DESIGN OF A TWO STAGE ROCKET WITH AEROSPIKE ENGINE

ASB 4341- DESIGN PROJECT-1 REPORT

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ABSTRACT

This project describes the design of a double stage Rocket with an aerospike engine, one designed with traditional subsystems for structural, avionics, combustion chamber and recovery integrated to give a desired altitude. Some key influential parameters and intricate behaviors are analyzed through sensitivity to the stage separation conditions. Selection of tentative parameters like height, gross weight, payload weight, apogee and burn time has been carried out by a conceptual study of fifteen single/multi stage rockets. J-2T-250K Toroidal aerospike engine was used for this design. Aerospike nozzle has overall better performance than the conventional bell-shaped nozzle and higher efficiency at lower altitudes. The material for the nose cone, the payload tube, transition, recovery or parachute tube and fins were selected based on the density and weight constraints. Linux and dual-core x86 processors were used onboard. Carbon composites (Si-C coating) was used for designing the nose cone, the body and the fin set. The combustion chamber, clamps, and nozzle were designed by making use of low carbon steel. Because of the high temperature and pressure being generated from the combustion of propellant, low carbon steel was suggested.

Keywords: *Toroidal aerospike engine, ogive nose cone, grid fins, LEO payload, specific impulse, thrust, mach number, burn time, apogee, propellant, fairings, retro thrust, orbit, lift, ascent, linear aerospike engine, aluminum alloys, gyroscope, reaction wheel.*

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LIST OF SYMBOLS AND ABBREVIATIONS

• M	- Rocket Mass
• Δv_R	- Increase in velocity of rocket
• $m°$	- Rate of mass discharge in exhaust
• Δt	- Time Interval
• v_e	- Effective exhaust velocity
• F	- Force
• π_{PL}	- Payload Mass Ratio
• M_{PL}	- Payload Mass
• M_{tot}	- Total Mass
• t_{bo}	- Burn Time
• T	- Thrust
• ISP	- Specific Impulse
• g	- Acceleration due to gravity
• SLI	- Space Launch Initiative
• SSTO	- Single-Stage-to-Orbit
• LEO	- Low Earth Orbit
• OMS	- Onboard Maneuvering System
• SABRE	- Synergistic Air Breathing Rocket Engine
• SOMA	- SKYLON Orbital Maneuvering Assembly
• VTOL	- Vertical Takeoff and Landing
• VTOVL	- Vertical Takeoff and Vertical Landing
• ROMBUS	- Reusable Orbital Module Booster & Utility Shuttle
• PSLV	- Polar Satellite Launch Vehicle
• TAT	- Thrust Augmented Thor
• LTTAT	- Long Tank Thrust Augmented Thor

- SL - Sea Level
- Vac - Vacuum

CHAPTER 01

INTRODUCTION TO DESIGN

INTRODUCTION TO DESIGN

Modern aircraft are a complex combination of aerodynamic performance, lightweight durable structures and advanced systems engineering. Air passengers demand more comfort and more environmentally friendly aircraft. Hence many technical challenges need to be balanced for an aircraft to economically achieve its design specification. Aircraft design is a complex and laborious undertaking with a number of factors and details that are required to be checked to obtain optimum the final envisioned product. The design process begins from scratch and involves a number of calculations, logistic planning, design and real-world considerations, and a level head to meet any hurdle head on.

Aerodynamics is the study of how air flows around an airplane. In order for an airplane to fly at all, air must flow over and under it's every airplane goes through many changes in design before it is finally built in a factory. These steps between the first ideas for an airplane and the time when it is actually flown make up the design process. Along the way, engineers think about four main areas of aeronautics: *Aerodynamics, Propulsion, Structures and Materials, and Stability and Control.*

Wings. The more aerodynamic, or streamlined the airplane is, the less resistance it has against the air. If air can move around the airplane easier, the airplane's engines have less work to do. This means the engines do not have to be as big or eat up as much fuel which makes the airplane more lightweight and easier to fly. Engineers have to think about what type of airplane they are designing because certain airplanes need to be aerodynamic in certain ways. For example, fighter jets maneuver and turn quickly and fly faster than sound (supersonic flight) over short distances. Most passenger airplanes, on the other hand, fly below the speed of sound (subsonic flight) for long periods of time.

Propulsion is the study of what kind of engine and power an airplane needs. An airplane needs to have the right kind of engine for the kind of job that it has. A passenger jet carries many passengers and a lot of heavy cargo over long distances so its engines need to use fuel very efficiently. Engineers are also trying to make airplane engines quieter so they do not bother the passengers onboard or the neighborhoods they are flying over. Another important concern is making the exhaust cleaner and more environmentally friendly. Just like automobiles, airplane exhaust contains chemicals that can damage the earth's environment.

Structures and Materials is the study of how strong the airplane is and what materials will be used to build it. It is really important for an airplane to be as lightweight as possible. The less weight an airplane has, the less work the engines have to do and the farther it can fly. It is tough designing an airplane that is lightweight and strong at the same time. In the past, airplanes were usually made out of lightweight metals like aluminum, but today a lot of engineers are thinking about using composites in their designs. Composites look and feel like plastic but are stronger than most metals. Engineers also need to make sure that airplanes not only fly well but are also easy to build and maintain.

Stability and Control is the study of how an airplane handles and interacts to pilot input and feed. Pilots in the cockpit have a lot of data to read from the airplane's computers or displays. Some of this information could include the airplane's speed, altitude, direction, and fuel levels as well as upcoming weather conditions and other instructions from ground control. The pilot needs to be able to process the correct data quickly, to think about what kind of action needs to be taken, and to react in an appropriate way. Meanwhile, the airplane should display information to the pilot in an easy-to-read and easy-to-understand way. The controls in the cockpit should be within easy reach and just where the pilot expects them to be. It is also important that the airplane responds quickly and accurately to the pilot's instructions and maneuvers.

When you look at aircraft, it is easy to observe that they have a number of common features: wings, a tail with vertical and horizontal wing sections, engines to propel them through the air, and a fuselage to carry passengers or cargo. If, however, you take a more critical look beyond the gross features, you also can see subtle, and sometimes not so subtle, differences. This is where design comes into play. Each and every aircraft is built for a specific task, and the design is worked around the requirement and need of the aircraft. The design is modelled about the aircraft role and type and not the other way around. Thus, this is why airplanes differ from each other and are conceptualized differently. Aircrafts that fall in the same category may have similar specifications and performance parameters, albeit with a few design changes.

Design is a pivotal part of any operation. Without a fixed idea or knowledge of required aircraft, it is not possible to conceive the end product. Airplane design is both an art and a science. In that respect it is difficult to learn by reading a book; rather, it must be experienced and practiced. However, we can offer the following definition and then attempt to explain it. Airplane design is the intellectual engineering process of creating on paper (or on a computer screen) a flying machine to (1) meet certain specifications and requirements established by potential users (or as perceived by the manufacturer) and/or (2) pioneer innovative, new ideas and technology. An example of the former is the design of most commercial transports, starting at least with the

Douglas DC-1 in 1932, which was designed to meet or exceed various specifications by an airplane company. (The airline was TWA, named Transcontinental and Western Air at that time.) An example of the latter is the design of the rocket-powered Bell X1, the first airplane to exceed the speed of sound in level or climbing flight (October 14, 1947). The design process is indeed an intellectual activity, but a rather special one that is tempered by good intuition developed via experience, by attention paid to successful airplane designs that have been used in the past, and by (generally proprietary) design procedures and databases (handbooks, etc) that are a part of every airplane manufacturer.

1.1 DEFINING A NEW DESIGN

The design of an aircraft draws on a number of basic areas of aerospace engineering. These include aerodynamics, propulsion, light-weight structures and control. Each of these areas involves parameters that govern the size, shape, weight and performance of an aircraft. Although we generally try to seek optimum in all these aspects, with an aircraft, this is practically impossible to achieve. The reason is that in many cases, optimizing one characteristic degrades another.

There are many performance aspects that can be specified by the mission requirements. These include:

- The aircraft purpose or mission profile
- The type(s) and amount of payload
- The cruise and maximum speeds
- The normal cruise altitude
- The range or radius with normal payload
- The endurance
- The take-off distance at the maximum weight
- The purchase cost

1.1.1 Aircraft Purpose

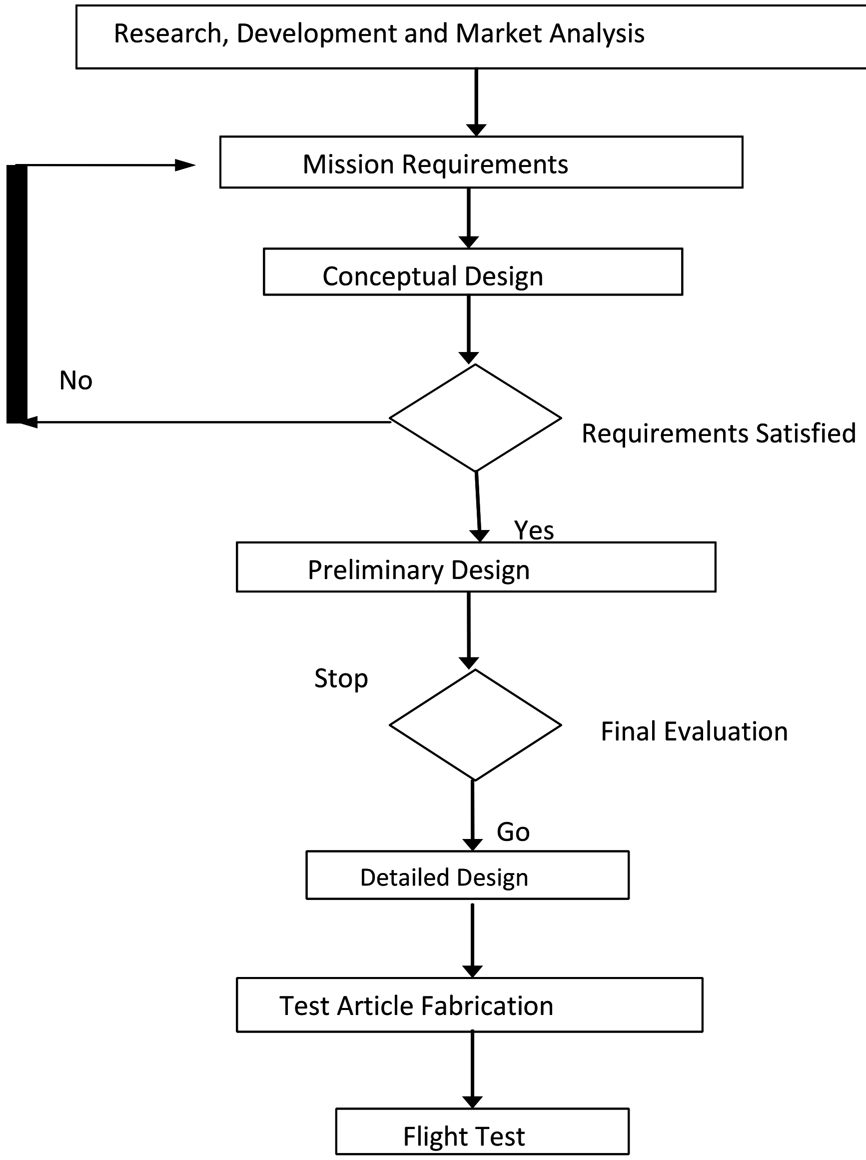
The starting point of any new aircraft is to clearly identify its purpose. With this, it is often possible to place a design into a general category. Such categories include combat aircraft, passenger or cargo transports, and general aviation aircraft. These may also be further refined into subcategories based on particular design objectives such as range (short or long), take-off or landing distances, maximum speed, etc. The process of categorizing is useful in identifying any existing aircraft that might be used in making comparisons to a proposed design. With modern military aircraft, the purpose for a new aircraft generally comes from a military program office. For example, the mission specifications for the X-29 pictured in figure 1.1 came from a 1977 request for proposals from the U.S. Air Force Flight Dynamics Laboratory in which they were seeking a research aircraft that would explore the forward swept wing concept and validate studies that indicated such a design could provide better control and lift qualities in extreme maneuvers. With modern commercial aircraft, a proposal for a new design usually comes as the response to internal studies that aim to project future market needs. For example, the specifications for the Boeing commercial aircraft (B-777) were based on the interest of commercial airlines to have a twin-engine aircraft with a payload and range in between those of the existing B-767 and B-747 aircraft. Since it is not usually possible to optimize all of the performance aspects in an aircraft, defining the purpose leads the way in setting which of these aspects will be the "design drivers." For example, with the B-777, two of the prominent design drivers were range and payload.

1.2 DESIGN MOTIVATION

Fundamentally, an aircraft is a structure. Aircraft designers design structures. The structures are shaped to give them desired aerodynamic characteristics, and the materials and structures of their engines are chosen and shaped so they can provide needed thrust. Even seats, control sticks, and windows are structures, all of which must be designed for optimum performance. Designing aircraft structures is particularly challenging, because their weight must be kept to a minimum. There is always a trade-off between structural strength and weight. A good aircraft structure is one which provides all the strength and rigidity to allow the aircraft to meet all its design requirements, but which weighs no more than necessary. Any excess structural weight often makes the aircraft cost more to build and almost always makes it cost more to operate. As with small excesses of aircraft drag, a small percentage of total aircraft weight used for structure instead of payload can make the difference between a profitable airliner or successful tactical fighter and a failure. Designing aircraft structures involves determining the loads on the structure, planning the general shape and layout, choosing materials, and then shaping, sizing and optimizing its many components to give every part just enough strength without excess weight. Since aircraft structures have relatively low densities, much of their interiors are typically empty space which in the complete aircraft is filled with equipment, payload, and fuel. Careful layout of the aircraft structure ensures structural components are placed within the interior of the structure so they carry the required loads efficiently and do not interfere with placement of other components and payload within the space. Choice of materials for the structure can profoundly influence weight, cost, and manufacturing difficulty. The extreme complexity of modern aircraft structures makes optimal sizing of individual components particularly challenging. An understanding of basic structural concepts and techniques for designing efficient structures is essential to every aircraft design.

1.3 DESIGN PROCESS

The process of designing an aircraft and taking it to the point of a flight test article consists of a sequence of steps, as illustrated in the figure. It starts by identifying a need or capability for a new aircraft that is brought about by (1) a perceived market potential and (2) technological advances made through research and development. The former will include a market-share forecast, which attempts to examine factors that might impact future sales of a new design. These factors include the need for a new design of a specific size and performance, the number of competing designs, and the commonality of features with existing aircraft. As a rule, a new design with competitive performance and cost will have an equal share of new sales with existing competitors. The needs and capabilities of a new aircraft that are determined in a market survey go to define the mission requirements for a conceptual aircraft. These are compiled in the form of a design proposal that includes (1) the motivation for initiating a new design and (2) the "technology readiness" of new technology for incorporation into a new design. It is essential that the mission requirements be defined before the design can be started. Based on these, the most important performance aspects or "design drivers" can be identified and optimized above all others. Following the design proposal, the next step is to produce a conceptual design. The conceptual design develops the first general size and configuration for a new aircraft. It involves the estimates of the weights and the choice of aerodynamic characteristics that will be best suited to the mission requirements stated in the design proposal. The conceptual design is driven by the mission requirements, which are set in the design proposal. In some cases, these may not be attainable so that the requirement may need to be relaxed in one or more areas. This is shown in the iterative loop in the flow chart. When the mission requirements are satisfied, the design moves to the next phase, which is the preliminary design.



Conceptual design

This article deals with the steps involved in the conceptual design of an aircraft. It is broken down in to several elements, which are followed in order. These consist of:

1. Literature survey
2. Preliminary data acquisition
3. Estimation of aircraft weight
    a. Maximum take-off weight
    b. Empty weight of the aircraft
    c. Weight of the fuel
    d. Fuel tank capacity
4. Estimation of critical performance parameters
    a. Wing area
    b. Lift and drag coefficients
    c. Wing loading
    d. Power loading
    e. Thrust to weight ratio
5. Engine selection
6. Performance curves
7. 3 View diagrams

1.4 DESIGN BREAKDOWN

Table 1.1 (Design Process Breakdown)

• **Conceptual Design:** - Competing concepts evaluated - Performance goals established - Preferred concept selected	What drives the design? Will it work/meet requirement? What does it look like?
• **Preliminary Design:** - Refined sizing of preferred concept tests - Design examined data/establish parameters - Some changes allowed	Do serious wind tunnel tests Make actual cost estimate

• **Detail Design:** - Final detail design - Drawings released - Detailed performance - Only “tweaking” of design allowed	Certification process Component/systems tests Manufacturing Flight control system design

CHAPTER 02

INTRODUCTION TO ROCKETS

Rocket is a type of jet-propulsion device carrying either solid or liquid propellants that provide both the fuel and oxidizer required for combustion. The term is commonly applied to any of various vehicles, including firework skyrockets, guided missiles, and launch vehicles used in spaceflight, driven by any propulsive device that is independent of the atmosphere.

The rocket differs from the turbojet and other "air-breathing" engines in that all of the exhaust jet consists of the gaseous combustion products of "propellants" carried on board. Like the turbojet engine, the rocket develops thrust by the rearward ejection of mass at very high velocity.

The fundamental physical principle involved in rocket propulsion was formulated by Sir Isaac Newton. According to his third law of motion, the rocket experiences an increase in momentum proportional to the momentum carried away in the exhaust,

$$M\Delta v_{\mathrm{R}} = \dot{m}v_e\Delta t = F\Delta t, \qquad (1)$$

where M is the rocket mass, Δv_{R} is the increase in velocity of the rocket in a short time interval, Δt, $m°$ is the rate of mass discharge in the exhaust, v_e is the effective exhaust velocity (nearly equal to the jet velocity and taken relative to the rocket), and F is force. The quantity $m°v_e$ is the propulsive force, or thrust, produced on the rocket by exhausting the propellant,

$$F = \mathring{m}v_e. \qquad (2)$$

Most rockets derive their energy in thermal form by combustion of condensed-phase propellants at elevated pressure. The gaseous combustion products are exhausted through the nozzle that converts most of the thermal energy to kinetic energy. The maximum amount of energy available is limited to that provided by combustion or by practical considerations imposed by the high temperature involved. Higher energies are possible if other energy sources (e.g., electric or microwave heating) are used in conjunction with the chemical propellants on board the rockets, and extremely high energies are achievable when the exhaust is accelerated by electromagnetic means.

The effective exhaust velocity is the figure of merit for rocket propulsion because it is a measure of thrust per unit mass of propellant consumed—

$$\frac{F}{\overset{\circ}{m}} = v_e. \qquad (3)$$

A technique called multiple staging is used in many missions to minimize the size of the takeoff vehicle. A launch vehicle carries a second rocket as its payload, to be fired after burnout of the first stage (which is left behind). In this way, the inert components of the first stage are not carried to final velocity, with the second-stage thrust being more effectively applied to the payload. Most spaceflights use at least two stages. The strategy is extended to more stages in missions calling for very high velocities.

