lift-off mass were estimated. Here, the dry mass of rocket for both stages is assumed to be 30kg to make light weight rocket design.

3.1.3 | Thrust and Altitude estimation

Thrust is the force which moves the rocket through the air, and through space. Thrust is generated by the propulsion system of the rocket through the application of Newton's third law of motion; For every action there is an equal and opposite re-action. In the propulsion system, an engine does work on a gas or liquid, called a working fluid, and accelerates the working fluid through the propulsion system. The re-action to the acceleration of the working fluid produces the thrust force on the engine. The working fluid is expelled from the engine in one direction and the thrust force is applied to the engine in the opposite direction.[6]

Considering estimated mass and minimum thrust to weight ratio for the rocket to be greater than 1, Required thrust was selected for both stages 4200N for stage 1 and 2700N for stage 2. Burnout time for rocket is depended on specific impulse, mass of rocket and thrust value of rocket.The burnout time is given by,

$$\Delta t = I_{sp} g_0 (m_0 - mf) \frac{1}{T}$$

Using burnout time,thrust and mass of rocket, acceleration of rocket can be derived,

$$a = (T - Mg)\frac{1}{M}$$

Also, the burnout height of stages can be derived by[7],

$$h_b = g\left[-t_b I_{sp}\frac{\ln\frac{m_0}{m_f}}{\frac{m_0}{m_f}-1} + t_b I_{sp} - \frac{1}{2}{t_b}^2\right]$$

As rocket continues motion in vertical direction, Velocity will be increasing; the initial velocity at lift-off time is always zero,and the final velocity of rocket can be calculated using burnout time, burnout height and acceleration of rocket. The following table shows vehicle parameters, engine parameters and height estimation of rocket.

STAGE 1

Vehicle Parameters	
Initial Mass	258kg
Final Mass	106.1kg
Propellant	151.85kg
ΔV	1975.7m/s
Acceleration	6.5 m/s^2
Engine Parameters	
Thrust	4200N
ISP	227s
Mass flow rate	1.89kg/s
Burnout time	80.51s
Altitude Estimation	
Initial Velocity	0m/s
Final Velocity	520.8m/s
Burnout Height	36186.1m

Table 3.2: Stage 1 Parameters

STAGE 2

Vehicle Parameters	
Initial Mass	76.15kg
Final Mass	31.33kg
Propellant	44.82kg
ΔV	1975.7m/s
Acceleration	25.6m/s 2
Engine Parameters	
Thrust	2700N
ISP	227s
Mass flow rate	1.21kg/s
Burnout time	36.97s
Altitude Estimation	
Initial Velocity	520.8m/s
Final Velocity	1468.8m/s
Burnout Height	60696m
Final Height	170661.3m
Cruise time	149.73s

Table 3.3: Stage 2 Parameters

Liftoff	T = 0	H = 0m
First stage burnout	T + 80.51s	H= 36.186km
First stage separation	within T + 81.51s	H = 36.1865km
Second stage ignition	T+ 81.51s	H = 36.187km
Second stage burnout	T + 118.48s	H = 60.696km
Karman line	T + 177.88s	H = 100,000km
Apogee	T + 268.21s	H = 170.661km

Table 3.4: Time stamps

3.2 | Propulsion System

Propulsion is achieved by applying a force to a vehicle, that is, accelerating the vehicle or alternatively maintaining a given velocity against a resisting force. This propulsive force is obtained by ejecting propellant at high velocity.

In solid propellant rocket motors, the propellant to be burnt is contained within the combustion chamber or case. The solid propellant charge is called the grain and it contains all the chemical elements for complete burning. Once ignited, it usually burns smoothly at a pre-determined rate on all the exposed internal surfaces of the grain.

Each stage of a multistage launch vehicle is essentially a complete vehicle in itself and carries its own propellant, its own rocket propulsion system, and its own control system. Once the propellant of a given stage is expended, the dead mass of that stage is no longer useful in providing additional kinetic energy to the succeeding stages. By dropping off this useless mass it is possible to accelerate the final stage with its useful payload to a higher terminal velocity than would be attained if multiple staging were not used.[8]

3.2.1 | Propellant

Before moving forward, a right solid propellant that contents the requirements has to be chosen. After a literature survey and analyses, Cherry Limeade which is a specific combination of Ammonium Perchlorate and Aluminium having properties as mentioned below was selected as it had all the useful parameters readily available, required for further calculations, comparatively.

Performance and combustion properties:

- Density: 1688 kg/m3
- a: 4.404x10-5
- n: 0.327
- ISP: 227 secs
- Ratio of specific heats: k = 1.21
- Molar product mass: M = 23.67g/mol
- Universal gas constant: R = R′/M = 351.077 N-m/kg-K
- Burn temperature: 3500 K

Formula:

INGREDIENT	PERCENTAGE
Binder	17.10
Castor oil	0.30
PDMS	0.05
Triton X100	0.05
Al	7.50
200 AP	65.50
90 AP	9.50

Table 3.5: Propellant mixture and their mass fractions

Total	100%
Solids	82.50%
Metals	7.50%
AP	75.00%

Table 3.6: Metal and non-metal fractions

INGREDIENTS	PERCENTAGE
HTPB	10.884
IDP	4.275
MDI	1.942
Castor oil	0.30
PDMS	0.05
Triton X100	0.05
Al	7.50
200 AP	65.50
90 AP	9.50

Table 3.7: Sample formula (assuming a binder percentage of 25, EW of 1250 for HTPB, 185 for MDI, and 164 for castor oil)

Procedure:

1. Add HTPB, IDP, and Castor Oil
2. Mix for 5 minutes

3. Add AL
4. Hand mix until wetted out, machine mix for 10 minutes
5. Add PDMS and Triton X-100
6. Mix for 30 minutes
7. Vacuum for 45 minutes
8. Repeat until all 200 AP is in:
    a. Add a third of the original mass of AP
    b. Mix for 1 minute
    c. Scrape down
    d. Mix for 10 minutes
9. Add the 90 AP
10. Mix for 1 minute
11. Scrape down
12. Mix for 45 minutes
13. Add curative
14. Mix for 20 minutes
15. Vacuum for 45 minutes

Method of preparation:

1. Arrived at a formula that performed drastically better than XB (+32% ISP) and slightly better than OW (+6.5% ISP) while also improving density over OW (+6%).