The unique features of rockets that make them useful include the following:

1. Rockets can operate in space as well as in the atmosphere of Earth.
2. They can be built to deliver very high thrust (a modern heavy space booster has a takeoff thrust of 3,800 kilonewtons (850,000 pounds).
3. The propulsion system can be relatively simple.
4. The propulsion system can be kept in a ready-to-fire state (important in military systems).
5. Small rockets can be fired from a variety of launch platforms, ranging from packing crates to shoulder launchers to aircraft (there is no recoil).

CHAPTER 03
STUDY OF DIFFERENT ROCKETS

3.1 Lockheed Martin X-33

With the cost of space flight still hovering at around $10,000 per pound, NASA spearheaded the development of several potentially revolutionary technology demonstration projects. The purpose of these efforts was to reduce the cost of lifting a pound into orbit to as little as $1,000. The most significant project under the umbrella of the Space Launch Initiative (SLI) was the Lockheed Martin X-33 announced in 1996. The ultimate goal of the X-33 was to develop a completely reusable single-stage-to-orbit (SSTO) launch vehicle to replace the aging Space Shuttle by about 2010.

The Lockheed Martin X-33 design is based on a lifting body shape with two revolutionary linear aerospike rocket engines and a rugged metallic thermal protection system was judged most promising among Rockwell & McDonnell Douglas. It will be an autonomously piloted SSTO, launched vertically like a rocket, reaching an altitude of 60 miles, speeds faster than Mach 13 and landing horizontally like an airplane and lightweight composite fuel tanks, an integrated thermal protection system to make the heat-absorbing tiles used on the Space Shuttle unnecessary. Although suborbital, the X-33 will fly high enough and fast enough to encounter conditions similar to those experienced on an orbital flight path to fully test its systems and performance. Highlighting the vehicle's aircraft-like operations, the X-33 program will demonstrate a standard seven-day turnaround, as well as an emergency two-day turnaround, between selected flights. If the X-33 technology demonstrator proved successful, NASA and Lockheed Martin hoped to develop a full-sized vehicle twice the size of the X-33 called the VentureStar that would supersede the Space Shuttle as America's primary launch platform. The ambitious project set in motion a rapid test program with the first flight set for early 1999. Unfortunately, these goals proved too ambitious since the X-33 was beset by a number of difficult and time-consuming technical problems. Early wind tunnel and flight tests of X-33 models proved the wedge shape to be longitudinally unstable requiring changes to the control surfaces. In addition, the aerospike engine suffered development delays, as did the thermal protection system. Most critical, however, was the failure of the composite fuel tanks that eventually forced Lockheed Martin to abandon them altogether. Aluminum tanks were instead substituted, but the change forced modifications to the vehicle structure and increased overall weight.

Individually, each of these difficulties was probably not fatal, but together they convinced NASA that the X-33 relied on too many untried and unproven technologies that could not be developed for as low a cost and with as great a reliability as the X-33 performance requirements demanded. As a result, NASA canceled further spending on the X-33 in March 2001 after the vehicle was about 75% complete. Lockheed Martin had already invested over $200 million of company funds in the project, and NASA's cancellation left the firm free to complete and fly the X-33 alone. Lockheed elected not to do so, however, and the VentureStar program appears to have been permanently shelved. Nevertheless, Congressional supporters of the X-33 argued for a renewal of the project as a US Air Force program, but this idea too has never progressed any further.

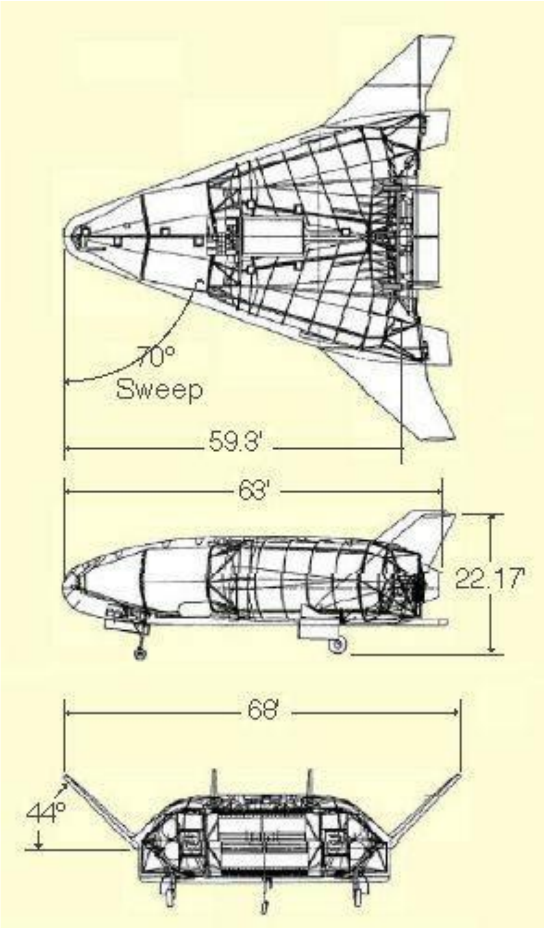


Figure.3.1: Lockheed Martin X-33

3.2 Skylon

The Skylon is being designed to take off from a robust concrete runway, climb into low Earth orbit (LEO) and fly back to the earth in a single stage. It will consume liquid hydrogen during take-off and liquid oxygen during ascent into orbit. It is being designed for 200 flights and will have capacity to store 66,000kg of liquid hydrogen and 150,000kg of liquid oxygen.

The vehicle will operate in two modes: air breathing mode and rocket mode. During air breathing mode, it will attain a speed of Mach 5 at an altitude of 26,000m by collecting atmospheric air through two shock inlets. The atmospheric air entering the inlet will be heated up as it is compressed in the engine. It will be chilled to cryogenic temperatures before compression by a precool heat exchanger using liquid hydrogen and helium. Before switching to rocket mode, the compressed air will be converted to vapor but not liquefied, to reduce fuel consumption for cooling. The decrease of air density being inversely proportional to the altitude will allow the engine to switch to rocket mode. Pitch, yaw, and roll control in rocket mode will be rendered by the combustion chambers. Liquid oxygen will be used to fuel the engines in the rocket mode and aid the spacecraft's ascent to the orbit.

Upon arriving at the LEO, the vehicle ignites the onboard maneuvering system (OMS) to disseminate the orbit. The payloads will be deployed in orbit after completing routine checkouts. Once the payload is released, the lightweight aircraft re-enters the atmosphere and lands back on the runway. Heat radiation at altitude ranging between 90 km and 60 km during the re-entry stage will be intercepted from entering the vehicle by covering the fuselage with ceramic composite skin.

The Skylon will feature a thin fuselage comprising canard fore planes, hydrogen tanks, oxygen tanks and auxiliary propellant tankage. It will be fitted with delta shaped wings alongside the midway fuselage and incorporate SABRE (synergistic air breathing rocket engine) on the axisymmetric nacelles of wingtips.

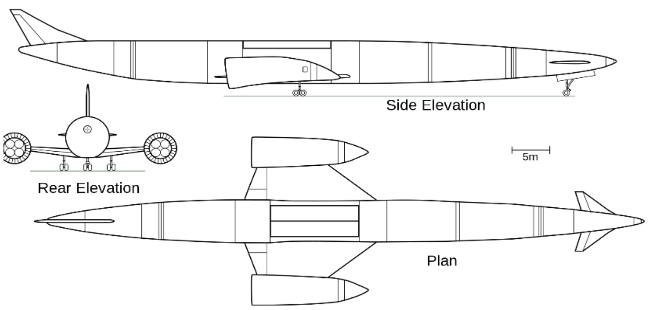


Figure.3.2: Skylon

The vehicle will boast a payload bay with U-shaped cross section in the middle of the fuselage section. The bay will comprise two interfaces, one at the front and another at rear. The front interface will be used to load cryogenic oxygen, hydrogen, and helium. Both interfaces will be utilized to load payloads. The length and diameter of the bay will be 12.3m and 4.6m respectively. The spacecraft can conciliate 40 ft-long cargo containers in the payload bay for space transport missions. A passenger cabin module is also planned to be placed in the bay to carry 30 to 40 passengers upon receiving endurance certification.

The SABRE engine system is the most innovative and important part of Skylon. It is this propulsion system which enables the spacecraft to be single stage to orbit. The system is based around a rocket engine that uses liquid hydrogen for fuel and ignites with either liquid oxygen or condensed air. The choice between the two oxidizers is determined by the altitude of the spacecraft's flight. When Skylon is below a predetermined threshold, or 28.5 km in altitude and Mach 5 speed, the air-breathing capabilities are used to avoid using on-board liquid oxygen. An innovative helium loop system chills and compresses the ingested air to an almost liquid state to be used by the rocket engine.

After SKYLON reaches an altitude of 28.5 km and speed of Mach 5, the compressor is unable to supply the rocket engine, and the liquid oxygen is supplied from the launched reserves to complete the ascent. Combining both of these capabilities into one system reduces the mass of launching a separate air-breathing system and a rocket in the same flight, therefore eliminating the need to have multiple staging. air is guided through an internal bypass system. The air mixed with hydrogen in a bypass burner, increasing the overall thrust. The bypass system is throttle-able to allow for the most efficient burn. As SKYLON reaches altitude, the internal center body moves forward, and three conical frustums are stacked in front of the intake. This closes the intake nacelle and creates a very aerodynamic surface.

When the SKYLON vehicle reaches Mach 5 and an altitude of 28.5 km, it has reached a point where the condensed atmosphere doesn't meet the needs of the core rocket engine. As previously stated, the intake of the nacelle closes up and the liquid oxygen tanks supply the engine. This mode is used through the rest of the mission until orbit and descent. During orbit the SOMA (SKYLON Orbital Maneuvering Assembly) and reaction engines are used for correctional maneuvers, and in descent, the vehicle is converted to a glide-to-land configuration.

SABRE engine:

1. Intake

The stagnant intake has been sized to consume enough air in the end of its air breathing capabilities (Mach 5 and 28.5 km altitude). However, this means that during lower velocity and altitude flight, the intake allows for too much air to come through. Bypass chambers direct extra flow around the core inside the nacelle to a special burner chamber. These chambers are able to expand to capture more overflow or close towards the end of air breathing flight. The intake also contains cowls to reduce the speed of the incoming air so that it can be more efficiently cooled.

2. Precooler

Since the stagnation temperature of the Mach 5 intake air can be in excess of 950C, a pre-cooler is required to reduce the overall power necessary to run the compressor. Cooling the incoming

air decreases the energy state of the gas and allows it to be compressed more easily. The pre-cooler uses a helium loop to remove the heat from the air, and then transfers that heat to the hydrogen fuel loop.

3. Turbo Compressor

This high pressure-ratio compressor (150:1) is what feeds the rocket combustion chamber. It allows for in situ resources use of air in the rocket instead of carrying more liquid oxygen up during launch. The compressor increases the pressure of the gas to just below liquid level. Doing this does not compromise the burn characteristics of the rocket and reduces the amount of energy needed to be put into the compressor system. It also helps to protect the material from degrading faster due to the higher pressures. The heated hydrogen from the pre-cooler is fed into a turbo pump to drive the air turbo compressor, resulting in a higher efficiency system.

4. Pre-Burner

The pre-burner in the SABRE system has a slightly different function than that in a typical liquid propellant engine. Instead of the output powering the turbo pumps or compressors in the upstream, it instead gives additional heat to the helium loop feeding the turbo machinery. After some of the heat is removed from the pre-burner output, it is fed into the combustion chamber for further burning since the mixture is always fuel rich.

5. Helium Circulator

The innovative cooling for the incoming air is provided by a helium loop, instead of the commonly used hydrogen. Helium acts as an intermediate thermal buffer between the extremely hot air and the cryogenic liquid nitrogen, reducing material embrittlement failure. After the helium cools the intake air to 250C, it travels past the output of the pre-burner to absorb more heat, and then drives the liquid oxygen turbo pump and drives the turbo compressor for the intake air. Any of the remaining energy left in the helium as energy is removed by running the helium past a cooler loop of hydrogen. After the nacelles are closed and SABRE is converted to a conventional rocket engine, the leg powering the turbo compressor is turned off and only the liquid oxygen turbo pump is powered.

6. Bell Nozzle & Thrust Chambers

SABRE currently employs conventional bell nozzles in their design but has left the possibility open for an extendable nozzle for greater efficiency in higher altitude flight. The chamber also encounters a unique problem with cooling, since the hydrogen loop is already used for cooling the helium loop. Instead, film cooling with compressed air is used during the air-breathing portion of flight. When in rocket mode, this is switched to a liquid oxygen loop fed through a coil around the nozzle.

3.3 Roton

The American Roton company developed this unique manned SSTO VTOL orbital launch vehicle until it was canceled in 2000. The Roton was a piloted commercial space vehicle design intended to provide rapid and routine access to orbit for both its two-person crew and their cargo. The Roton was a fully reusable, space vehicle designed to transport up to 3200 kg to and from a 300 km / 50-degree inclination earth orbit. The Roton was planned in 1998 to reach commercial service in the year 2000 with a target price per flight of $7 million. The cargo envelope for the vehicle was 3.66 m in diameter and 5.08 m in height. The original Roton design was to use an innovative novel rotary engine harking back to the Aerojet Rotojet design of World War II.

Later this was changed to a version of NASA's Fastrac Engine using Kerosene and Liquid Oxygen propellants. Several would be required to provide the necessary thrust at take-off.

Roton would have a height of 19.5 m, a diameter of 6.7 m and a maximum gross lift-off weight of under 180 metric tons. Using the Fastrac engines, Roton would have to have an empty weight under 5 % of its gross take-off mass - a very challenging figure for an expendable rocket, let alone the additional mass of the crew, rotor system, and thermal protection system.

The Roton would take-off vertically like a conventional rocket powered by a novel rotary engine burning liquid oxygen and jet fuel. The launch system infrastructure for the Roton was expected to be minimal with only a towing truck, standard refueling equipment and support lines, a blast pad, and a small launch platform. The Roton would lift-off under command of the crew and would fly to orbit from the Mojave, California launch site, using standard GPS and inertial navigation flight control systems. The launch path of the vehicle would be very steep, traveling only 28 km downrange at an altitude of 30 km. Following the main engine cut-off at about 130 km, the Roton would circularize its orbit using its orbital maneuvering engines. In orbit the vehicle had a maximum stay time of 72 hours.

Once its payload was deployed, the Roton returned to Earth via a nose-mounted rotor. The rotor recovery system was lightweight and provided a slow, pilot-controlled approach to the landing site. The rotor system consisted of a set of four rotor blades, a rotor hub and nose cap. During launch, the blades were folded down along the sides of the vehicle and before re-entry they were deployed and angled upward away from the nose. During the hypersonic and supersonic phases of flight, the base of the vehicle produced most of the drag while the rotor remained windmilling behind the vehicle, stabilizing it until it reached subsonic speed. Use of water in an active-cooling system was considered.

The rotor would then be spun up and the blades entered a helicopter-style autorotation flight mode. While in this mode the Roton was able to glide with the pilot's direction to a precision landing point. The vehicle's rotors would be equipped with tip rockets so that the Roton could touchdown softly under rotor power. After landing the vehicle would undergo minor maintenance and checkout procedures to prepare it for its next flight. To meet the target operational cost goals of this project, the Roton was being designed to operate much like a commercial aircraft.

Take a helicopter. Pull the engine out. Run fuel lines down inside the length of the blades. Put rockets on the rotor tips. Use the centrifugal force on the fuel in the spinning blades to pressurize the combustion chamber to very high pressure. Use the rockets to turn the rotor.

Throttle the rockets way down to conserve fuel and use the aerodynamic lift of the rotor to go up. As the air gets thinner, increase the collective pitch on the blades for more lift. This also increases the downward component of thrust from the rockets. At the same time, gradually increase the propellant flow to the rocket motors. By the time there is virtually no atmosphere, the rockets will be pointing almost straight down (although not completely, since the blades have to keep spinning in order to keep the pressure up). The downward component of the rockets will continue to propel the craft.

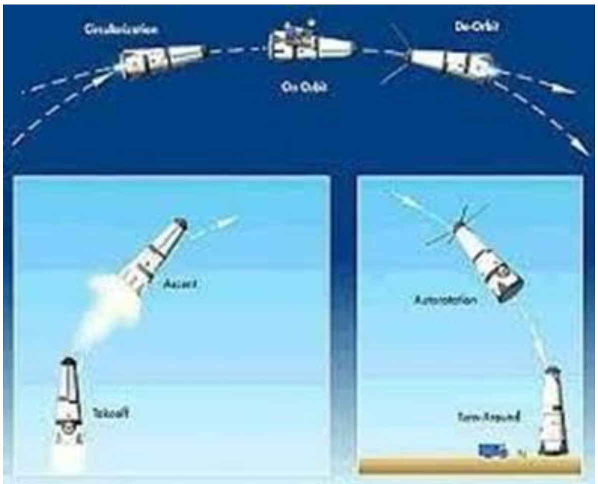


Figure.3.3: Roton

From the outset, the Roton was also designed to return from space with a fully loaded payload bay. Therefore, if a spacecraft was damaged or malfunctioned prior to deployment, it could be returned for repair. Rotary Rocket expected a market to develop whereby satellite operators, manufacturers, and insurance companies send rescue missions to repair or retrieve damaged or outdated satellites. In addition to this service, the Roton's low cost of operation and payload- return capability would also allow for the development of an orbital materials processing industry. The vacuum and zero-g environment of space provided ideal conditions to manufacture a variety of materials as well as conduct research.

The Roton large-scale Air Test Vehicle began flight tests in 1999 to demonstrate the autorotation and rotor capabilities. Thereafter technical challenges and lack of sufficient investment stalled the project, then finally killed it.

3.4 Rombus

Rombus (Reusable Orbital Module-Booster & Utility Shuttle) is an American SSTO VTOVL orbital launch vehicle. Bono original design for ballistic single-stage-to-orbit (not quite - it dropped liquid hydrogen tanks on the way up) heavy lift launch vehicle. The recoverable vehicle would re-enter, using its actively cooled plug nozzle as a heat shield.

In 1964, Phil Bono of Douglas Aircraft Co. proposed a low-cost heavy lift VTVL SSTO RLV plus lunar base as a logical follow-on to the Apollo project. Bono's "Reusable Orbital Module-Booster & Utility Shuttle " (ROMBUS) concept was based on his patented plug nozzle rocket engine design, which doubles as a heat shield during atmospheric re-entry. The vehicle's base-first re-entry mode assures a stable condition during recovery since the mass of the engine is very high, i.e., far aft center of gravity. The plug nozzle would be cooled by circulating liquid hydrogen through the same regenerative system used for cooling the engines and base of the vehicle while the engines are operating during ascent. Mixture ratio of liquid oxygen to hydrogen was raised to 7:1 -- about as close as one can get to the stoichiometric value of 8:1 without running into combustion chamber cooling problems. During ascent, the plug nozzle provides automatic altitude compensation and therefore good performance at both sea level atmospheric pressure and in space. For final orbital insertion, 16 of the 36 engines would burn for 3 seconds to provide the required velocity. ROMBUS would typically spend 24 hours in orbit before the ground track passes close enough to the launch site for deorbit. Parachutes and (beginning at 0.73km altitude-) retrorockets would be used to safely land the vehicle. The final touchdown burn would be provided by four engines running at 25% thrust for approximately twelve seconds. To reduce the size and weight of the vehicle, the hydrogen fuel was to be stored in eight external jettisonable tanks. The tanks were jettisoned and then recovered by parachute as they were depleted during ascent to orbit.

The total life cycle cost would have been $10 billion 1964 $'s over 10 years including $4.088 billion for the development program. Bono mentioned the following SSTO RLV-specific advantages: increased reliability since each vehicle has a history, reduced development cost & complexity vs. multi-stage vehicles, economies of scale possible since plug nozzle engine units & tanks could be mass produced. The estimated direct launch cost was $22.4 million (=$28/lb. to a 568 km orbit at 1964 economic conditions) and the planned vehicle turnaround time was about 76 days. Bono also mentioned a direct operations cost goal of $12/lb (5-6 reuses) - $5/lb (>20 reuses) for a vehicle payload capability of 450t by the year 1975. In comparison, the Saturn V was then expected to cost $150-250/lb. The vehicle would have used the same Kennedy Space Centre facilities as the Saturn boosters, although a new launch pad would have been required.

3.5 Atlas B (PGM-16B or SM-65B or X-12)

Originating as the X-12, Atlas B was the name given to the second series of Atlas missiles delivered to Cape Canaveral for flight testing. Since the Atlas B missile was built to test booster and nose cone separation as well as the overall propulsion system, longer-range flights were needed. This required the Atlas B to employ an operating one and one-half stage booster/sustainer engine combination. In order for a B Series Test to be considered completely successful, five events called "Marks" needed to be completed. Mark One was booster engine shutdown, Mark Two was booster engine separation, Mark Three was sustainer engine cut-off, Mark Four was vernier engine cut-off and Mark Five was nose cone separation. To do this, the Atlas B needed to fly nearly ten times farther than the Atlas A.

The Atlas B employed two North American booster engines and one North American sustainer engine. The sustainer engine was located in between the two booster engines. The engines each had a thrust of 120,000 pounds at lift-off. Like the Atlas A, the missile used two vernier engines to control roll and trim final velocity. All engines were fed by liquid oxygen/RP-1 (kerosene) liquid propellant. Like the Atlas A, the Atlas B carried a semi-inertial guidance system supported by radio commands from ground stations. All engines ignited at lift-off. At an altitude of about 16,000 feet, the Atlas B performed a pre-programmed turn to a ballistic trajectory. About two minutes into the flight, the two booster engines shut down and were jettisoned. The sustainer engine remained firing for about an additional two minutes, staying attached to the missile body to its final water impact.

Following the sustainer engine cut-off, the two vernier engines remained firing for about 30 seconds. Following the cut-off of the vernier engines, the nose cone separated, continuing on toward its target without further guidance. An Atlas B was first launched from Cape Canaveral on July 19, 1958. The missile lost thrust about 43 seconds into the flight, exploded and fell into the Atlantic Ocean about three miles downrange.

In stark contrast, an Atlas B launched from Cape Canaveral successfully completed the first full-range Atlas test flight on November 28, 1958. The missile ended its flight about 6,000 miles downrange of the launch site.

3.6 Saturn V

The Saturn V was a rocket NASA built to send people to the moon. (The V in the name is the Roman numeral five.) The Saturn V was a type of rocket called a Heavy Lift Vehicle. That means it was very powerful. It was the most powerful rocket that had ever flown successfully. The Saturn V was used in the Apollo program in the 1960s and 1970s. It also was used to launch the Skylab space station.

The Saturn V was developed at NASA's Marshall Space Flight Center in Huntsville, Ala. It was one of three types of Saturn rockets NASA built. Two smaller rockets, the Saturn I (1) and IB (1b), were used to launch humans into Earth orbit. The Saturn V sent them beyond Earth orbit to the moon. The first Saturn V was launched in 1967. It was called Apollo 4. Apollo 6 followed in 1968. Both of these rockets were launched without crews. These launches tested the Saturn V rocket.

The first Saturn V launched with a crew was Apollo 8. On this mission, astronauts orbited the moon but did not land. On Apollo 9, the crew tested the Apollo moon lander by flying it in Earth orbit without landing. On Apollo 10, the Saturn V launched the lunar lander to the moon. The crew tested the lander in space but did not land it on the moon. In 1969, Apollo 11 was the first mission to land astronauts on the moon. Saturn V rockets also made it possible for astronauts to land on the moon on Apollo 12, 14, 15, 16 and 17. On Apollo 13, the Saturn V lifted the crew into space, but a problem prevented them from being able to land on the moon. That problem was not with the Saturn V, but with the Apollo spacecraft. The last Saturn V was launched in 1973, without a crew. It was used to launch the Skylab space station into Earth orbit.

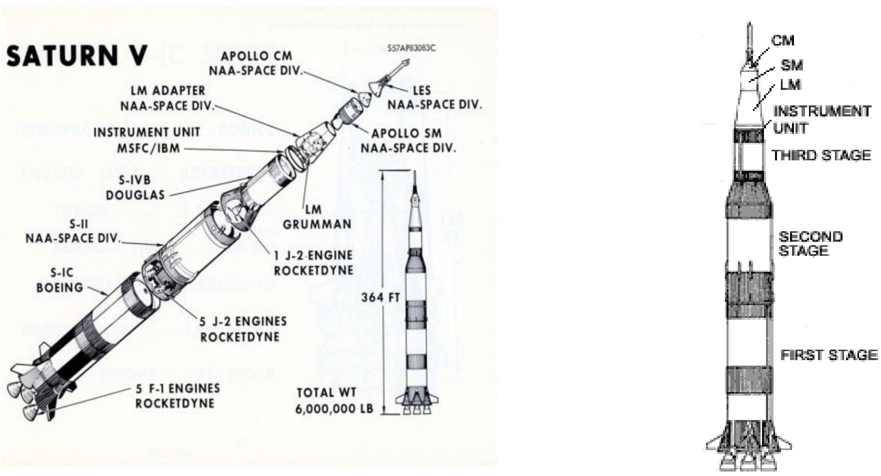


Figure.3.4: Saturn V

The Saturn V that launched the Skylab space station only had two stages. The Saturn V rockets used for the Apollo missions had three stages. Each stage would burn its engines until it was out of fuel and would then separate from the rocket. The engines on the next stage would fire, and the rocket would continue into space. The first stage had the most powerful engines, since it had the challenging task of lifting the fully fueled rocket off the ground.

The first stage lifted the rocket to an altitude of about 68 kilometers (42 miles). The second stage carried it from there almost into orbit. The third stage placed the Apollo spacecraft into Earth orbit and pushed it toward the moon. The first two stages fell into the ocean after separation. The third stage either stayed in space or hit the moon.

3.7 Falcon 9

Falcon 9 is a reusable, two-stage rocket designed and manufactured by SpaceX for the reliable and safe transport of people and payloads into Earth orbit and beyond. Falcon 9 is the world's first orbital class reusable rocket. Reusability allows SpaceX to re-fly the most expensive parts of the rocket, which in turn drives down the cost of space access.

Merlin is a family of rocket engines developed by SpaceX for use on its Falcon 1, Falcon 9 and Falcon Heavy launch vehicles. Merlin engines use a rocket grade kerosene (RP-1) and liquid oxygen as rocket propellants in a gas-generator power cycle. The Merlin engine was originally designed for recovery and reuse.

Falcon 1 could place a 1,010-kg (2,227-pound) payload into orbit at lower cost than other launch vehicles. Falcon 9 was designed to compete with the Delta family of launchers in that it can lift payloads of up to 8,300 kg (18,300 pounds) to geostationary orbit. One payload it launched to low Earth orbit is Dragon, a spacecraft designed to carry crew and cargo to the International Space Station (ISS). Falcon Heavy has the first stages of three Falcon 9 launch vehicles joined together as its first stage and is designed to carry 53,000 kg (117,000 pounds) to orbit, nearly twice that of its largest competitor, the Boeing Company's Delta IV Heavy.

The first test flight of Falcon 9 was on June 4, 2010, from Cape Canaveral, Florida, and the first resupply mission to the ISS was made on October 7, 2012. In 2014 tests began on a reusable first stage for the Falcon 9 that would land on a floating platform. On December 21, 2015, a Falcon 9 launched a payload into orbit, and its first stage made a landing at Cape Canaveral. The first Falcon 9 first-stage ship landing happened on April 8, 2016, and SpaceX did its first relaunch of a previously flown Falcon 9 first stage on March 30, 2017. The first

Falcon Heavy test flight occurred on February 6, 2018. The central core stage was not recovered, but the two side boosters successfully returned to Cape Canaveral. The payload, a Tesla Roadster, with a SpaceX spacesuit buckled into the driver's seat, was placed into orbit around the Sun. A Falcon 9 launched the first private crewed spacecraft, a Dragon carrying astronauts Doug Hurley and Robert Behnken, to the ISS on May 30, 2020.

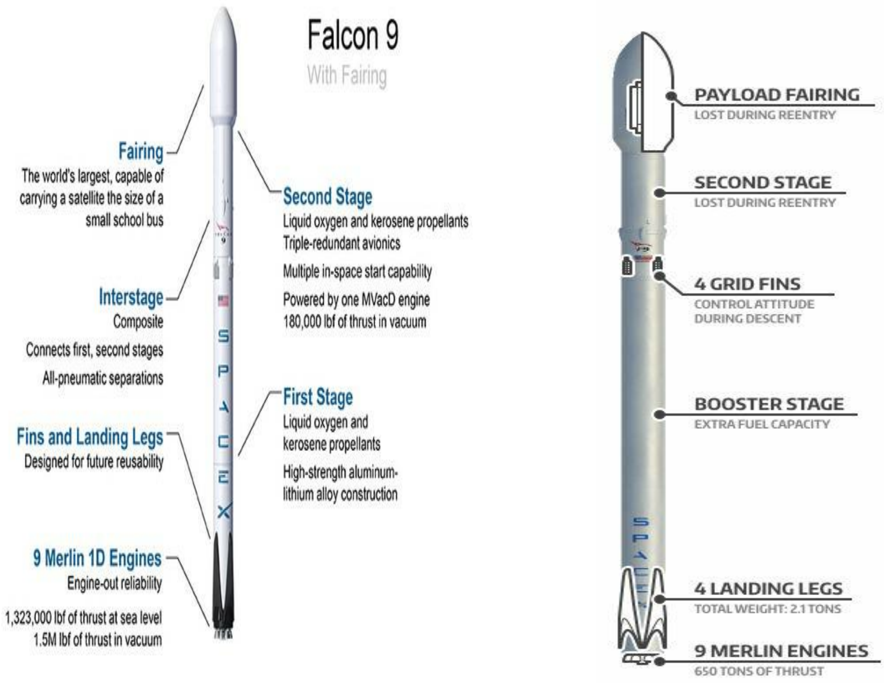


Figure.3.5: Falcon 9

3.8 PSLV

Polar Satellite Launch Vehicle (PSLV) is the third generation launch vehicle of India. It is the first Indian launch vehicle to be equipped with liquid stages. After its first successful launch in October 1994, PSLV emerged as the reliable and versatile workhorse launch vehicle of India with 39 consecutively successful missions by June 2017. During the 1994-2017 period, the vehicle launched 48 Indian satellites and 209 satellites for customers from abroad. Besides, the vehicle successfully launched two spacecraft – Chandrayaan-1 in 2008 and Mars Orbiter Spacecraft in 2013 – that later traveled to Moon and Mars respectively. While the PSLV-G uses 6 HTPB based solid strap-on motors of 9 tonnes each and PSLV-XL uses 6 extended strap-ons of 12 tonnes each, the PSLV-CA (core alone version) does not use any strap-on motors. The PSLV is capable of placing multiple payloads into orbit, thus multi-payload adaptors are used in the payload fairing. This allowed the feat of launching 10 satellites into different orbits in 2008. More recently, on June 30, 2014, PSLV-C23 launched SPOT-7, CAN-X4, CAN-X5, AISAT and VELOX-1 into their designated orbits successfully.

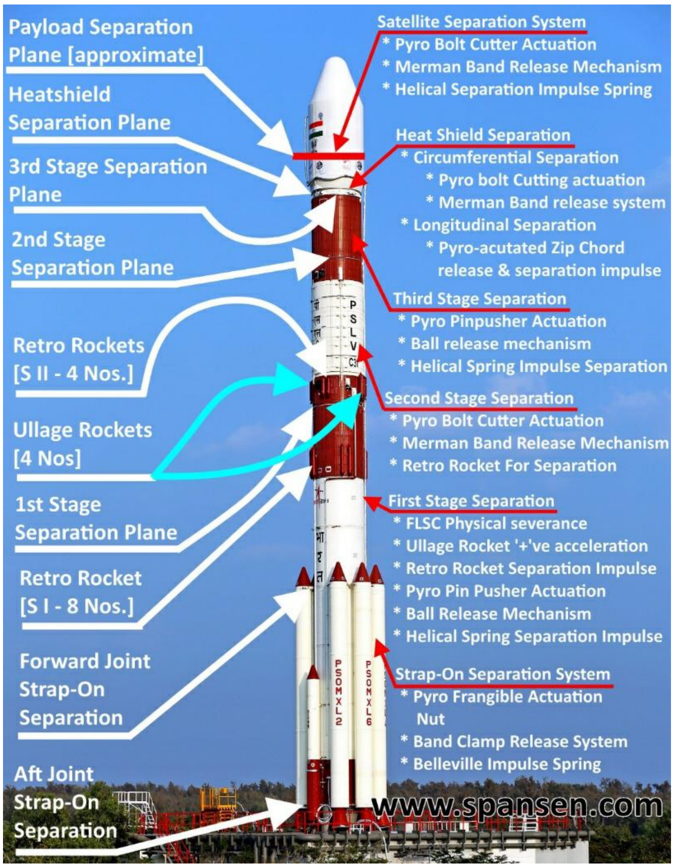


Figure.3.6: PSLV

3.9 New Shepard

Named after Mercury astronaut Alan Shepard, the first American to go to space, New Shepard is a reusable suborbital rocket system designed to take astronauts and research payloads past the Kármán line – the internationally recognized boundary of space. It was developed by Blue Origin.

With room for six astronauts, the spacious and pressurized crew capsule is environmentally-controlled for comfort and every astronaut gets their own window seat. The vehicle is fully autonomous. There are no pilots.

For a typical flight, New Shepard launches vertically and soars for about two and a half minutes before the main engine cuts off and the capsule separates from the rocket. Passengers are weightless for about four minutes during the 11-minute flight, and are high enough (at an altitude of 307,000 feet or 93,573 meters) to see the curvature of Earth. The spacecraft coasts for a few minutes in space before re-entering the atmosphere and using an autonomous, rocket-powered vertical landing system to touch down.

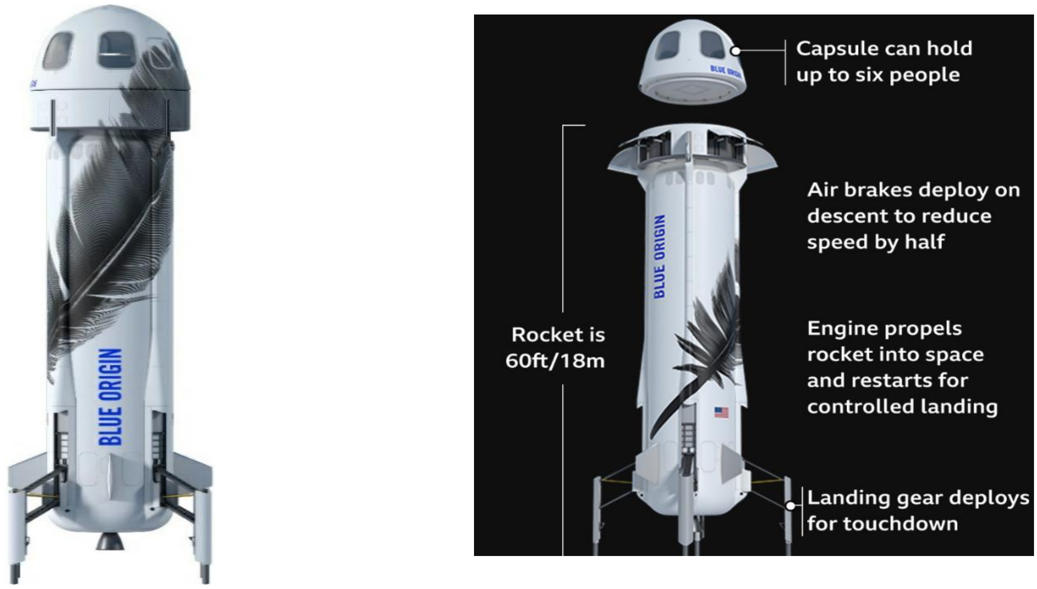


Figure.3.7: New Shepard

3.10 New Glenn

Named after pioneering astronaut John Glenn, New Glenn is a single configuration heavy-lift launch vehicle capable of carrying people and payloads routinely to Earth orbit and beyond. It was developed by Blue Origin. It features a reusable first stage built for 25 missions. New Glenn lifts off from Launch Complex 36 at Cape Canaveral. Following stage separation, the first stage flies back to Earth and lands nearly 1,000 km downrange on a moving ship, allowing the booster to land in heavy sea-states. The second stage engines ignite and the 7-meter fairing separates. The mission is complete when the payload is delivered safely to orbit.

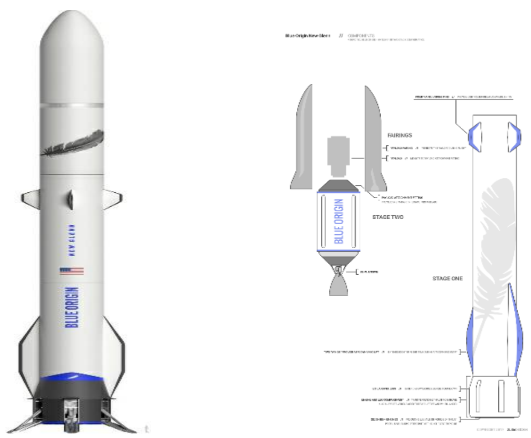


Figure.3.8: New Glenn

3.11 SS-520

The SS-520 is a two-stage rocket, the first stage of which comes from the main booster of the S-520. It has a capability for launching a 140 kg payload to an altitude of about 800 km.

Unlike the previous S-Series rockets, the SS-520 is intended to demonstrate how small an orbital launch vehicle can be.

When used as a suborbital sounding rocket, it can launch a 140 kilograms (310 lb) payload to an altitude between 800 and 1000 km. The first two SS-520s were launched in 1998 and 2000 respectively and successfully carried their payloads on sub-orbital missions.