2. An 80.7% solids propellant was significantly more pourable. Slight tweaks to the OW procedures were used for CL at 82.5% solids and the resulting propellant was still easily pourable.

3. 400 AP was originally included for this goal, but we believe that it contributed to the erosivity.

4. Though higher burn rates are typically associated with higher ISP, we were able to design our high aspect ratio motor to have tighter cores and thus higher volume loading because it featured lower burn rate propellant and didn't suffer from erosive burning as much.

5. Test mix was done using HTPB on a single 98mm grain.

6. The lack of HX-752 did not require recharacterization as the formula did not have large (400u+) AP.

7. After the P motor static fire, an experimental mix was made using only the liquid ingredients of CL to see how they behaved and found that the curative interacted very quickly with some ingredient to form clumps that the mixer broke up within a few minutes of mixing. The mixture poured well and took about 2 days to harden fully to a

springy rubber.[9]

3.2.2 | Grain Design

Neutral Thrust was selected for the sake of simplicity. For this, Cigarette burning configuration of the grain was chosen.

From the Vieille's equation [10],

$$r = ro + aPc^n$$

where r is the burn rate, ro is a constant (usually taken as zero), a is the burn rate coefficient, and n is the pressure exponent. The values of a and n are determined empirically for a particular propellant formulation, and cannot be theoretically predicted.
The burn rate of the propellant was determined to be as 7.82mm/s.The following are the grain properties and dimensions were obtained from formulas.[8] An iterative calculations were done in order to design right configuration for selected thrust.

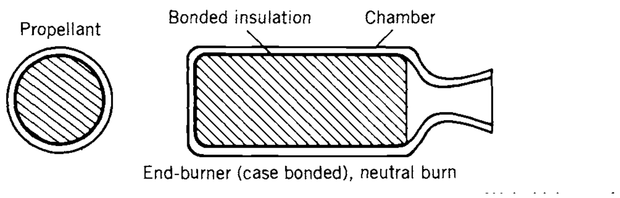

Figure 3.1: Propellant grain configuration[8]

Type of Thrust	Neutral
Grain burning configuration	End burning (Cigarette configuration)
Burn rate coefficient, a	4.4035E-5
Burn rate exponent, n	0.327
Chamber pressure	6.89 MPa
Rate of burning	7.82x10-3 m/s

Table 3.8: Grain Parameters

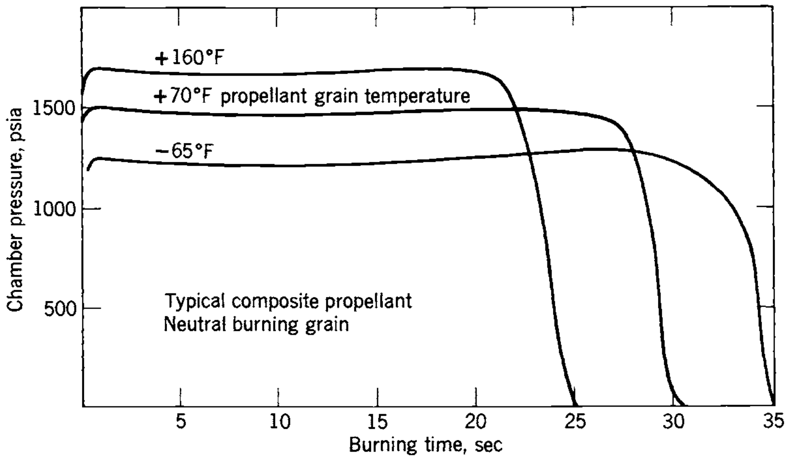

Figure 3.2: Effect of propellant temperature on burning time and chamber pressure for a particular motor. The integrated areas under the curves are proportional to the total impulse, which is the same for the three curves [8]

	Stage 1	Stage 2
Burn Area	$0.143m^2$	$0.092m^2$
Grain Radius	0.214m	0.17m
Grain Length	0.628m	0.288m

Table 3.9: Grain dimensions for both stages

3.2.3 | Nozzle Design

Aircraft exhaust nozzles serve two primary functions. First, they must control the engine backpressure to provide the correct, and optimum, engine performance, which is accomplished through jet area variations. Second, they must efficiently convert the potential energy of the exhausting gas to kinetic energy by increasing the exhaust velocity, which is accomplished through efficiently expanding the exhausting gases to the ambient pressure.

Designing Procedure:

1. Firstly, we have fixed to use the convergent-divergent nozzle. The CD nozzle is very efficient, as it gives high expansion rates.

2. Considering the nozzle as over expanded condition at the initial stage1, assumed that Pe=Pa/3 (Pe=exit pressure, Pa= ambient pressure) by using the relation between area

ratio (Ae/A*) and chamber pressure (Pc=6.9MPa), exit pressure(Pe) we got area ratio Ae/A* as 20.0287 for first stage.
Expansion Rate (Area Ratio): 3. As gain in the altitude, there is laps in ambient pressure

$$\varepsilon = \left(\frac{\left(\frac{2}{\gamma}+1\right)^{\frac{1}{\gamma}-1} \left(\frac{Pc}{Pe}\right)^{\frac{1}{\gamma}}}{\sqrt{\{\frac{\gamma+1}{\gamma-1} * (1-(Pe/Pc)\hat{}(\gamma-\frac{1}{\gamma}))\}}} \right)$$

so stage 2 operation is carried out as under expansion condition because ambient pressure is very less that even a nozzle is expanding at lower expansion rate the exit pressure is much higher than ambient pressure ,so we considered Pa=100 Pa and Pe=50000Pa and with same chamber condition (Pc=6.9MPa),calculated the nozzle area ratio i.e 14.8314.
4. Using 1dimensional flow isentropic relations and chocked condition , calculated the Throat parameters and effective velocity (C*), and mass flow rate. and then calculated Isp value using Thrust equation. the Isp value(230s) which is near to mission requirement Isp (227s).In this all process we considered the nozzle half divergence angle(alpha) 15 degrees with this divergence angle loss due to divergence is just about 1.7% , and finally we got 98.3% nozzle efficiency.

Isentropic Flow Mach Number Relationships:	
Stagnation to Static Temperature	$\frac{T_t}{T} = \left(1 + \frac{\gamma-1}{2} M^2\right)$
Stagnation to Static Pressure	$\frac{P_t}{P} = \left(1 + \frac{\gamma-1}{2} M^2\right)^{\frac{\gamma}{\gamma-1}}$
Mass Flow Rate	$\dot{m} = \frac{P_t}{\sqrt{RT_t}} A\sqrt{\gamma}\, M \left(1 + \frac{\gamma-1}{2} M^2\right)^{\frac{\gamma+1}{2-2\gamma}}$
Stagnation to Static Density	$\frac{\rho_t}{\rho} = \left(1 + \frac{\gamma-1}{2} M^2\right)^{\frac{1}{\gamma-1}}$
Arbitrary Area to Choked Area in Duct	$\frac{A}{A*} = \frac{1}{M} \left(\frac{\frac{\gamma+1}{2}}{1 + \frac{\gamma-1}{2} M^2} \right)^{\frac{\gamma+1}{2-2\gamma}}$

5. we have fixed nozzle half divergence angle(alpha) using trigonometric relations calculated the divergence and convergence lengths of nozzles for both stage 1 and stage 2.

Length of divergent Part :

$$L_d = 0.5(D_e - D_t)cot\alpha$$

6. For finding out the total mass of the nozzle for both stages we have considered material as Titanium alloy,that is Ti-6Al-4V.The thickness was considered to be 0.05 meters. The total weight to for the nozzle for stage 1 is 2.4365 Kg and stage 2 is 1.1141 Kg.

PARAMETER	STAGE 1	STAGE 2
Chamber Pressure	6900000 Pa	6900000 Pa
Chamber Temperature	3477.7 K	3477.7 K
Inlet Radius	0.213 m	0.17 m
Ambient Pressure	101325 Pa	100 Pa
Exit Pressure	33775 Pa	50000 Pa
Mass Flow Rate	1.89 Kg/S	1.21 Kg/S
Gamma	1.21	1.21
R	326	326
Area Ratio	20.028	14.831
Throat Pressure	3881501.6 Pa	3881501.6 Pa
Throat Temperature	3147.239 K	3147.239 K
Density	3.783 Kg/m^3	3.783 Kg/m^3
Velocity at Throat	1114.2083 m/s	1114.2083 m/s
Throat Area	0.000448m^2	0.000287m^2

Table 3.10: Nozzle parameters

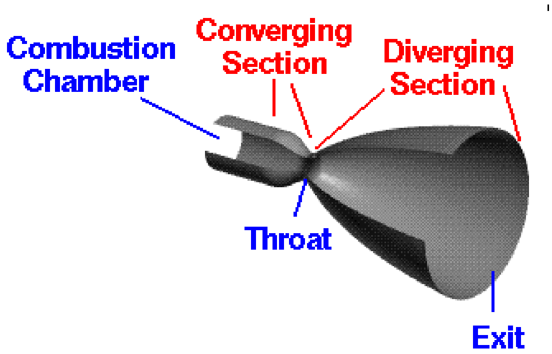

Figure 3.3: CD nozzle

PARAMETER	STAGE 1	STAGE 2
Exit Area	0.0089 m^2	0.0042 m^2
Exit Diameter	0.1069 m	0.0736 m
Throat Diameter	0.0238m	0.0190 m
Length of Divergent Part	0.1549 m	0.1017 m
Length of Convergent Part	0.0834 m	0.0664 m
Total Length	0.2383 m	0.1681m
Mach Number	3.801	3.587
Exit Temperature	1381.462 K	1478.795 K
Exit Velocity	2806.2548 m/s	2740.3299 m/s
Coefficient of Nozzle	1.357	1.363
Nozzle Divergence Angle	15 degrees	15 degrees
Loss due to Divergence	1.7%	1.7%
Nozzle Efficiency	98.3%	98.3%
Total Mass	2.436 Kg	1.114 Kg

Table 3.11: Nozzle Dimensions

3.3 | Structural Design

3.3.1 | Propellant tank design

The design process for the solid propellant tank was fairly simple - it began with assuming the tank to be a thin-walled cylindrical pressure vessel. The deciding factor in the thickness of the tank was the relation between hoop stress on the tank and the chamber pressure inside the tank, also known as Barlow's formula [(11)].

The material chosen for the tank was Ti-6Al-4V, owing to its high strength-to-weight ratio [(12)], i.e., it can hold up well in the harsh operating conditions that the tank is subjected to, while still being light. The thickness of a heat shielding liner [(13)] was also accounted for in the dimensioning of the tank, but further considerations of its properties are beyond the scope of this paper. For the chosen tank wall material and propellant, the properties we considered are in in table 3.12.

The formula used to calculate the thickness t of the propellant tank, is a rearrangement of Barlow's formula, as

$$t = \frac{PD}{2\rho}$$

Factors of safety of 1.33 and 3 were respectively accounted for in the values of the yield stress and chamber pressure P. The thickness thus calculated was for the material at

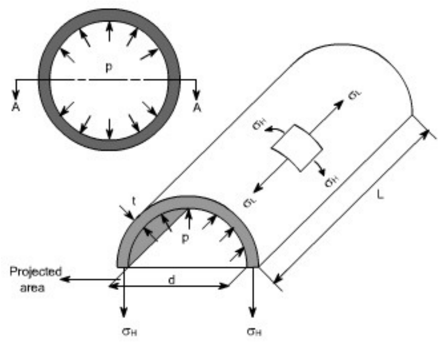

Figure 3.4: Thin walled pressure vessel stresses[14]

	STAGE 1	STAGE 2
MATERIAL DENSITY (kg/m3)	4430	4430
CHAMBER PRESSURE (MPa)	6.890	6.890
YIELD STRENGTH (MPa)	880	880
GRAIN DIAMETER (m)	0.427	0.340
LINER THICKNESS (m)	2.5E-03	2.5E-03
GRAIN HEIGHT (m)	0.628	0.288

Table 3.12: Material and propellant properties

yield point, operating at its chamber burst pressure. This estimation, along with the factor of safety, gave the structure a wide enough margin against failure (15).