After further development and the addition of a third stage, the fourth and fifth instances of the SS-520 were launched on full orbital trajectories. As the second stage is heavier than the head of S-520, the aerodynamic margin is secured more than ever. The whole motor case of the second stage is made of CFRP. The spin generated in the first stage is succeeded by the second stage, and it is utilized in the Rhumb-line control and spin stabilization. The SS-520 debuted in January, 1998, and ISAS has a plan to launch it from Spitsbergen, Norway, to send a payload into the cusp region of the geo magnetosphere.

Figure.3.9: SS-520

3.12 JUNO 2

American orbital launch vehicle. Satellite launcher derived from Jupiter IRBM. Basic 4 stage vehicle consisted of 1 x Jupiter + 1 x Cluster stage 2 + 1 x Cluster stage 3 + 1 x RTV Motor Number of Cape Canaveral Launches: 10

A marriage of the Jupiter IRBM as first stage and Juno I upper stages, the Juno II was able to carry a 100-pound payload to low-Earth orbit. A Rocketdyne first stage engine burned liquid oxygen/RP-1 (kerosene) liquid fuel and could produce a thrust of 150,000 pounds at liftoff. The second, third and fourth stage configuration was identical to that of the Juno I. Like the

Juno I, the heavily clustered second and third stages were covered by a rotating “tub” to provide balance and stability. This “tub” was not visible on the Juno II, however, since an outer fairing was incorporated to improve aerodynamics and safety.

The Juno name was also applied to the next generation of space launch vehicles designed by the Army Ballistic Missile Agency (ABMA), including a proposed Juno V super booster. The Juno program was eventually transferred to NASA and renamed Saturn. In a very real sense, the pioneering work of the ABMA, culminating in the successful use of Juno I and Juno II space launch vehicles, led directly to the development of the rockets which would carry men to the Moon.

Figure.3.10: JUNO 2

3.13 DELTA IV HEAVY

American orbital launch vehicle. Heavy lift all-cryogenic launch vehicle using two Delta-4 core vehicles as first stage flanking a single core vehicle as second stage. A heavy upper stage is carried with a 5 m diameter payload fairing.

Capacity of the Delta IV Heavy:

Low Earth orbit (LEO), 200 km × 28.7°: 28,790 kg (63,470 lb)

Low Earth orbit (ISS), 407 km × 51.6°: 25,980 kg (57,280 lb)

Geosynchronous transfer orbit (GTO): 14,220 kg (31,350 lb)

Geosynchronous orbit (GEO): 6,750 kg (14,880 lb)

Lunar transfer orbit (LTO): 10,000 kg (22,000 lb)

Mars transfer orbit: 8,000 kg (18,000 lb)

The Delta IV Heavy's total mass at launch is approximately 733,000 kg (1,616,000 lb) and produces around 952,000 kg (2,099,000 lb) of thrust to power the rocket skyward at liftoff.

Stage Data - Delta IV Heavy

Stage 0. 2 x Delta RS-68. Gross Mass: 226,400 kg (499,100 lb). Empty Mass: 26,760 kg (58,990 lb). Thrust (vac): 3,312.755 kN (744,737 lbf). Isp: 420 sec. Burn time: 249 sec. Isp(sl): 365 sec. Diameter: 5.10 m (16.70 ft). Span: 5.10 m (16.70 ft). Length: 40.80 m (133.80 ft). Propellants: Lox/LH2. No Engines: 1. Engine: RS-68. Status: In production. Comments: Low-cost expendable stage using lower performance engine. Used in Delta 4, Boeing EELV. Engine can be throttled to 60%.

Stage 1. 1 x Delta RS-68. Gross Mass: 226,400 kg (499,100 lb). Empty Mass: 26,760 kg (58,990 lb). Thrust (vac): 3,312.755 kN (744,737 lbf). Isp: 420 sec. Burn time: 249 sec. Isp(sl): 365 sec. Diameter: 5.10 m (16.70 ft). Span: 5.10 m (16.70 ft). Length: 40.80 m (133.80 ft). Propellants: Lox/LH2. No Engines: 1. Engine: RS-68. Status: In production. Comments: Low-cost expendable stage using lower performance engine. Used in Delta 4, Boeing EELV. Engine can be throttled to 60%. Stage 2. 1 x Delta 4H - 2. Gross Mass: 30,710 kg (67,700 lb). Empty Mass: 3,490 kg (7,690 lb). Thrust (vac): 110.050 kN (24,740 lbf). Isp: 462 sec. Burn time: 1,125 sec. Diameter: 2.44 m (8.00 ft). Span: 5.00 m (16.40 ft). Length: 12.00 m (39.00 ft). Propellants: Lox/LH2. No Engines: 1. Engine: RL-10B-2. Status: In production. Comments: Delta 4 second stage with hydrogen tank increased to 5.1 m diameter.

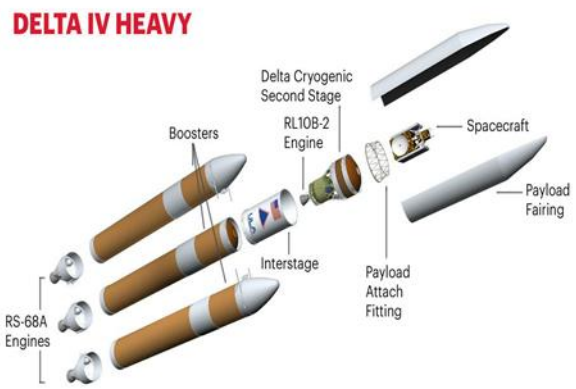


Figure.3.11: Delta IV Heavy

3.14 THOR ABLE

The Thor-Able was an American expendable launch system and sounding rocket used for a series of re-entry vehicle tests and satellite launches between 1958 and 1960. It was a two-stage rocket, consisting of a Thor IRBM as a first stage and a Vanguard-derived Able second stage.

Introduced in 1960, Thor-Able Star was a two-stage vehicle designed to carry military payloads into orbit. Employing improved Thor-Able first and second stages and eliminating the weight of additional upper stages, Thor-Able Star was able to carry a maximum 1,000-pound payload into low-Earth orbit. A Rocketdyne first stage engine burned liquid oxygen/RP-1 (kerosene) liquid fuel and could produce 172,000 pounds of thrust at launch. An Aerojet second stage engine burned IRFNA/UDMH solid fuel and could produce a 7,730-pound thrust.

This upgraded engine could be re-started to augment and adjust the orbit of the payload. Thor-Able Star was the last version of the Thor-based rockets to be launched from Cape Canaveral carrying the “Thor” name. In 1960, an improved Thor-based rocket called the Thor-Delta was introduced. The name of this vehicle was officially shortened to Delta in order to distinguish it from military relatives which continued to carry the Thor name. From this point on, all Thor-based rockets launched from the Cape carried the Delta name only. However, military vehicles carrying the Thor name continued to be launched from Vandenberg Air Force Base, California. Rockets launched from Vandenberg Air Force Base, California, but never launched from Cape Canaveral, include the Thor-Agena A, Thor-Agena B, Thor-Agena D, Thor-Altair, Thor-Burner II and Thor-Burner IIA.

Indeed, military variants of the Thor-based rockets continued to evolve following the elimination of the Thor name from Cape Canaveral launches. This evolution affected the Delta fleet as well. The Thrust Augmented Thor (TAT) Agena D was introduced in 1963 by adding three Castor solid rocket boosters to augment the first stage of the Thor-Agena D.

This method of thrust augmentation was incorporated into the Delta program beginning with the introduction of the Delta D launch vehicle in 1964. The TAT-Agena D was later improved by lengthening the first stage fuel tanks. The resulting rocket was called Long Tank Thrust Augmented Thor (LTTAT) Agena D. The LTTAT-Agena D was alternately referred to by the shortened names Thorad and Long Tank Thor. The vehicle remained in operation until 1972. From 1972 on, military and civilian satellites were both launched on vehicles bearing the Delta name. It should be noted that although the Thor designation was officially removed from the Delta fleet in 1972, some sources continued to refer to the “Thor-Delta” in reference to Delta launches taking place well beyond that date.

The first launch attempt on April 23 ended at T+146 when the Thor exploded due to a turbopump main bearing failure, and mouse Mia (inauspiciously named 'Missing in Action') perished. The same problem had occurred on other Thors and it took a while to identify the cause, take corrective action, and fix the turbopumps in the field. The second launch on July 9, with a ballistic nose cone and the Mouse Mia II aboard, was successful, although the nose cone could not be located by the recovery ship before it sank.

On the third launch, on July 23, telemetry indicated that mouse Wickie survived the flight into space through splashdown. But again, the nose cone was not located. This completed the nose cone tests. In February it had been suggested that the configuration could be converted into a low-cost ICBM, dubbed Thoric (Thor Inter-Continental). This was an obvious threat to the Atlas program and got nowhere.

Figure.3.12: Thor Able

Thor B or Thor Baker was to use the three-stage combination originally proposed in October 1957 to allow the Air Force to send the first space probe to the moon. It was renamed Thor Able-I and the first launch attempt on August 17 was thwarted when the first stage failed at T+70 seconds. The next month NASA took over the program, and the Thor Able-I's payloads became the first of NASA's long series of Pioneer deep-space probes. The USAF used an upgraded version of the Aerojet Able stage for another series of reentry vehicle tests in 1959, while NASA and DARPA continued to use developed versions of the rocket for launch of Explorer, Transit, and Tiros satellites.

Finally, the addition of solid rocket boosters allowed larger upper stages to be carried and payload to be increased in the Delta series of rockets. The evolutionary descendant of the Thor Able, the Delta 7000, was still flying in the 21st Century. Payload had increased from 100 kg to over 5 metric tons, and the Delta was the most reliable and economical launch vehicle ever produced in the United States.

3.15 PROTON M

The upgraded Proton M inherited a lot from the highly reliable Proton K. More than 77 percent of parts, subassemblies and systems have transferred 'off the shelf', 18 percent of subassemblies or systems have upgraded and only 5 percent is new.

The use of larger payload fairings (including those with a diameter of 5 m) in the Proton M configuration will more than double the usable volume and make the new LV competitive with its foreign counterparts including Ariane 5. The larger PLF will also make it possible to employ a number of new and promising upper stages as part of the new launch vehicle

The key task under this upgrade program is to replace the navigation system. Developed way back in the 1960s this GN&C is outdated in terms of both its design philosophy and the components used. Moreover, this navigation system has been produced outside Russia.

A new and improved computerized navigation system is installed at the upgraded launcher. Major components of this system have been successfully tested in flight on other launchers and are widely used. The new system will solve several significant problems, more specifically it will

- Make propellants utilization more efficient due to more complete depletion thereby increasing LV performance while eliminating if not totally ruling out hazardous residues;
- Enable 3D maneuvering during powered flight, which will expand the range of feasible parking orbit inclinations;
- Simplify the avionics since computations performed until now by the depletion system and the safety system will in future be relegated to the onboard computer;
- Limit the dynamic pressure times pitch (yaw) product in flight, which will open the way for larger payload fairings without sacrificing noticeably the LV load capability;
- Enable on-line loading or updating the mission definition; and improve the mass properties of the launch vehicle

Another advantage will be drastically reduced drop fields, an extremely important issue now that the Stage 1 drop fields are located in the Republic of Kazakhstan and are leased by Russia. A reduction in the size of drop fields will be achieved through controlled landing of Stage 1 onto a limited area.

In addition to reduced rent payments the smaller drop fields will alleviate Stage 1 debris recovery. Moreover, the stage will reach the ground virtually clean since the Proton M Stage 1 engine timeline will ensure complete depletion of propellants. Thus, the new Russian launcher will be much more environment-friendly.

Another advantage of integrating the Breeze M upper stage with NTO and UDMH as propellants will be that the weight of payloads to be injected into a geostationary orbit will be brought up to 3300 kg.

Figure.3.13: Proton M

CHAPTER 04

PREPARATION OF COMPARATIVE DATA SHEET OF DIFFERENT ROCKETS

4.1 General Characteristics

Table 4.1 General Characteristics

Rocket	Height (m)	Diameter (m)	Crew	Stages	Powerplant	Propellant
X-33	21.0312	23.4696	N/A	1	2 XRS-2200 linear aerospike rocket engines	LH2/LOX
Skylon	83.133	26.818	24	1	2 × SABRE 4 liquid air cycle engine	LH2/LOX
Roton	19.5072	6.7056	2	1	1 x rotary RocketJet™ aerospike engine and 72 Fastrac Engines	LOX/Kerosene
Rombus	29	24	6	1.5	36 x plug-nozzle engines	LH2/LOX
Atlas B	26	3.05	N/A	1.5	2 x XLR-89-5 1 x XLR-105-5	LOX/Kerosene
Saturn V	110.6	10.1	3	2-3	1st Stage: 5 Rocketdyne F-1 2nd Stage: 5 Rocketdyne J-2 3rd Stage: 1 Rocketdyne J-2	1st Stage: RP-1/LOX 2nd & 3rd Stage: LH2/LOX
Falcon 9	70	3.7	4	2	1st Stage: 9 Merlin 1D+ 2nd Stage: 1 Merlin 1D+ Vacuum	1st and 2nd Stage: LOX/RP-1

PSLV	44	2.8	N/A	4	1st Stage: 1 S139 2nd Stage: 1 Vikas 3rd Stage: 1 S-7 4th Stage: 2 L-2-5	1st Stage: HTPB 2nd Stage: N2O4/UDMH 3rd Stage: HTPB 4th Stage: MMH/MON
New Shepard	18	1.8	6	1.5	1 BE-3	LH2/LOX
New Glenn	98	7	N/A	2	1st Stage:7 BE-4U 2nd Stage: 2 BE-3U	1st Stage: CH4/LOX 2nd Stage: LH2/LOX
SS-520	9.70	0.52	N/A	2	N/A	BP-202J
JUNO-2	24.0	4.47	N/A	4	1st stage: Rocketdyne S-3D 2nd stage: 11 solid 3rd stage: 3 solid 4th stage: 1 solid	RP-1/LOX Solid - Polysulfide-aluminum and ammonium perchlorate
Delta lV Heavy	72	5	N/A	2	RS-68A	LH2/LOX
Thor-able	26.9	2.44	N/A	3	1st stage: LR79-7 2nd stage: AJ-10 3rd-stage: X-248	RP-1/LOX HNO3/UDMH SOLID
Proton-M	58.2	7.4	N/A	4	1st stage: RD-275M 2nd stage: 3 RD-0210 1 RD-0211 3rd stage: RD-0212 4th stage: RD-58M/RD-58MF	N2O4 / UDMH

4.2 Weight Configuration

Table 4.2 Weight Configuration

Rocket	Empty Weight (kg)	Gross Weight (kg)	LEO Payload (kg)
X-33	28,440	123,965	N/A
Skylon	53,400	325,000	12,000kg to 300 km 9,500kg to 460 km at 28.5 deg
Roton	4399.84599	181436.948	3200kg to 300 km at 50 deg
Rombus	3,24,318	63,63,000	450000 kg to 185 km at 28 deg
Atlas B	7,030	110,740	70kg to 185 km at 32 deg
Saturn V	185,300	2,822,000-2,965,000	140000 kg to 170 km at 30 deg
Falcon 9	29,500	549,054	10,450kg to 200 km at 28 deg
PSLV	39,530	295,000	3,800kg to 200km at 49.5deg
New Shepard	20,569	75,000	N/A
New Glenn	N/A	N/A	45,000kg
SS-520	787	2600	140kg
Juno-2	5443	55,110	41kg to 200km

Delta lV Heavy	46,000	733,000	25,800 kg to a 185 km orbit at 28.50 degrees. 10,843 kg to a GTO, 27 deg
Thor Able	3125	51,608	120 kg to a 640 km orbit at 48.00 degree
Proton-M	48,765	705,000	LEO Payload: 21,000 kg Payload: 4,500 kg to a GTO.

4.3 Performance

Table 4.3 Performance

Rocket	Max. Speed (Mach No. or km/h)	Max. Thrust (kN)	Specific Impulse (km/s or sec)	Burn Time (sec)	Apogee (km)	Max. Range (km)
X-33	Mach 15	Sea level: 1823.77086 Vacuum: 2384.24679	Sea level: 3.32km/s Vacuum: 4.13km/s	1,600	96	1530
Skylon	Mach 27.8	4000	Air breathing: 4100-9200 s Rocket mode:460 s	1,200	300	N/A
Roton	N/A	2224.11081	340 s	253	300	190
Rombus	N/A	79769	455 s	215	185	N/A
Atlas B	N/A	1587.20	309 s	375	185	6000
Saturn V	N/A	1st Stage: 35,100 2nd Stage: 5,141 3rd Stage: 1,033.2	1st Stage: 2.58km/s 2nd Stage: 4.13km/s 3rd Stage: 4.13km/s	1st Stage: 168 2nd Stage: 360 3rd Stage: 165+335	175	1504.74

Falcon 9	N/A	1st Stage:7600 2nd Stage:934	1st Stage: Sea level: 2.77km/s Vacuum: 3.05km/s 2nd Stage: 3.41km/s	1st Stage: 162 2nd Stage: 397	200	N/A
PSLV	27,000 km/h	1st Stage: 4846.9 2nd Stage: 803.7 3rd Stage: 240 4th Stage:14.66	1st Stage: Sea level: 2.32km/s Vacuum: 2.64km/s 2nd Stage: 2.87km/s 3rd Stage: 2.89km/s 4th Stage: 3.02km/s	1st Stage: 110 2nd Stage: 133 3rd Stage: 83 4th Stage: 525	200	N/A
New Shepard	3,568 km/h	490	2560 s	141	101.7	N/A
New Glenn	N/A	1st Stage: 17100 2nd Stage: 1400	1st Stage:335 s 2nd Stage: 440 s	1st Stage: 240 2nd Stage: 500	200	N/A
SS-520	27,000 km/h	185	265 s	31.7	1000	800
Juno-2	265,000 around Jupiter relative to earth	667.20	220 s	200	200	N/A
Delta lV Heavy	N/A	8670	Sea level:360s (3.5 km/s) Vacuum:412s (4.04 km/s)	1st stage: 328 2nd stage: 462	185	N/A
Thor able	N/A	1st stage: 758.711 2nd stage: 34.300	1st stage: 248 s 2nd stage: 270 s	1st stage: 165 2nd stage: 115	640	N/A

Proton-M	6465.6	1st stage: 10,532 2nd stage: 2399.216 3rd stage: 630.170 4th stage: 19.6	1st stage: 285 s 2nd stage: 327 s 3rd stage: 325 s 4th stage: 326 s	1st stage: 108 2nd stage: 206 3rd stage: 238 4th stage: 3000	40,000	N/A

4.4 Two Stage Rocket profiles

Table 4.4 Mission Profile for New Glenn

New Glenn	**Total mass (kg)**	**Dry mass (kg)**	**Propellant mass (kg)**	**Altitude (km)**	**Thrust (KN)**	**Burn time (s)**	**Diameter (m)**	**Length (m)**	**Propellant**
1st stage	11,10,000	1,10,000	10,00,000	90	2,041	240	7	57.5	LOX, LNG
2nd stage	1,20,000	12,000	1,08,000	275	110	500	7	16.1	LOX, LH2
Payload fairing	6,000	-	-	121	-	-	7	21.9	

Table 4.5: Mission Profile for Saturn V

Saturn V	**Total mass (kg)**	**Dry mass (kg)**	**Propellant mass (kg)**	**Altitude (km)**	**Thrust (KN)**	**Burn time (s)**	**Diameter (m)**	**Length (m)**	**Propellant**
1st stage	22,78,059	1,30,929	21,47,130	61	33,361	168	29.3	42.1	LOX, RP1
2nd stage	4,80,996	36,387	444,609	185	4,448	366	10.1	24.8	LOX, LH2
3rd stage	1,18,648	11,362	107,286	180	889	144 (1st burn) 336 (2nd burn)	6.6	18.1	LOX, LH2
Instrument Unit	1,953	1.953	-	-	-	-	6.6	0.9	

Table 4.6: Mission Profile for Falcon 9

Falcon 9	**Total mass (kg)**	**Dry mass (kg)**	**Propellant mass (kg)**	**Altitude (km)**	**Thrust (KN)**	**Burn time (s)**	**Diameter (m)**	**Length (m)**	**Propellant**
1st stage	404000	19000	385000	80	SL:588 Vac:6672	185	3.66	40.9	LOX, RP1
2nd stage	99000	4500	93000	199.2	Vac:800	375	3.66	14.6	LOX, RP1
Payload Fairing	2000	-	-	-	-	-	5.2	13.9	-

Table 4.7: Mission Profile for Delta IV Heavy

Delta IV Heavy	**Total mass (kg)**	**Dry mass (kg)**	**Propellant mass (kg)**	**Altitude (km)**	**Thrust (KN)**	**Burn time (s)**	**Diameter (m)**	**Length (m)**	**Propellant**
1st stage	226400	26000	200400	185	3140 KN	328	5.1	40.8	LH2 / LOX
2nd stage	30700	3480	27220	324	110 KN	1125	5.1	13.7	LH2 / LOX
Booster (CBC)	226400	26000	200400	99	3140 KN	242	5.1	40.8	LH2 / LOX

Table 4.8: Mission Profile for SS-520

SS-520	**Total mass (kg)**	**Dry mass (kg)**	**Propellant mass (kg)**	**Altitude (km)**	**Thrust (KN)**	**Burn time (s)**	**Diameter (m)**	**Length (m)**	**Propellant**
1st stage	1985	498	1587	83	143KN	31.7	0.52	5.8	BP-202J
2nd stage	503	178	325	186	41KN	24	0.52	3.9	BP-202J
Payload Fairing	78	-	-	205	-	-	-	-	-

Table 4.9: Rocket Mass profile

Rocket	**Payload Mass (M_{PL})**	**Total Mass (M_{tot})**	$\pi_{PL} = M_{PL}/M_{tot}$
X-33	N/A	123,965 kg	-
Skylon	12,000 kg	325,000 kg	0.0369
Roton	3,200 kg	181,437 kg	0.0176
Rombus	450,000 kg	6,363,000 kg	0.0707
Atlas B	70 kg	110,740 kg	0.0006
Saturn V	140,000 kg	2,965,000 kg	0.0472
Falcon 9	10,450 kg	549,054 kg	0.0190
PSLV	3,800 kg	295,000 kg	0.0128
New Shepard	N/A	75,000 kg	-
New Glenn	45,000 kg	1,236,000 kg	0.0364
SS-520	140 kg	2,600 kg	0.0538
Juno-2	41 kg	55,110 kg	0.0007
Delta lV Heavy	25,800 kg	733,000 kg	0.0352
Thor Able	120 kg	51,608 kg	0.0023
Proton-M	21,000 kg	705,000 kg	0.0297

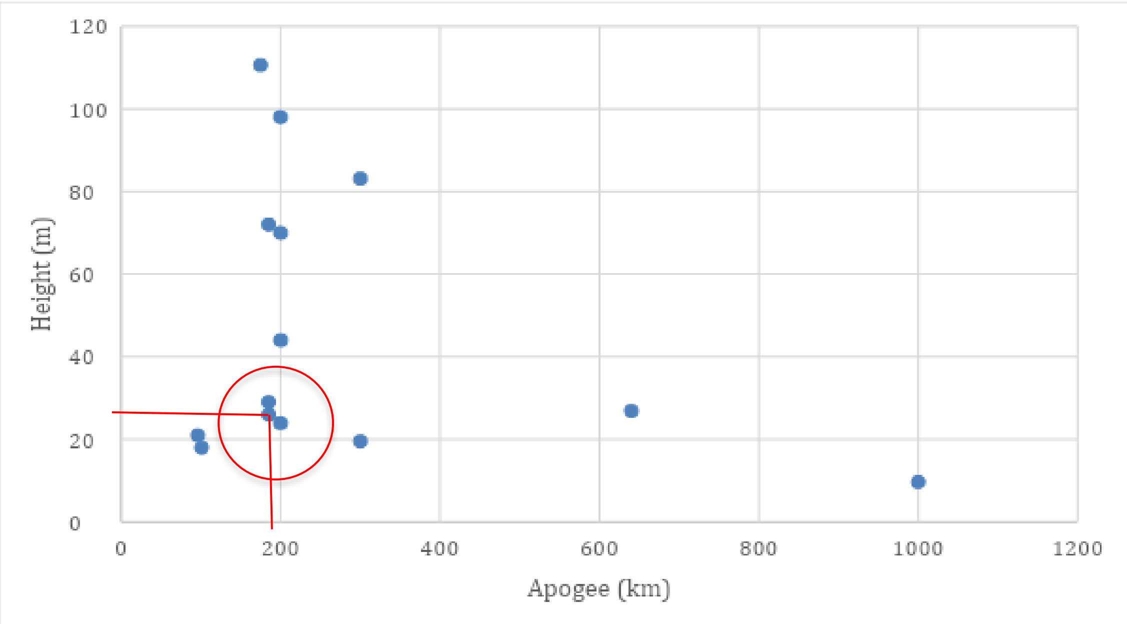
CHAPTER 05

PREPARATION OF COMPARATIVE GRAPHS

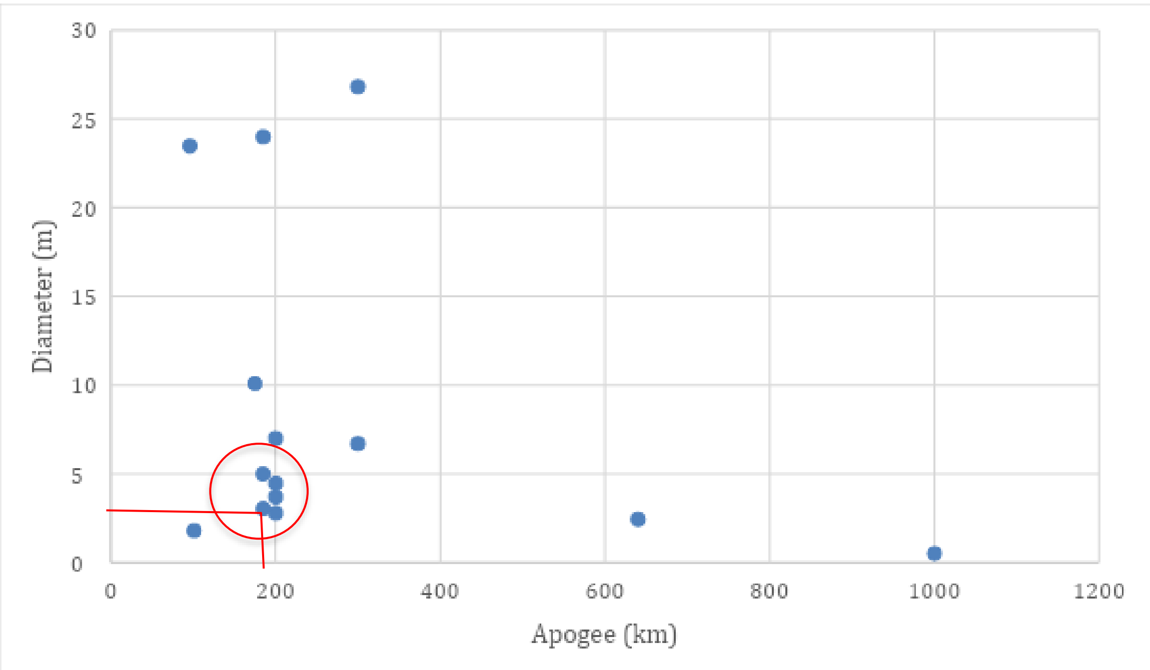
5.1 GRAPH PLOTTING

The graphs are plotted for the tabulated data from the previous section. The inference from these graphs will give the tentative design parameters.