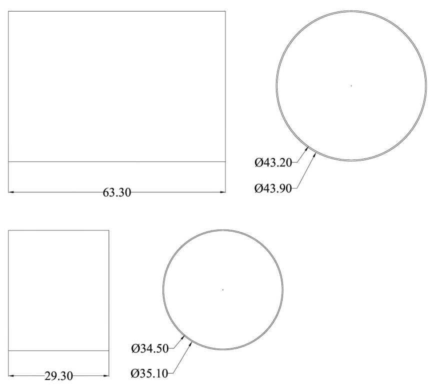

Figure 3.5: Tank dimensions (cm)

	STAGE 1	STAGE 2
TANK INNER DIAMETER (m)	0.432	0.345
TANK HEIGHT (m)	0.633	0.293
TANK MATERIAL VOLUME (m^3)	3.32E-03	1.00E-03
MASS(kg)	14.643	4.326

Table 3.13: Tank dimensions

3.3.2 | Outer shell design

The main consideration while designing the shell was the total force acting on the rocket. The forces acting on a thin walled cylinder such as the rocket's shell or propellant tank are the internal pressure and the longitudinal force. The internal pressure load caused due to combustion in the propellant tank would be withstood by the tank wall, so the remainder of the forces acting on the shell of the rocket were assumed to be axial forces [16]. These axial forces are the drag force and the inertial force.

Before analyzing the acting forces on the rocket, a preliminary calculation [17] was done for the shell to determine its applied compressive stress. To compute this value, a basic axial stress formula [18] relating the applied axial compressive force F and cross sectional area A was used,

$$\sigma_c = \frac{F}{A} = \frac{F}{\pi D t}$$

where D and t are respectively the cylinder's average diameter and thickness. At compressive yielding, the axial stress for Ti-6Al-4V is 970MPa [19]. The ratio R/t between the average radius of the cylinder R, and its thickness was taken as 300 [20], for which the maximum allowable load Fmax from the above formula was 841.49kN.

From the R/t taken for the Fmax calculation, the thickness of the shell was computed. The total length of the shell was taken as the total length of the space occupied by the components inside it, i.e., the lengths of the propellant tank, nozzle and packaged length of the parachute. The inner diameter of the shell was taken as the outer diameter of the widest component inside it, in this case, the propellant tank. The same methodology was followed to get the dimensions of the shell for both the stages.

Once the maximum allowable load on the structure and the structure's dimensions were calculated, the task was to ensure that the aforementioned axial loads due to drag and gravity, would take a value less than Fmax for this structure. To calculate drag

	STAGE 1	STAGE 2
INNER DIAMETER (m)	0.439	0.351
R/t	300	300
t (m)	8.56E-04	7.85E-04
OUTER DIAMETER (m)	0.441	0.353
LENGTH(m)	1.051	0.641
VOLUME(m3)	1.24E-03	5.56E-04
MASS (kg)	5.514	2.465

Table 3.14: Outer shell dimensions

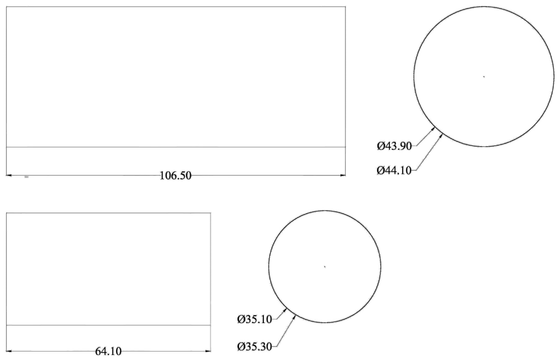

Figure 3.6: Outer shell dimensions (cm)

force, the standard formula [21] relating drag force FD, density , velocity v, reference area A and drag coefficient CD was used,

$$F_D = \frac{1}{2}\rho v^2 C_D A$$

For this rocket, for the value of density was chosen conservatively as the density at sea level, which is 1.225 kg/m3, v was taken as the maximum velocity of our rocket, which was Mach 3.8, for which the Drag coefficient is 0.9, and the A was taken as the cross sectional area of the widest part, in this case, the first stage. For these conservative estimates, the drag force was 308.137kN. To calculate inertial force, a simple application of Newton's second law of motion on the rocket's dry mass m at maximum acceleration a was done, for a m value of 60kg and a value of approx. 500m/s. This gave the inertial force value of 30kN. The total force acting on the rocket, calculated using conservative estimates, was hence

$$F = F_D + F_G = 308.137 + 30 = 338.137kN$$

This value is much less than the value of Fmax, which implies that the structure is safe for the calculated dimensions.

LOAD TYPE	LOAD (N)
Max. permissible load	
Axial compressive	841490.3
Actual load	
Drag force	308137.2
Inertial force	30000
Total force	338137.2

Table 3.15: Loads acting on the rocket

3.4 | Aerodynamic optimization and stability

3.4.1 | Nose cone design

For the design of the nose cone, the main factor to consider was minimizing drag for our specific mission. Since we were operating in the transonic to supersonic flow regime, the shape of the nose cone had to be chosen in such a way as to be efficient in minimizing drag for this regime. Keeping this mind, the Von Karman ogive was selected as the shape of the nose cone.[22]

The profile of the Von Karman nose cone, derived from the LD-Haack series nose cone, has a profile described by the equation [23]

$$y = \frac{R}{\sqrt{\pi}}\sqrt{\theta - \frac{sin(2\theta)}{2}}$$

$$\theta = arcos(1 - \frac{2x}{L})$$

BASE DIAMETER (m)	0.351
L/D	2
LENGTH (m)	0.706
MASS (kg)	2.045

Table 3.16: Nose cone dimensions

The equations for the Von Karman ogive profile were applied, using the previously derived dimensions and a fineness ratio, i.e., ratio between length and base diameter of the nose cone, of 2. This fineness ratio was decided based on a trade-off analysis between how much mass the nose cone would add to the rocket, and how much stability it would give. For this analysis, the rocket simulation tool OpenRocket v15.03 was used extensively, which helped come to a compromise between the two factors. OpenRocket was also used to determine the mass of the nose cone.