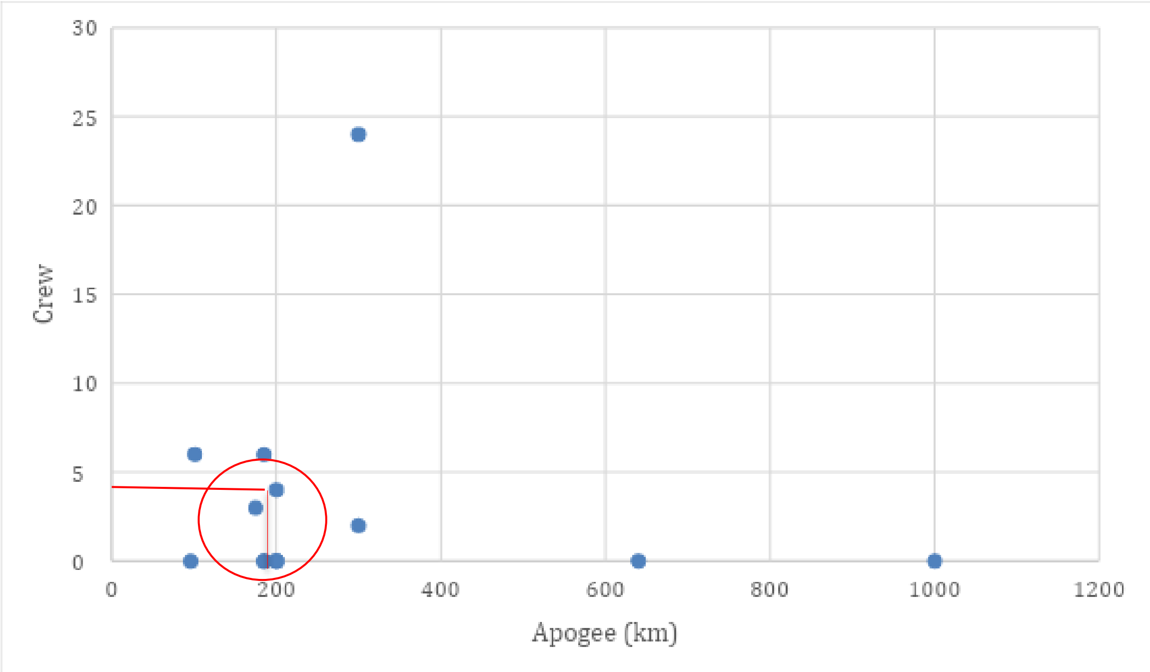
5.1 APOGEE vs HEIGHT



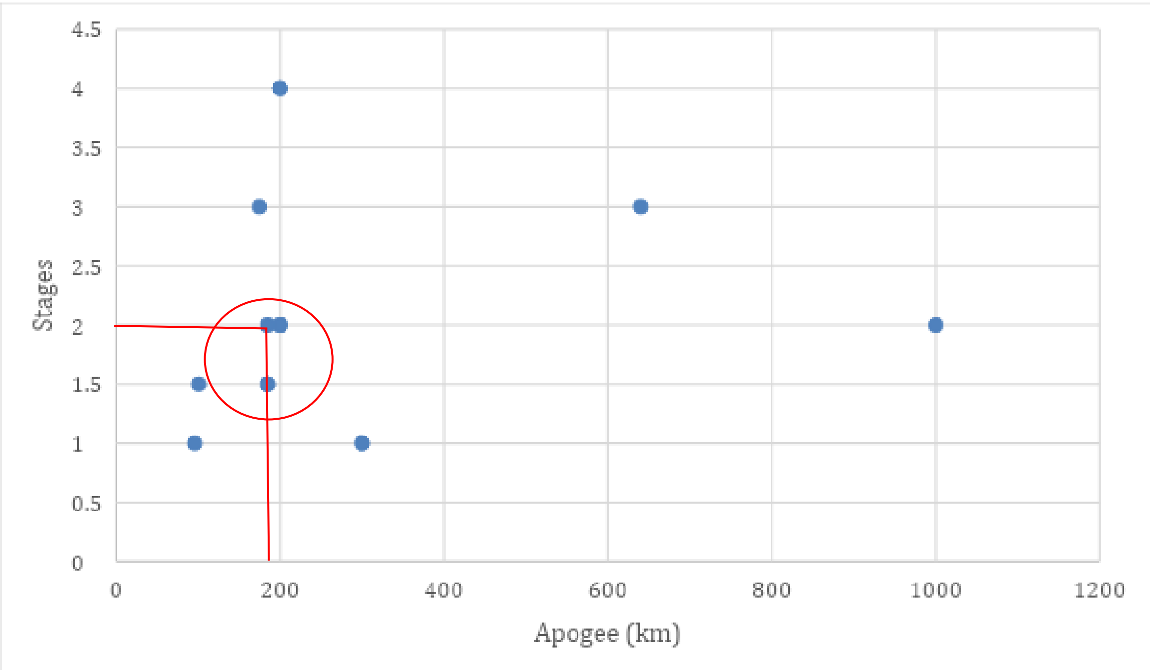
5.2 APOGEE vs DIAMETER



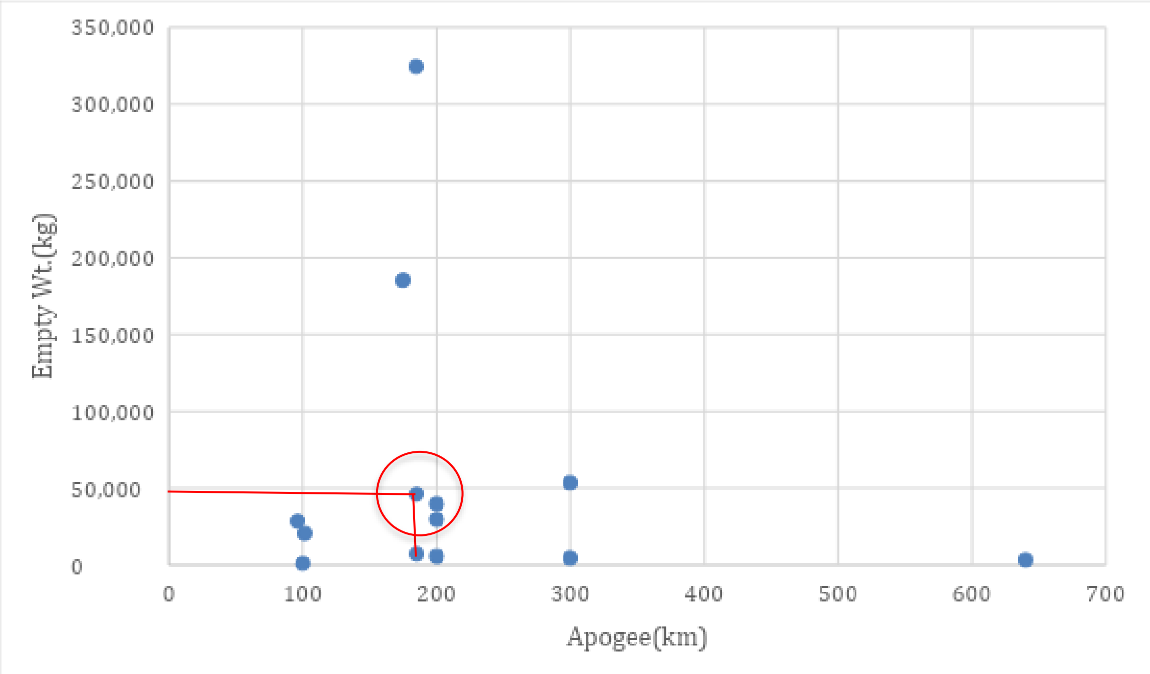
5.3 APOGEE vs CREW



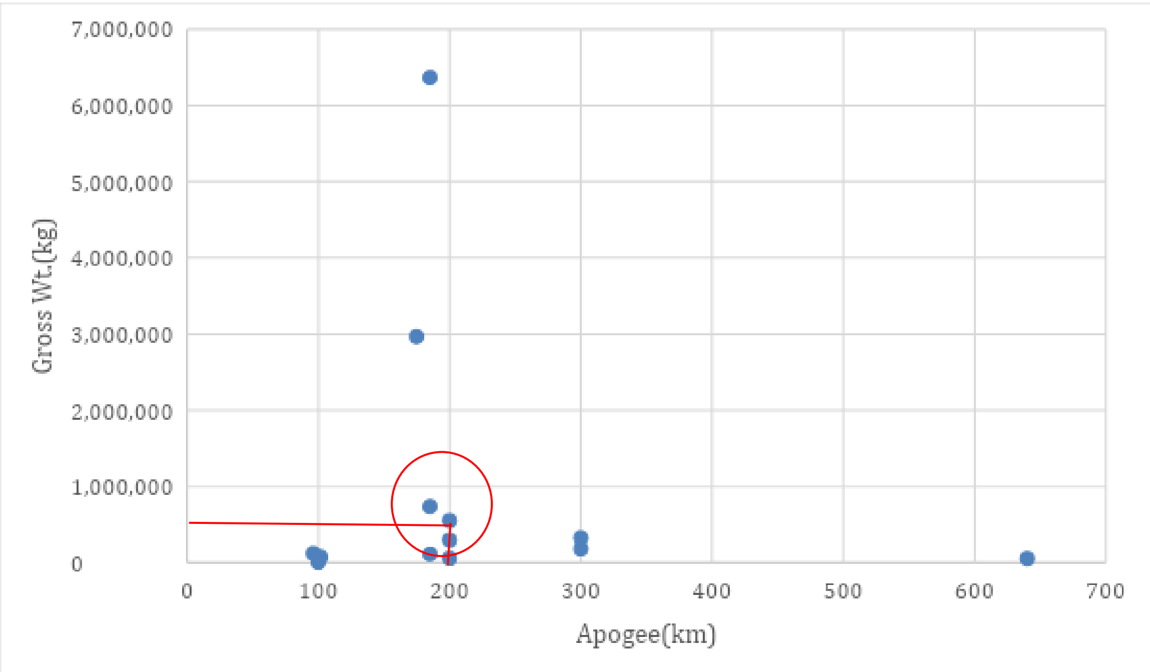
5.4 APOGEE vs STAGES



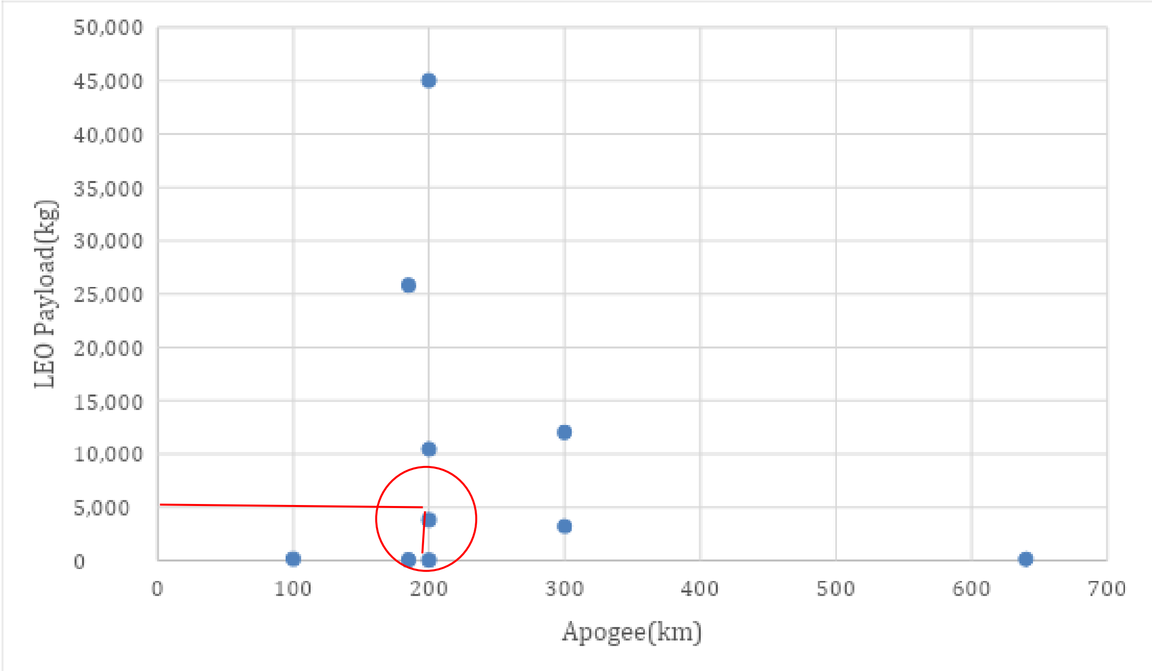
5.5 APOGEE vs EMPTY WEIGHT



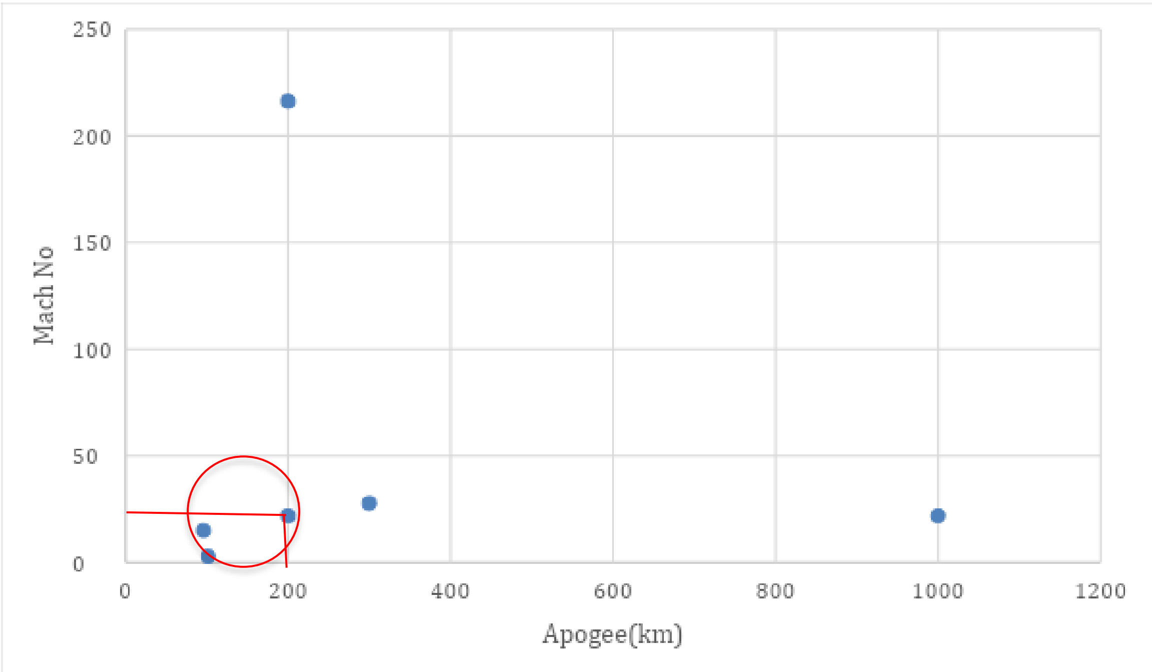
5.6 APOGEE vs GROSS WEIGHT



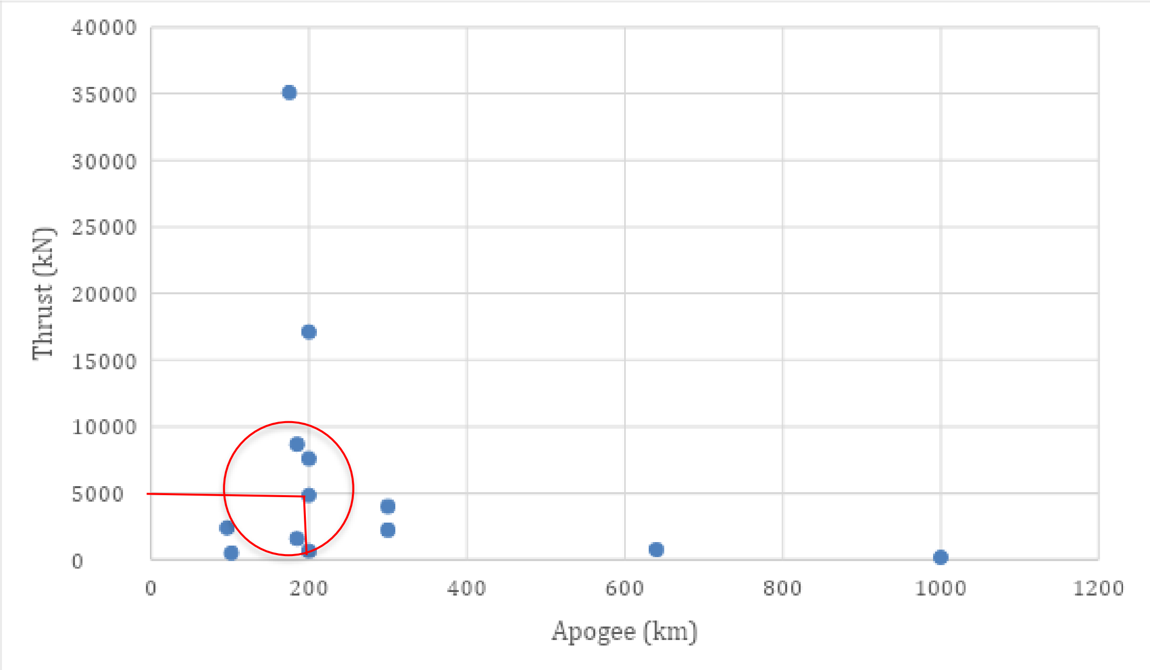
5.7 APOGEE vs LEO PAYLOAD



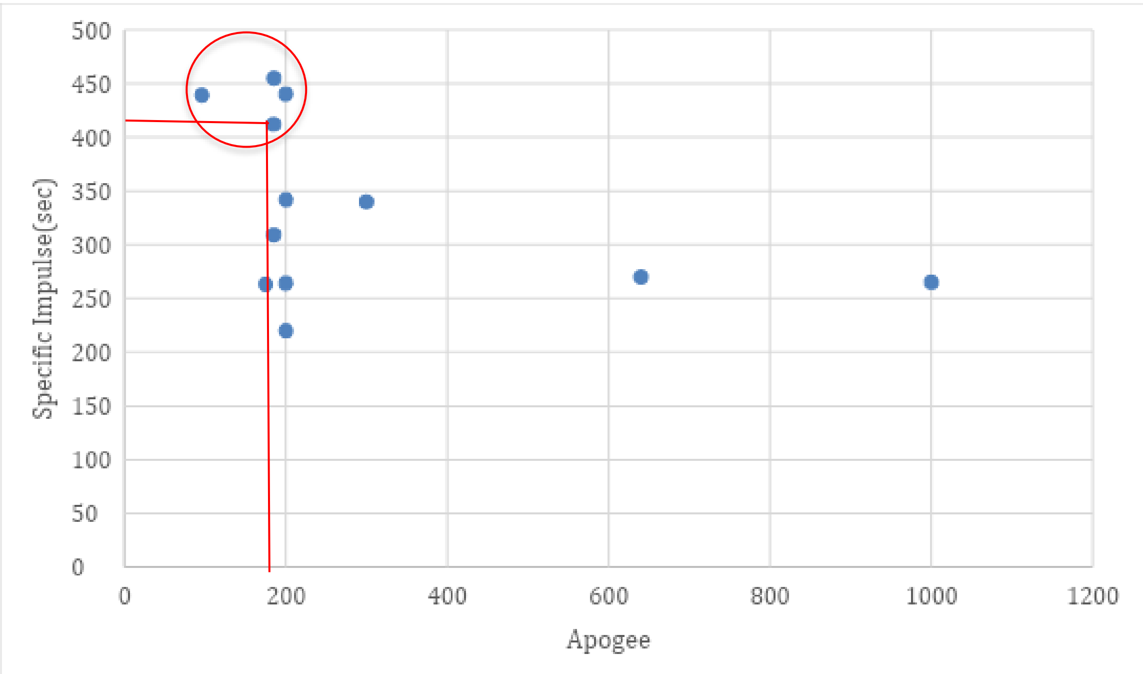
5.8 APOGEE vs MACH No.



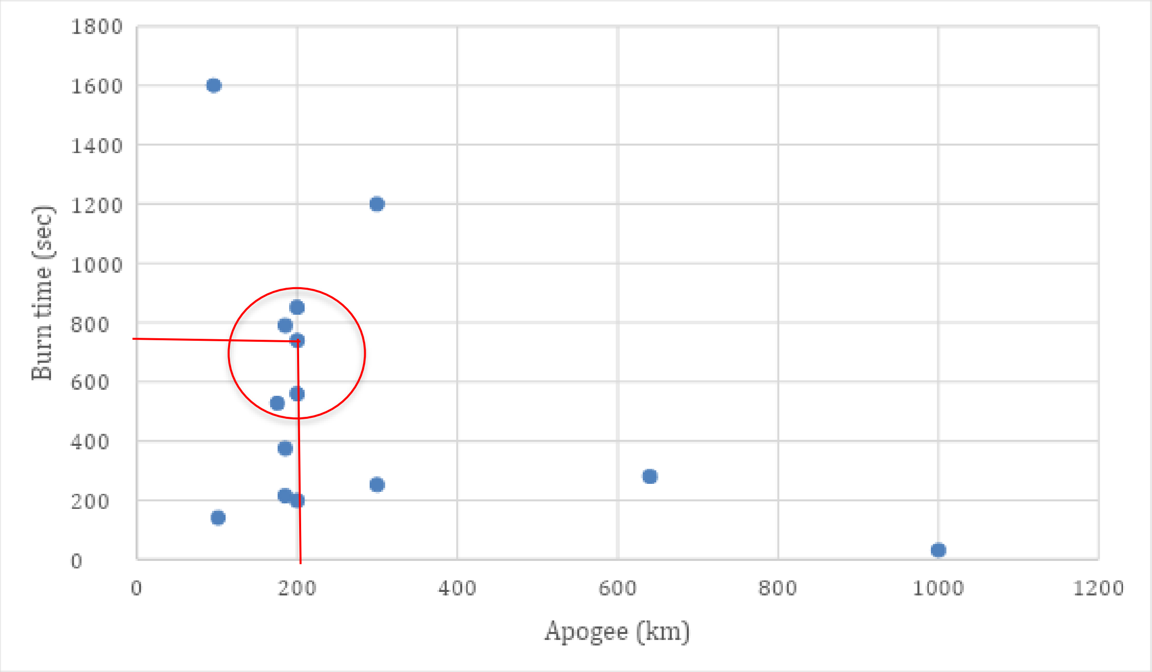
5.9 APOGEE vs THRUST



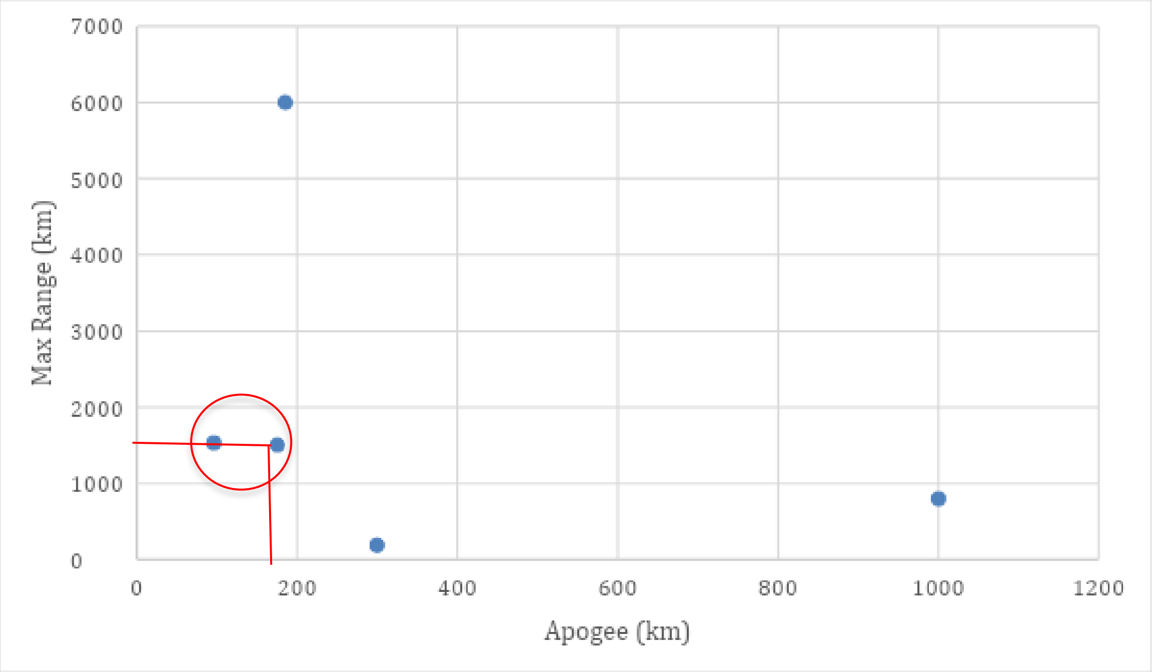
5.10 APOGEE vs SPECIFIC IMPULSE



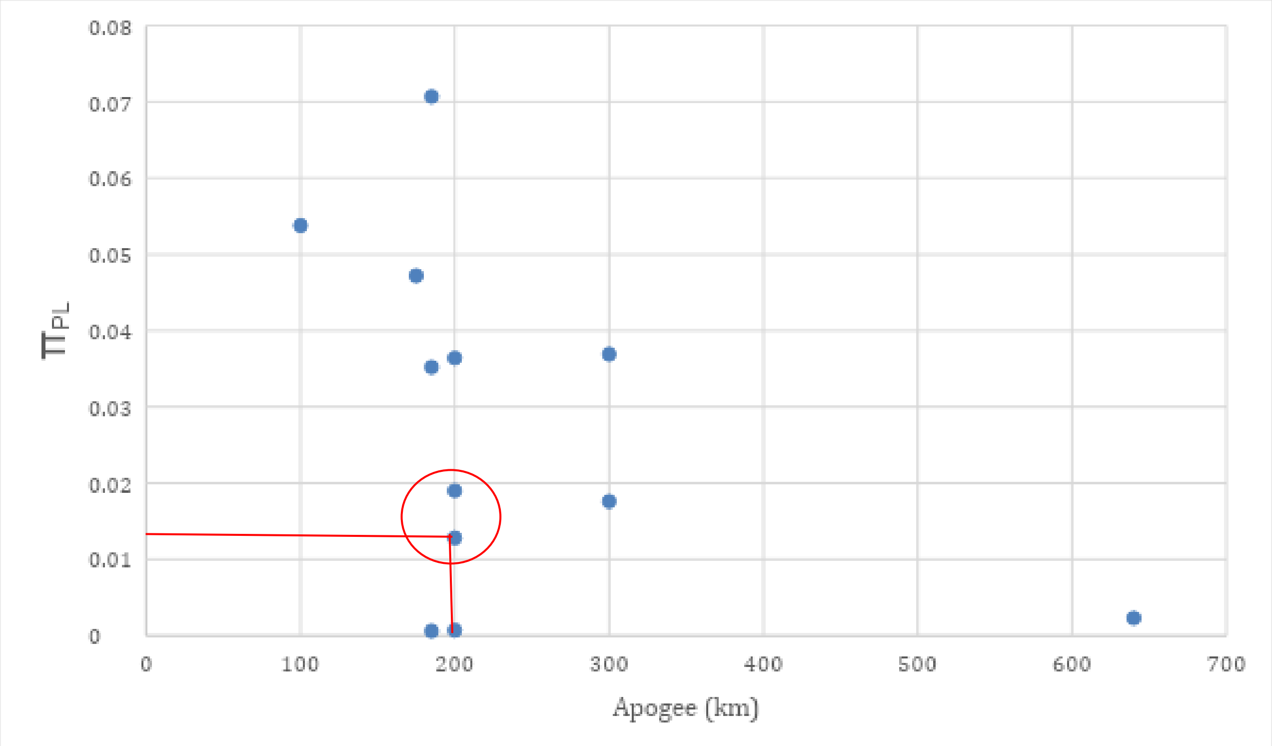
5.11 APOGEE vs BURN TIME



5.12 APOGEE vs MAX. RANGE



5.13 APOGEE vs PAYLOAD MASS RATIO



CHAPTER 06

SELECTION OF TENTATIVE DESIGN PARAMETERS

6.1 Tentative Design Parameters

Based on the literature review as well as the graphs plotted from the tabulated data of the selected 15 rockets, the tentative design parameters are selected for the rocket. These parameters will be used to proceed with the design steps and weight estimation processes.

6.1.1 General Characteristics

The tentative parameters for the general characteristics for the design are:

1. Height : 38.5 m
2. Diameter : 6 m
3. Crew : 0
4. Stages : 2

6.1.2 Weight Configuration

The tentative parameters for the weight configuration for the design are:

1. Empty Wt. : 59,874 kg
2. Gross wt. : 5,00,000 kg
3. LEO Payload : 5000 kg
4. π_{PL} : 0.01

6.1.3 Performance

The tentative parameters for the performance for the design are:

1. Mach no. : 23
2. Thrust : 5000 kN
3. Specific Impulse : 410 sec
4. Burn Time : 745 sec
5. Max. Range : 1500 km
6. Apogee : 200 km

CHAPTER 07
WEIGHT ESTIMATION

7.1 MISSION PROFILE

Mission profile can be defined as the trajectory of the rocket or the flight plan which consists of the altitude, speed, distance of the rocket, the maneuvers to be performed, the number of stops etc. A flight plan plays a very important role as it helps us to be prepared in advance. The following is the mission profile of our rocket.

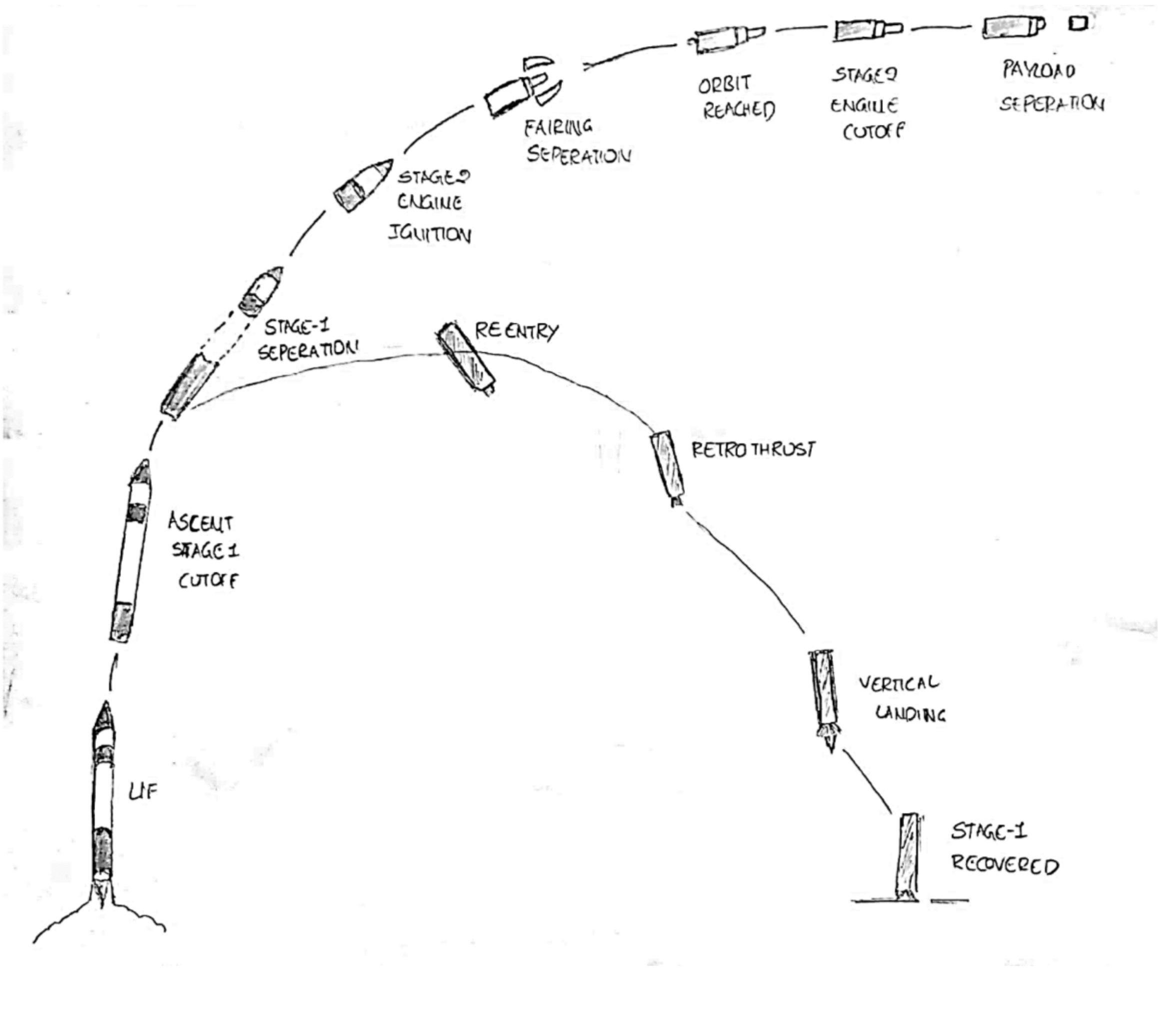


Figure.7.1: Mission Profile

1. ENGINE IGNITION & LIFTOFF:

Main engine ignition and rocket lifts off from the ground. The rocket on the launch pad is balanced. The surface of the pad pushes the rocket up while gravity tries to pull it down. The electrical resistance of the igniter wire causes heat as the current passes through. That heat is enough energy to push what's called the “pyrogen” into ignition. As the engines are ignited, the thrust from the rocket unbalances the forces, and the rocket travels upward full-scale launchers rely on a sophisticated guidance system to balance and steer the rocket during its flight. The thrust of the rocket is gimbaled, or rotated, during the flight to produce maneuvers.

2. ASCENT & STAGE 1 CUTOFF:

The rocket ascends to 82.6km & stage 1 engine is cut off. Leaving the pad, the rocket begins a powered vertical ascent. The first stage is ignited at launch and burns through the powered ascent until its propellants are exhausted. The vehicle accelerates because of the high thrust and decreasing weight and rather quickly moves out of the thick atmosphere near the surface of the earth. Although the rocket is traveling supersonically, the drag on the vehicle is small because of the shape of the rocket and the lower air density at altitude. As the rocket ascends, it also begins to pitch over and its flight path becomes more inclined to the vertical. The first stage engine is then extinguished.

3. STAGE 1 SEPARATION

The first stage of the rocket separates. Several minutes into the ascent, most launchers discard some of the weight of the rocket. The discarded first stage continues on a ballistic flight back to earth. The first stage may be retrieved, as with the Space Shuttle solid rocket engines, or it may be completely discarded.

4. STAGE 2 ENGINE IGNITION

The second stage engine ignites. The second stage separates from the first stage, and the second stage engine is ignited. The payload is carried atop the second stage into orbit. The second stage continues to accelerate under the power of its engine and to pitch over to the horizontal.

5. STAGE 1 RE-ENTRY BURN

The first stage engine produces retro thrust for landing. Immediately after stage separation, the first stage or booster goes through what is called a "flip maneuver". This is done by using cold gas thrusters located at the top of the booster. This rotates the booster by ~180 degrees and gets it oriented in the right direction for the upcoming engine burns.

Next in the sequence is Boostback burn. In this 3 of the engines are ignited for a few seconds and the booster trajectory is reversed. Usually, the booster reaches up to a height of ~200 km before ascending down. The descent happens a few seconds after the boostback burn.

After this, the booster descends for a few more seconds and soon is followed by "Entry burn", in which 3 engines ignite again to slow down the speed of the booster. This happens at a height of ~ 55 km

Finally, just a few meters away from landing the final "landing burn" is executed using only one of the Merlin engines. Keep in mind that throughout the journey grid fins are deployed that continuously control and orient the booster through minute changes in its aerodynamic behavior.

Typically, 6-10% of the total fuel mass is required for executing all three re-entry burns. If the landing is occurring offshore, then the fuel needed is ~6%. In case, the 1st stage has to be taken to the launch site, then ~10% of the total fuel is used.

6. FAIRING SEPARATION

The metal shell or fairings, covering the spacecraft separate and are discarded. A vertical separation system splits the fairing in two when it separates from the launcher by the means of a pyrotechnic cord incorporated into one side of the shell edge. When this is detonated, a flexible fully sealed sleeve pushes the other half of the shell away.

7. STAGE 1 LANDING

The first stage of the rocket is landed.

8. ORBIT REACHED

The vehicle reaches Low Earth Orbit.

9. STAGE 2 ENGINE CUTOFF

The second stage engine is cut off at an altitude of 233.84km. At a carefully determined altitude and speed the upper stage engine is cut off and the stage and payload are in orbit. The exact speed needed to orbit the earth depends on the altitude, according to a formula that was developed by Johannes Kepler.

10. PAYLOAD SEPARATION

The payload separates from the second stage and the second stage is discarded.

7.2 WEIGHT PROFILE

1^{st} stage total mass = 417220.23 kg

1^{st} stage empty mass = 46451 kg

1^{st} stage propellant mass = 364220.23 kg

2^{nd} stage total mass = 77779.77 kg

2^{nd} stage empty mass = 13423 kg

2^{nd} stage propellant mass = 70779.97 kg

Payload mass = 5000 kg

Payload Mass: -

$$\pi_{PL} = M_{PL} / M_{tot}$$

$$M_{PL} = M_{tot} * \pi_{PL}$$

$$M_{PL} = 5{,}00{,}000 * 0.01$$

$$\boxed{M_{PL} = 5{,}000 \text{ kg}}$$

Effective exhaust velocity:

$$V_e = ISP*g = 455 * 9.81 = 4463.55 \text{ m/sec}$$

Dry Mass:

For rockets to enter LEO orbit the ΔV should be 9.5 km/sec

Then,

$$\Delta V = V_e * \ln (M_{tot} / M_{dry})$$

$$9500 = 4463.55 * \ln (5{,}00{,}000 / M_{dry})$$

$$\boxed{M_{dry} = 59{,}874 \text{ kg}}$$

1st stage: -

Engine: - J-2T-250K x 2

Thrust = 2223.20 kN (at vacuum)

ISP = 441 sec

Area ratio = 173: 1

Thrust to weight ratio = 106.5: 1

Engine Mass = 20,875.1kg

2nd stage: -

Engine: - J-2T-250K x 1

Thrust = 1111.60 kN (at vacuum)

ISP = 441 sec

Area ratio = 173: 1

Thrust to weight ratio = 106.5: 1

Engine Mass = 10,437.5kg

The total mass consists of dry mass and propellant mass. The dry mass consists of empty mass and payload mass. The empty mass consists of the structure, propulsion systems and other systems. The structure consists of the rocket consisting of a nose cone, body, fins and shielding. The propulsion system consists of engines, tanks and pressurizing systems. Other systems consist of avionics, separation systems and cooling systems.

Table 7.1: Rocket Weight Profile

<table>
<tr><td rowspan="12">Total mass = 5,00,000 kg</td><td rowspan="11">Dry Mass = 59,874 kg</td><td rowspan="10">Empty Mass = 54,874 kg</td><td rowspan="4">Structure</td><td>Nose Cone = 200 kg</td></tr>
<tr><td>Body = 3000 kg</td></tr>
<tr><td>Fins = 90 kg</td></tr>
<tr><td>Shielding = 2210 kg</td></tr>
<tr><td rowspan="3">Propulsion Systems</td><td>Engines = 31,312.2348kg</td></tr>
<tr><td>Tank = 5000kg</td></tr>
<tr><td>Pressurizing Systems = 6,200kg</td></tr>
<tr><td rowspan="3">Other Systems</td><td>Avionics = 2550 kg</td></tr>
<tr><td>Separation System = 1520 kg</td></tr>
<tr><td>Coolant Systems = 2791.7652kg</td></tr>
<tr><td colspan="3">Payload = 5,000kg</td></tr>
<tr><td colspan="4">Propellant = 4,35,125.3 kg</td></tr>
</table>

CHAPTER 08

STRUCTURAL SELECTION

Table 8.1: Rocket Structural Design Considerations

Rocket	**Nose Cone**	**Body Shape**	**Fins**
New Glenn	Tangent Ogive	Cylinder	Trapezoidal
Saturn V	Biconic	Cylinder	Clipped Delta
Falcon 9	Elliptical	Boat tail	Grid Fins
Delta IV Heavy	-	Cylinder	-
SS-520	Conical	Cylinder	Canted/Raked

8.1 Nose Cone

Conic, spherically blunted conic, bi-conic, tangent ogive, spherically blunted tangent ogive, secant ogive, elliptic, parabolic, power series and Haack series are the cone designs generally used for aerospace applications. The Von-Karman nose cone is a special case of the Haack series.

8.1.1 Conic

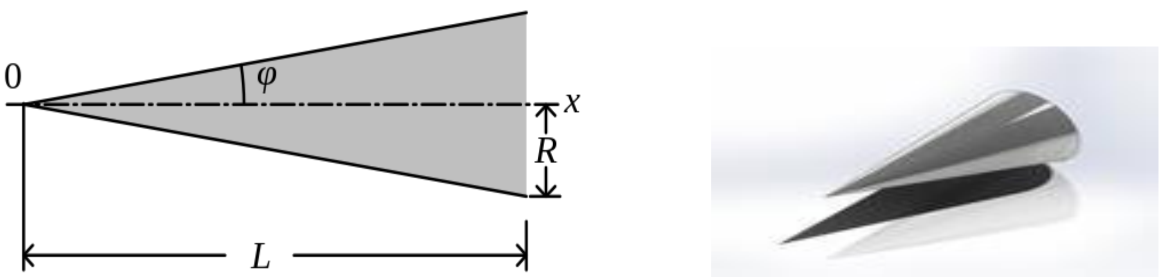


Figure.8.1: Conic shape

A very common nose-cone shape is a simple cone. This shape is often chosen for its ease of manufacture. Conic nose shapes are rocket noses that come to a point. These rocket noses call back to the first days of modern rocketry, and are often found in model rocket replicas of old rockets such as the Saturn V. This rocket nose design is easy to construct, but not very aerodynamic in comparison to more rounded designs.

8.1.2 Spherically blunted conic

This rocket nose shape is a cone shape in basic design but features a spherical endpoint. In practical applications, a conical nose is often blunted by capping it with a segment of a sphere.

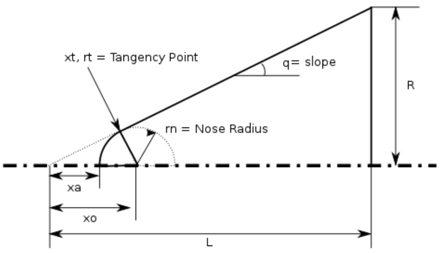


Figure.8.2: Spherically blunted conic shape

8.1.3 Bi-conic

Bi-conic nose shapes feature a two-cone design with a secondary cone stacked at the frustum of the first cone shape. This results in a staggered but planed tapering effect. Conical transition system.

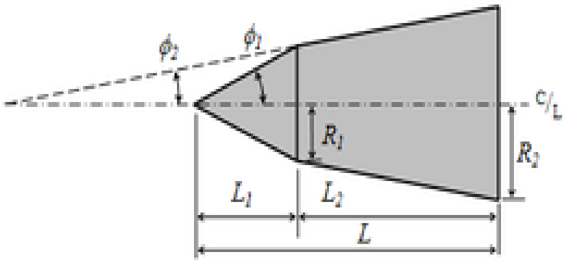
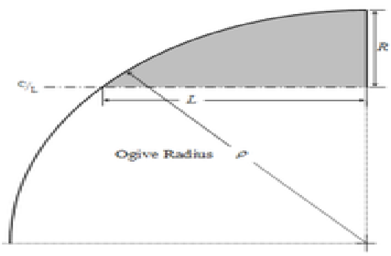


Figure.8.3: Bi-conic shape

8.1.4 Tangent - ogive

The tangent ogive shape is the most familiar in hobby rocketry. The profile of this shape is formed by a segment of a circle such that the rocket body is tangent to the curve of the nose cone at its base, and the base is on the radius of the circle. The popularity of this shape is largely due to the ease of constructing its profile, as it is simply a circular section.



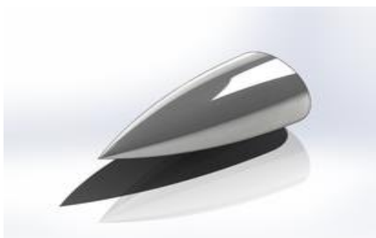


Figure.8.4: Tangent-ogive shape

8.1.5 Spherically blunted tangent ogive

A tangent ogive nose is often blunted by capping it with a segment of a sphere

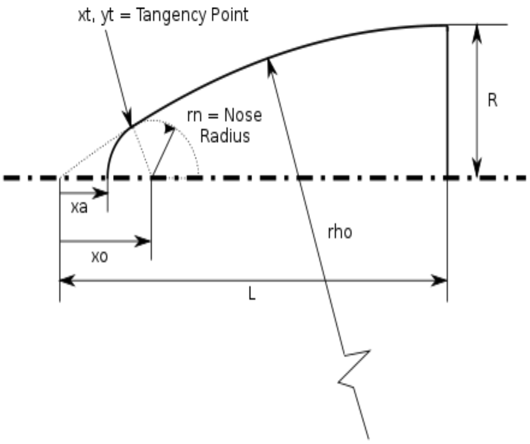


Figure.8.5: Spherically blunted tangent ogive shape

8.1.6 Secant ogive

The profile of this shape is also formed by a segment of a circle, but the base of the shape is not on the radius of the circle defined by the ogive radius. The rocket body will not be tangent to the curve of the nose at its base.

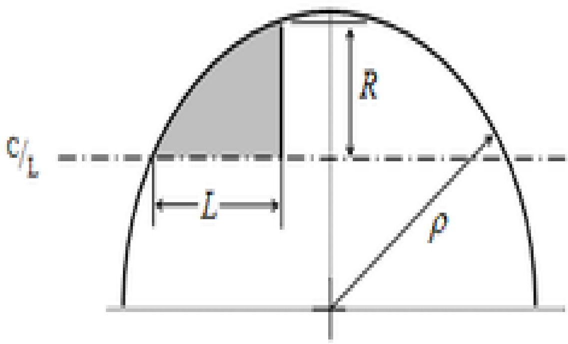


Figure.8.6: Secant ogive

8.1.7 Elliptical

The profile of this shape is one-half of an ellipse, with the major axis being the centerline and the minor axis being the base of the nose cone. A rotation of a full ellipse about its major axis is called a prolate spheroid, so an elliptical nose shape would properly be known as a prolate hemispheroid. This shape is popular in subsonic flight (such as model rocketry) due to the blunt nose and tangent base. This is not a shape normally found in professional rocketry, which almost always flies at much higher velocities where other designs are more suitable. If R equals L, this is a hemisphere.