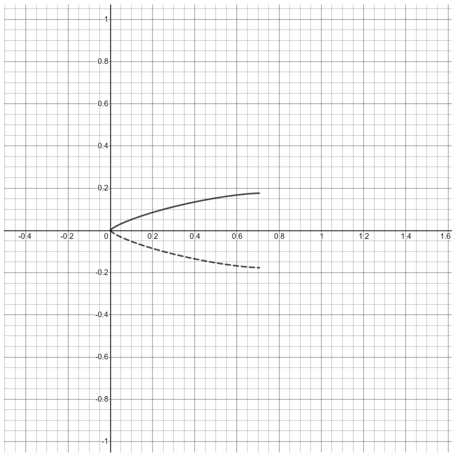

Figure 3.7: Nose cone profile

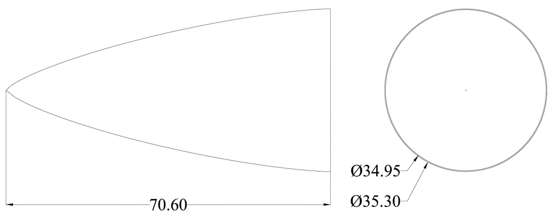

Figure 3.8: Nose cone dimensions (cm)

3.4.2 | Fin design

Fins on a rocket play a crucial role in its stability. Just like the other components, however, the fin dimensions and shape and to be chosen in such a way so as not to add too much to the overall dry mass of the rocket. More importantly, it had to suit the mission of the rocket. Again, since the flow regime this rocket would be operating in is transonic to supersonic, a suitable planform shape and dimensions would have to be chosen. For all the above considerations, a clipped-delta fin planform was chosen, [24][25] with a double wedge cross section [26].

For the dimensions of the fin, a general thumb rule [27] is followed,

$$C_{root} = 2D$$

$$C_{tip} = D$$

$$S = 2D$$

$$t = 0.1C_{av}$$

Where C(root) is the root chord, C(tip) is the top chord, S is the span, C(av) is the average chord, Dis the stage outer diameter and ti s the thickness of the fin. However, with these dimensions, despite the rocket being stable (0.6 caliber), the fins were unusually large and added much more mass than was necessary. In order to bring the rockets down to a reasonable size while maintaining at least 0.5 caliber of stability, the dimensions were scaled down to 0.375 times the original C(root), C(tip), S and t. This scaling down was done with the help of OpenRocket, where a number of different scaling factors were tried out until a reasonable mass and size was achieved for both stages.

	STAGE 1	STAGE 2
ROOT CHORD(m)	0.330	0.265
TIP CHORD(m)	0.165	0.13
SPAN(m)	0.330	0.265
THICKNESS(m)	0.004	0.004
MASS PER FIN(kg)	0.494	0.352
TOTAL FIN MASS(kg)	1.977	1.407

Table 3.17: Fin dimensions

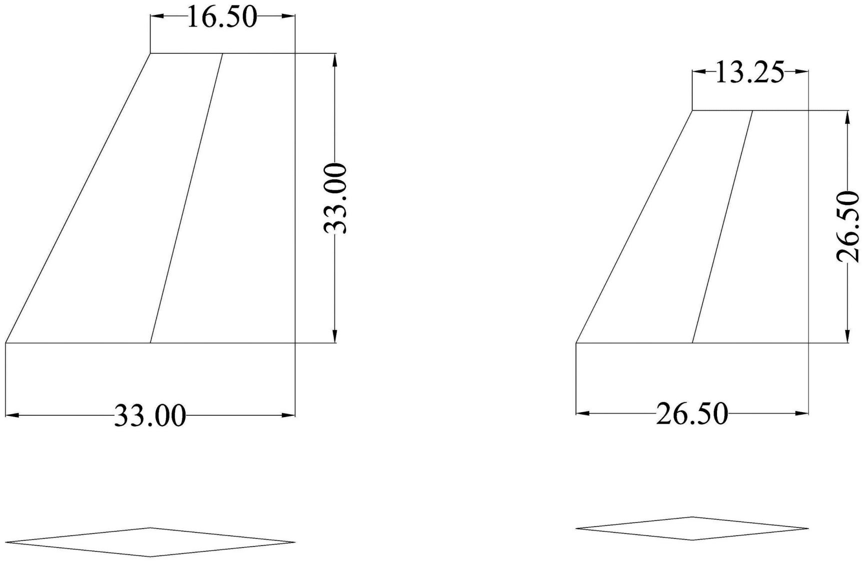

Figure 3.9: Fin dimensions (cm)

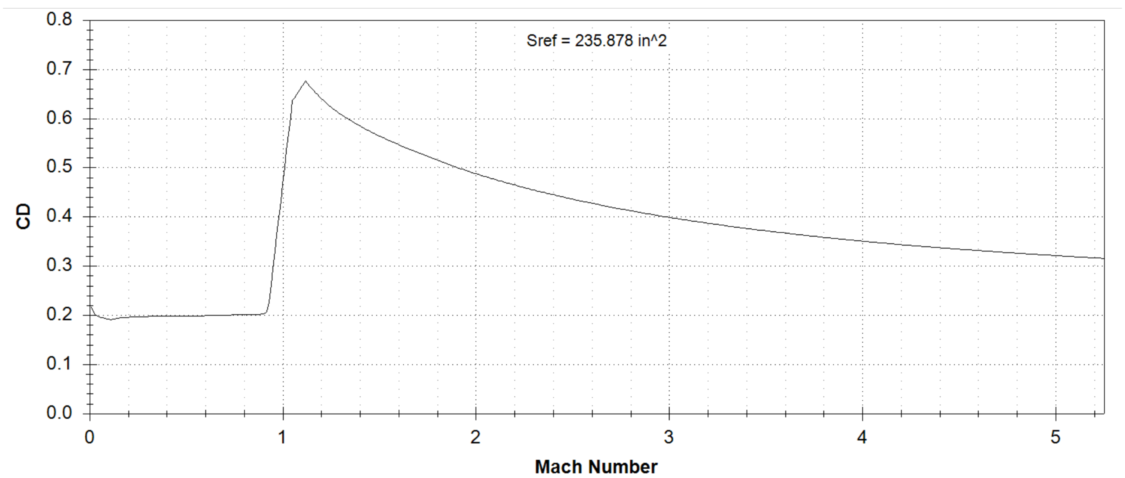

Figure 3.10: CD vs Mach number plot for rocket

Fig 3.10 illustrates the variation of drag coefficient acting on this rocket as a function of its Mach number. As shown, the maximum value of drag coefficient occurs at Mach 1.2, with a value of 0.68, and then declines steadily. At the rocket's peak Mach number, 3.8, the corresponding value of drag coefficient is 0.35.

After all the above calculations and distribution of components in the rocket, the stability attained is 0.539 caliber.

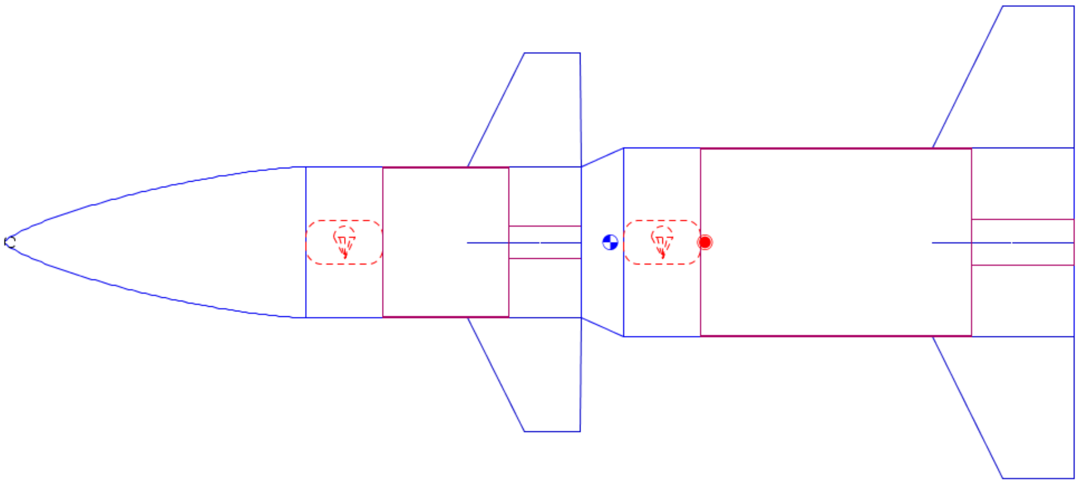

Figure 3.11: OpenRocket model of rocket with all components

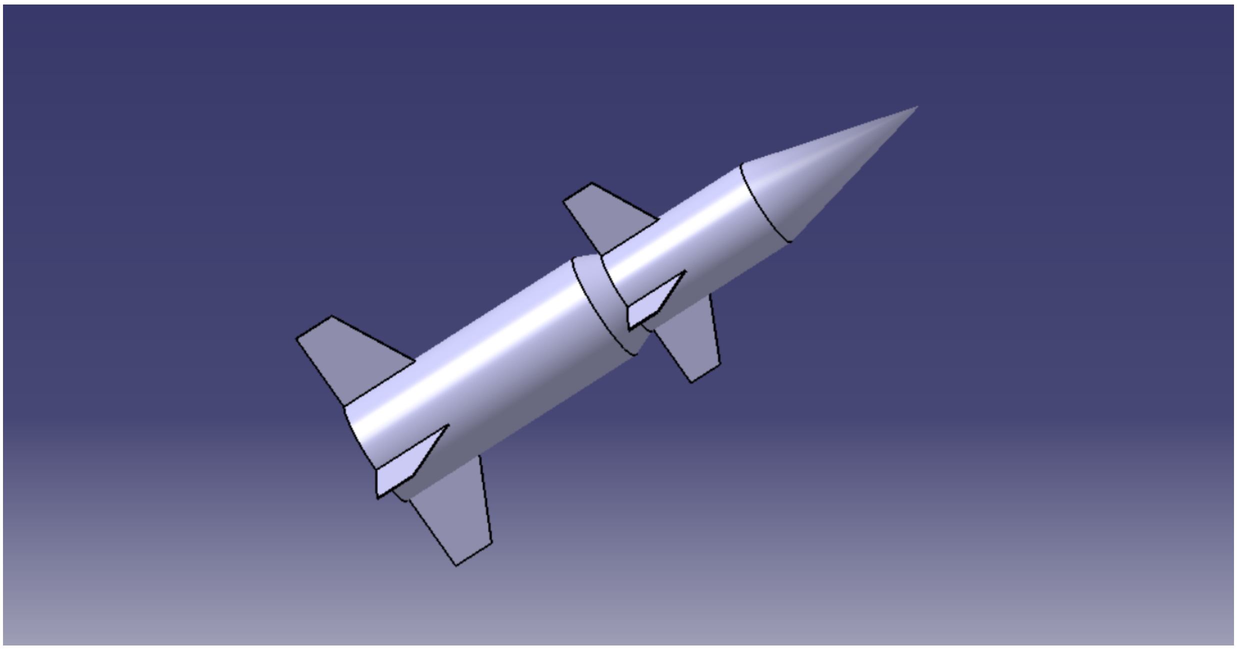

Figure 3.12: Simplified 3D model

3.4.3 | Mass Budget

The tables below present an overview of all the components besides the propellants, and how much of the dry mass each will constitute.
The total dry mass of the rocket is hence 45.814kg. Since the assumed dry mass of the

Stage 1	Mass(kg)	%	Stage 2	Mass(kg)	%
Propellant tank	14.643	24.405	Propellant tank	4.326	7.210
Nozzle	2.658	4.430	Nozzle	1.270	2.117
Outer shell	5.514	9.190	Outer shell	2.465	4.108
Transition Shroud	0.241	0.402	Nose cone	2.045	3.408
Fins	1.977	3.295	Fins	1.407	2.345
Separation system	1.300	2.167	Payload	1.000	
Total Stage I	30.317	50.529	Total Stage II	15.497	25.828

Table 3.18: Mass budget

rocket was 60kg initially, 14.186kg of mass remains unused for the rocket that has been designed. The mass of the electronic equipment other than the payload, however, has not been taken into consideration in this project. Any additional mass savings beyond this can be used in increasing the scope of the mission, such as apogee or flight time; or in carrying heavier or more sophisticated payloads.

3.5 | Interstage Separation

Generally, Interstage Separation is the process of separating the spent stage after its burnout which helps in reduction in mass of rocket and further improving the performance and efficiency of rocket in its further flight.

So as the name suggests above there are two main mechanisms in the interstage separation of our case,They are:
1. Hold and Release Mechanism(Devices that hold the two stages together and release whenever separation is needed.)
2. Separation Mechanism(Mechanism/Method by which Separation of stages occur.)

3.5.1 | Hold and Release mechanism

The basics of Flight separation mechanisms were known[28].When the research progressed on stage separation, the first and basic requirement or approach on which the further research started was the method or mechanism to hold the stages together and how to release them when needed. Considering the mass and power constraints in our mission, Mechanical Latches (with catch plates) are opted to hold two stages together. To be more specific, toggle latches are the choice.

The stages of rocket being of different diameter ,a small interstage sector in the form of a truncated cone is opted as the connector to compensate difference in diameters. A Carbon Fibre Shroud is placed over the interstage sector to reduce the effects of Form Drag.[29]

The bigger diameter end of this interstage sector is to be welded to the stage 1 and the smaller end will be attached to stage 2 with the help of Latches and catch plate. Latches are supposed to be fixed in the interstage sector and catch plates in stage 2. As Latches and catch plates are connected, both stage 2 and interstage sector are connected to each other. As the interstage sector is welded to stage 1 that implies stage 1 and stage 2 are connected too. Interstage sector is in the form of truncated cone. But, when this

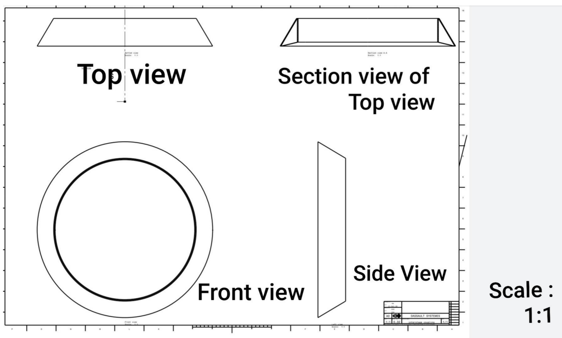

Figure 3.13: Interstage Sector

sector is to be joined to stage 2 by latches, It is like joining a linear part (stage 2 is linear-cylinder) and an angular part(sector is angular as it is a truncated cone). So, latch and catch plate has to be joined in an angular position which might be difficult. To compensate that, A small tube of height 6/7th of interstage sector's height is inserted into sector from upper end of sector and the upper end of tube is welded to upper end of sector. Now latch is in this tube inside the sector and is in line with stage 2 and catch plate is in stage 2.So we can have joining in a linear position. Stage 2 is inserted into this tube inside the interstage sector by a distance of just 1 or 2 mm for more grip and stability between stage2 and interstages sector.

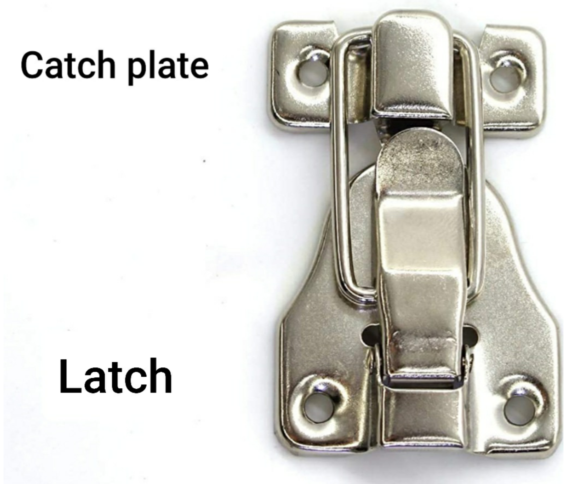

Figure 3.14: example sketch of Latch and catch plate

3.5.2 | Separation mechanism

To separate stages, Release mechanism of latches and Drag separation(Separation of stages by the Differential drag between the stages) subsequently is opted.Generally when a body is moving in a fluid, it will experience drag force. So similarly rocket experiences drag force too in its flight.
Drag Force is given by the formula[30] as,

$$F_D = \frac{1}{2} C_D \rho V^2 A$$

where, CD is drag coefficient.Taking into consideration the parameters in this equation, design of the rocket is done in such a way that the difference in drag between stage 1(including interstage) and stage 2(including nosecone) is optimum without any considerable effect on efficiency and performance of the rocket.

At the time of separation, an electric signal is sent to the surroundings of the latches that creates magnetic field at the end of the latches. This magnetic field pulls or pushes the ends of the latches which lead to release of latch from catch plate. Due to this, the bond between the 2 stages is released. Then, as there is excessive drag force on stage 1 relative to stage 2, this force pushes the stage 1 away from the stage 2. It acts like a jettison mechanism. As soon as there is considerably good separation (this happens in

a small time differential) ignition of stage 2 engine is done and it will shoot up by the thrust generation.
Drag force calculation:

	Stage 1	Stage 2
Drag Coefficient	0.346	0.272
Velocity at separation(m/s)	520.8	520.8
Area of stage in contact with air(m^2)	2.202	1.5212
Drag force(N)	711.15	386.24
Estimated Excessive Drag force on Stage 1 over Stage 2	324.91 N	

Table 3.19: Drag force calculation

3.6 | Recovery system

Recovery is the deployment of a primary recovery device that actively changes the physical configuration and it will also reduce the vertical descent rate of the rocket model dramatically whenever deployed. This device must be of sufficient size according to its weight of the model, so that the device will be capable of safely recovering the rocket. The active recovery device can include parachutes.

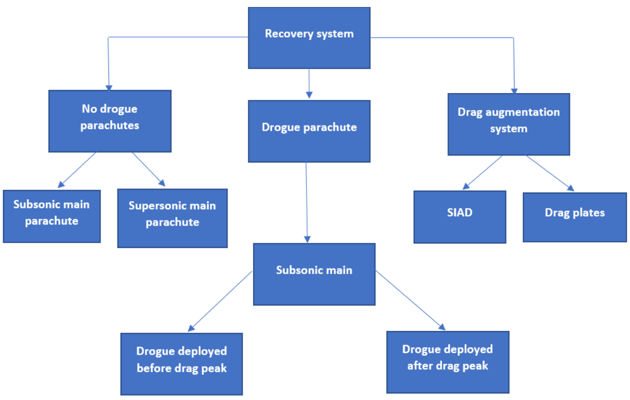

Figure 3.15: type of recovery system

the time of dual deployment, secondary recovery devices are used, the deployment of a secondary recovery device must actively change the configuration of the model in order to inhibit ballistic recovery and slow the decent rate. Hence, this will allow safe deployment of the primary recovery device. There is different type of recovery system like No Drogue parachute, Drogue parachute, drag augmentation system etc. which can be used as per the application, type of payload and weight of the outer shell or structure.

Figure 3.15 shows the type of recovery system. Here drogue parachute is used due to weight of payload and outer shell of structure, weight of system will be around 32-34 kgs so for tat drogue parachute will be used. With the help of drogue parachute, it will decrease the velocity of payload during free fall from apogee, after some time main chutes will be deployed so it will decrease the velocity of payload drastically and recover is safely. For drogue chutes its decent velocity will be 30m/s and for main chutes it will be 9.8m/s. if chutes reach these velocities, they can recover the payload safely. Need of recovery system in any rocket mission is to get experimental result which was done in the space or apogee. If there is no recovery system in rocket then payload will strike to earth and it will be damage.

Design consideration:
Area of chute canopy will be as follows,

$$A = \frac{2W}{\rho C_D V^2}$$

By using these formulas diameter of small drogue chute canopy will be around 1 meter and diameter of main chute canopy will be around 3.5 meter. Here decent velocity for drogue chutes will be 30m/s and 9.8m/s for main chutes.

Height selection criteria:

For the mission hemispherical shape of canopy was chosen for both due to an easy fabrication process. here decent velocity for both the stages is assumed, and based on that chute canopy area is calculated, for height of deployment expected height is around at 5km and 1km from Sea level for both parachutes respectively. If recovery system will be deployed and start working from height of 5km, at that height small drogue parachutes will be deploy so, that drogue parachute will be decreased the velocity of payload and structure also at time of free falling from apogee. After some time, main chutes will be deployed and recovery will be done. If main chutes deploy in range of height 5000 to 6000 meters it will achieve the decent velocity 9.8m/s and safely payload and structure will be recovered. It is calculated by motion equation and get approximation value of height range.

Approximately mass of whole recovery system for each stage is around 4 kg. total mass whole recovery system for rocket is 8kg dimension of parachute deployment sys-

	Chute type	Air Density	Decent velocity[m/s]	Drag coefficient	Diameter of canopy[m]	Weight [N]	Recovery mass [kg]
S1	drogue	1.225	30	0.6	1.063	294	3.968
	main	1.225	9.8	0.6	3.256	294	3.968
S2	drogue	1.225	30	0.6	1.156	346.92	3.968
	main	1.225	9.8	0.6	3.537	346.92	3.968

Table 3.20: parachute recovery system calculated data

Component	Material	Approximately mass
3 ring system	Steel and nylon webbing	200g
Wire cutter	NA	80g
Main chute canister	Glass fiber	150g
Cart insert	Polylactide	24g
Spring	Steel	55g
Lid	Polylactide	21g
Cover lid	Cork	26g
Wire clamp	Aluminum	13g
wire	Twaron	2g

Table 3.21: Caption

tem packaging is height 180mm, width 100mm and length 70mm.the chute system will be placed at top of the stages, so it can deploy easily. In second stages it will be at top of stage upper to payload. Because nosecone will not be separate so that place is feasible for recovery system.

Launch and Recovery site selection:
For launching the rocket some criteria will be there, like what is the effect of magnetic field at the launching site and what will be average environment condition. If average speed of wind is high that site is not beneficial for launching because it will affect the trajectory of rocket and change the direction of the same. At time of selection of launching site there will also be consider the recovery site criteria. If recovery will be done in water then launching site should be near coastal area. Because it will be easy to recovery them in coastal area. Recovery site should be in the range of 15 to 20 km area. Thumba Equatorial Rocket Launching Station is currently used for launching sounding rockets.

Component	Material	Approximately mass
Canopy	Twaron	200g
Bag	Paratex	50g
Suspension lines	Technora	84 g
Riser	Technora	250 g
Swivel	NA	73 g
Ring	Aluminum	7 g

Table 3.22: overview of drogue parachute system

Component	Material	Approximately mass
Canopy	Ripstop nylon	400g
Bag	Paratex	80g
Suspension lines	Spectra	32g
Riser	Nylon	40g
Swivel	NA	73g
Link	Aluminum	32g
Pilot chute	Nylon	80g
Soft link	NA	10g
Bridle line	Spectra	1g
H line	Spectra	1 g

Table 3.23: detailed overview of main chute recovery system

The presence of a strong 'equatorial electrojet' current over Thumba, which was also very close to the geomagnetic equator, made it an idea site for the launch of sounding rockets.

4

Conclusion

The expected result of reaching the Karman line and sustaining above it for more than 3 minutes to aid micro-gravity scientific experiments is achieved with full confidence. The paper has not dealt with guidance systems, electronics, and battery technology and flight computers. With present advancements, it is safe to assume that these are readily available to be equipped and installed at a short amount of time.

It is to be noted that the rocket is practically feasible and with improved designing techniques and analysis, the design can be perfected. Future developments can be based on improving fuel efficiency, stability and correcting trajectories. The design gives a large flexibility to change parameters and thus making it advantageous to carry heavier payloads to higher altitudes for longer experimentation periods.

The design will aid upcoming rocketry projects to build their own and thus achieving favourable conditions to make space affordable and available to everyone.

5

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