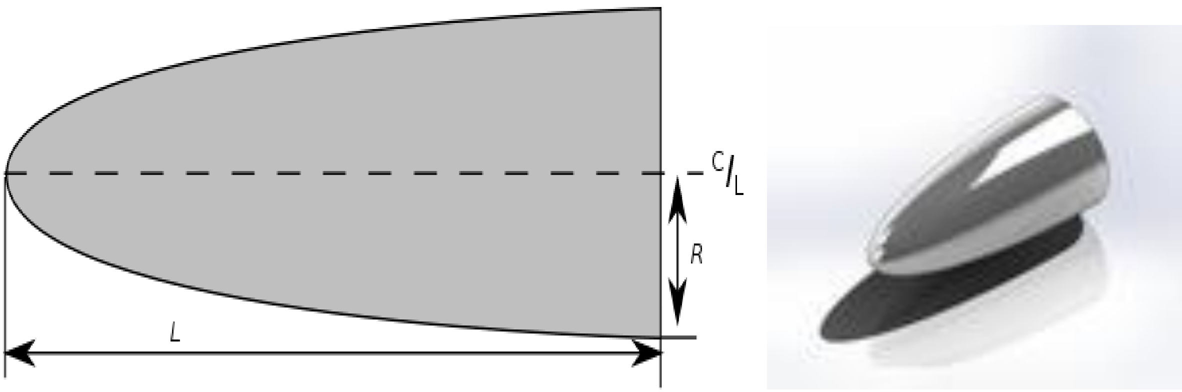


Figure.8.7: Elliptical shape

8.1.8 Parabolic

Parabolic nose cone designs are a rounded nose cone design like elliptical nose cones, but aren't necessarily at a flat angle at the point the nose cone meets the rocket body tube. Parabolic and elliptical shaped nose cones have been adopted in many commercial aircraft due to the reduced drag and corresponding reduced fuel consumption.

Figure.8.8: Parabolic shape

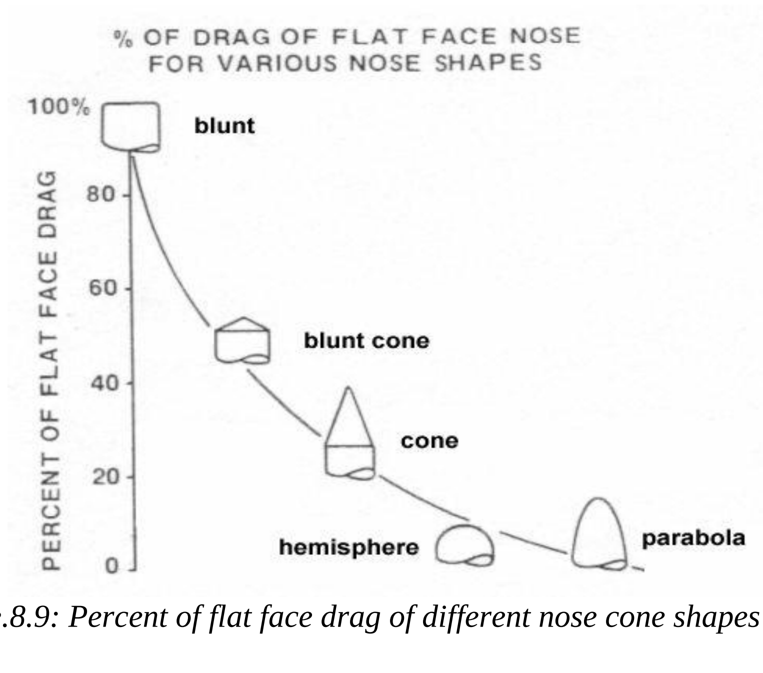


Figure.8.9: Percent of flat face drag of different nose cone shapes

8.2 FINS

The primary purpose of fins on a rocket is to serve as the rocket's control system. Fins give directional stability and help the rocket fly straight. Model rocket fins may be made of plastic, balsa wood or stiff cardboard. Fins should be attached in a symmetrical form of three, four or possibly more. Model rocket fins are usually fixed; while some actual rockets have fins that have movable components. Movable components allow for the in-flight control of the rocket's guidance.

The four most common shapes of fins are rectangular, elliptical, straight-tapered and swept-tapered (visual/overhead Common Fin Shapes). The four parts of a fin are leading edge, trailing edge, root edge and tip (Parts of a Fin). The shape of a fin is one factor that determines the amount of drag produced. Fin characteristics such as the total surface area, total span and sweep angle all help to determine the amount of drag produced by a rocket's fins. When viewing the fin from the fin's tip, the sectional shape is a determiner of the amount of drag produced by a rocket's fin.

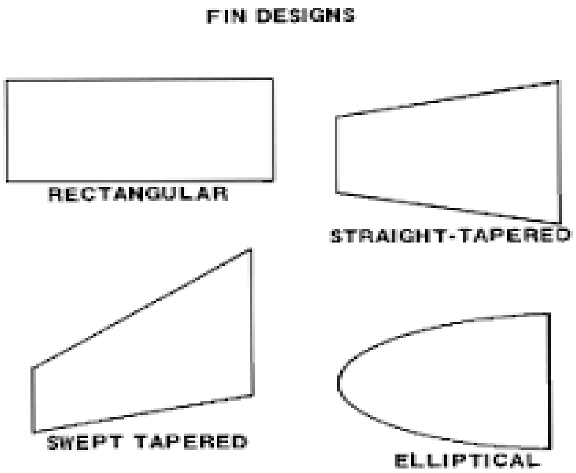


Figure.8.10: Types of fin designs

8.2.1 Grid Fins

Grid fins are a lattice of smaller aerodynamic surfaces arranged within a box. Grid fins can be folded, pitched forward (or backwards), against the cylindrical body of a missile more directly and compactly than planar fins, allowing for more compact storage of the weapon; this is of importance where weapons are launched from a tube or for craft which store weapons in internal bays, such as stealth aircraft. Generally, the grid fins pitch forward/backward away from the body shortly after the missile has cleared the firing craft.

Grid fins have a much shorter chord (the distance between leading and trailing edge of the surface) than planar fins, as they are effectively a group of short fins mounted parallel to one another. Their reduced chord reduces the amount of torque exerted on the steering mechanism by high-speed airflow, allowing for the use of smaller fin actuators, and a smaller tail assembly overall.

Grid fins perform very well at subsonic and supersonic speeds, but poorly at transonic speeds; the flow causes a normal shock wave to form within the lattice, causing much of the airflow to pass completely around the fin instead of through it and generating significant wave drag.

At high Mach numbers, grid fins flow fully supersonic and can provide lower drag and greater maneuverability than planar fins.

Figure.8.11: Grid fins

Selected Features:

1. **Nose Cone:** Ogive of 200kg with 12m length and 6m diameter and 0.05m thickness

Selection criteria:

- ❖ Ogive nose cone has Slightly greater volume for a given base and length, a blunter nose providing structural superiority and slightly lower drag
- ❖ The ogive nose cone has greater volume so the payload has larger space to fit into it.

2. **Body Shape:** Cylinder of 3000 kg with 26.5m height and 6m diameter and 0.01m thickness

Selection criteria:

- ❖ Pressure vessels are round, as they provide maximum strength from internal pressure. So, a cylindrical shape ensures less weight of the rocket's walls. (Same justification for oil tankers to be cylindrical in shape).
- ❖ Outer casing of the cylinder brings in line with internal structures like fuel tanks, tubes, ducts, wiring etc. This ensures area optimization.

3. **Fins:** Grid fins of 190 kg with 4x2m

CHAPTER 09
ENGINE SELECTION

9.1 Introduction

The engine in a rocket plays a major role as it generates thrust for the rocket to take-off and move forward. A rocket engine uses stored rocket propellants as the reaction mass for forming a high-speed propulsive jet of fluid, usually high-temperature gas. Rocket engines are reaction engines, producing thrust by ejecting mass rearward, in accordance with Newton's third law. Most rocket engines use the combustion of reactive chemicals to supply the necessary energy, but non-combusting forms such as cold gas thrusters and nuclear thermal rockets also exist. The rocket engines are classified as follows:

- **Thermal Rocket engines:** A thermal rocket is a rocket engine that uses a propellant that is externally heated before being passed through a nozzle to produce thrust, as opposed to being internally heated by a redox (combustion) reaction as in a chemical rocket.
- **Chemical Rocket Engines:** Chemical rocket engines use the combustion of propellants to produce exhaust gasses as the working fluid. The high pressures and temperatures of combustion are used to accelerate the exhaust gasses through a rocket nozzle to produce thrust.
    - Liquid Rocket Engine
    - Solid Rocket Engine
    - Hybrid Rocket Engines
    - Monopropellant Rocket Engines

The type of engine taken into consideration for this rocket is the aerospike engine in liquid rocket engines. The aerospike engine drops the outer bell of the nozzle. Instead, it directs the outflow from the combustion chamber towards a ‘plug’ in the middle and lets the ambient gasses surrounding the exhaust regulate its expansion. Thus, when the ambient pressure decreases, the exhaust shape is altered accordingly. In turn, this altitude compensation maximizes the thrust achievable at a particular altitude.

There are 2 types of aerospike engines:

- **Linear aerospike engine:** The linear aerospike features a series of small combustion chambers along the unwrapped bell, also called the ramp, that shoot hot gasses along the ramp's outside surface to produce thrust along the length of the ramp, hence the name ‘linear aerospike’.
- **Toroidal aerospike engine:** In toroidal aerospike engine the combustion chamber is like a donut and the throat is an opening pointing inward towards a spike.

9.2 Considerations of 1st and 2nd stage engine

Table 9.1: 1st and 2nd stage engine considerations

S.No.	**Name of the engine**	**Type of engine**	**Total thrust (kN)**	**Burn time (sec)**	**Isp(sec)**	**Diameter (m)**	**Height (m)**
1	RS-2200	Linear aerospike engine	2201	298	455	6.40	4.32
2	J-2T-250K	Toroidal aerospike engine	1111.60	348	441	2.97	1.25
3	XRS-2200	Linear aerospike engine	1192	301	439	3.38	2.01

9.2.1 RS-2200

The Rocketdyne RS-2200 was an experimental linear aerospike rocket engine developed by Rocketdyne for Lockheed Martin's VentureStar program. The program was ultimately canceled in 2001 before any RS-2200 engines were assembled.

Rocketdyne LOX/LH2 rocket engine. Development canceled 1999. Linear Aerospike Engine developed for use on the Lockheed Reusable Launch Vehicle, the production follow-on to the X-33. The RS-2200 Linear Aerospike Engine is being developed for use on the Lockheed Martin Skunk Works' Reusable Launch Vehicle, the production follow-on to the X-33. The Aerospike allows the smallest, lowest cost RLV to be developed because the engine fills the base, reducing base drag, and is integral to the vehicle, reducing installed weight when compared to a bell-shaped engine. The. Aerospike is the same as bell shaped rocket engines except that its nozzle is open to the atmosphere. The open plume compensates for decreasing atmospheric pressure as the vehicle ascends, keeping the engine's performance very high along the entire trajectory.

Figure.9.1: RS-2200

This altitude compensating feature allows a simple, low-risk gas generator cycle to be. used. Over $500 million were invested in Aerospike engines up to the contract award date of the X-33, and full-size linear engines have accumulated 73 tests and over 4,000 seconds of operation. Throttling, Percent Thrust: 20-109. Dimensions: Forward End: 6.4 m wide X 2.36 m long. Aft End: 2.36 m wide X 2.36 m long. Length: 4.32 m. Designed for booster applications. Gas generator, pump-fed.

9.2.2 J-2T-250K

Figure.9.2: J-2T-250K (2)

The culmination of the toroidal aerospike engine development has been done by Rocketdyne in the late 1960s with the J-2T-250k. This 1.1MN aerospike is touching the geometrical and manufacturing limits with an annular combustion chamber.

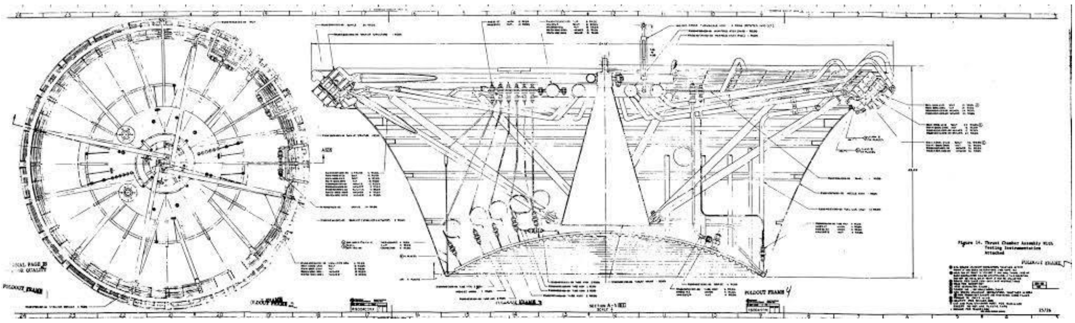


Figure.9.3: Schematic of J-2T-250K

This engine has partial internal expansion in order to reduce the turning angle, a moderate expansion area ratio of 74.1, an outer diameter of 2.5m and a throat gap of 7mm.

9.2.3 XRS-2200

The Rocketdyne XRS-2200 was an experimental linear aerospike engine developed in the mid-1990s for the Lockheed Martin X-33 program. The design was based on the J-2S linear aerospike engine developed in the 1960s and therefore used the J-2's combustion cycle and propellant choice. Rocketdyne intended to develop the subscale XRS-2200 into the RS-2200 for use on the VentureStar. While the X-33 program was canceled, two XRS-2200 engines were produced and tested.

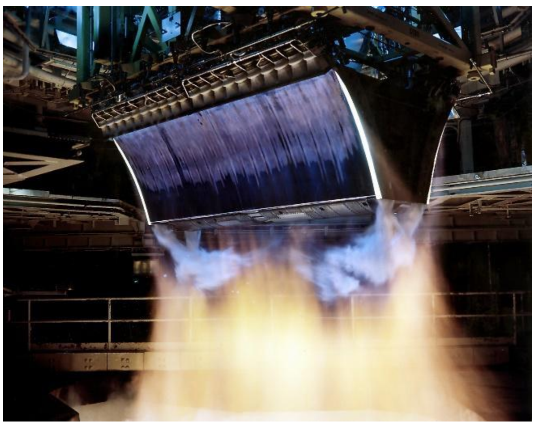


Figure.9.4: XRS-2200

Rocketdyne LOX/LH2 rocket engine. Development ended 1999. Linear aerospike engine for X-33 SSTO technology demonstrator. Based on the J-2S engine developed for improved Saturn launch vehicles in the 1960's.

Gas generator cycle; throttling 40% to 119% of nominal thrust; differential thrust between two engines plus-minus 15%. X-33 Advanced Technology Demonstrator Development. Designed for booster applications. Gas generator, pump-fed.

Rocketdyne intended to develop the subscale XRS-2200 into the RS-2200 for use on the VentureStar. While the X-33 program was canceled, two XRS-2200 engines were produced and tested.

Selected Engine: J-2T-250K

- Rs-2200 and XRS-2200 engines were not used in application, they were only tested for some Saturn V variants.
- J-2T-250K engine was selected because the size of toroidal aerospike engine was ideal for the rocket design and it was found to be more efficient.
- It has better cooling efficiency.

CHAPTER 10
MATERIAL SELECTION

10.1 Introduction

An important factor that must be considered when designing a structure for space is the type of materials the structure will utilize. Choosing the proper materials will provide the optimal operational environment for the structure. The materials selected must meet three main criteria. These criteria are the number of launches required, the structural support provided, and the ability to survive the space environment. The material selection criteria relate to the reliability of the solar power satellite. The reliability is an important measure because of the lack of accessibility for maintaining the structure. Consistent operation during the 50-year duration of the mission is essential for the satellite to be a feasible power source.

The materials used in rockets should have

- High Specific Strength.
- High Specific Modulus.
- Fabrication Easiness.
- Easy availability.
- Critical requirements.
- Service Conditions.

10.2 Materials used

10.2.1 Structural Metallic Materials

i) 15 CDV6: Low carbon steel used in solid rocket motor case.

15CDV6 is a low carbon, chromium, molybdenum, vanadium heat-treatable steel with high strength after heat treatment (1080-1280 N/mm^2).

The alloy is easily welded and does not require localized heat treatment after welding. 15CDV6 combines outstanding yield strength with good toughness. In motorsport applications, the product offers a cost-effective solution in areas where a combination of high-strength and excellent weldability is required.

ii) M250: Maraging steel with high strength and high toughness used in booster solid rocket motor case.

Maraging steels are steels that are known for possessing superior strength and toughness without losing ductility. Aging refers to the extended heat-treatment process. These steels are a special class of very-low-carbon ultra-high-strength steels that derive their strength not from carbon, but from precipitation of intermetallic compounds. The principal alloying element is 15 to 25 wt.% nickel. Secondary alloying elements, which include cobalt, molybdenum and titanium, are added to produce intermetallic precipitates. Original development was carried out on 20 and 25 wt.% Ni steels to which small additions of aluminum, titanium, and niobium were made; a rise in the price of cobalt in the late 1970s led to the development of cobalt-free maraging steels.

iii) Titanium alloy: (Ti-6Al-4V) used in high pressure gas bottles.
Titanium alloys were first considered on account of their very favorable specific strengths and therefore were found to be suitable for the aerospace industries. Ti-6Al-4V, which continues to be the workhorse of the titanium industry, has been used in airframes, landing gears, empennage, and wings and even in gas turbine engines. Only at temperatures below 300°C do carbon fiber reinforced plastics have a higher specific strength than titanium alloys, with the latter surpassing all other commercial materials at all other application temperatures in terms of specific strength. However, the other important advantages of titanium alloys have also been realized and are being exploited in a number of non-aerospace applications. These properties are excellent corrosion resistance, good creep resistance, good high temperature strength, low thermal coefficient of expansion, biocompatibility and relative abundance of the metal in the earth's crust. Counterbalancing these obvious advantages of titanium alloys is the high component cost arising out of the high cost of metal extraction and high cost of shaping and forming. Even in the aerospace industries, which has been largely performance driven, the cost of production is expected to have a prominent role in materials selection in the future.

iv) Aluminum Alloys: used in liquid propellant tanks, engine components, airframe in Reusable Launch Vehicles.
Types: AA 2219, AA2014, AA6061.
v) Magnesium/ Mg-Lithium alloys: used in upper stage structure like payload adopter, avionic decks, equipment bay structure.
vi) Powder metallurgy products: used in nozzle throat for bi-propellant control thrusters.

10.2.2. Composite Materials

i) Carbon F / Kevlar F – Epoxy Resin: used in solid motor case, pressure vessel, interstage, payload adopter.

ii) Carbon C / Silica C – Phenolic Resin: used in ablative liners, nozzle throat inserts.

SiC–SiC matrix composite is a particular type of ceramic matrix composite (CMC) which have been accumulating interest mainly as high temperature materials for use in applications such as gas turbines, as an alternative to metallic alloys. CMCs are generally a system of materials that are made up of ceramic fibers or particles that lie in a ceramic matrix phase. In this case, a SiC/SiC composite is made by having a SiC (silicon carbide) matrix phase and a fiber phase incorporated together by different processing methods. Outstanding properties of SiC/SiC composites include high thermal, mechanical, and chemical stability while also providing high strength to weight ratio

10.2.3. Thermo-structural materials

i)Carbon Fiber/ Silicon Carbide Ceramic Matrix

Used in the nose-cap of the heat-shield, leading edge and control surface of RLV.

ii) SiC F/SiC Ceramic Matrix Composite: used in high temperature / hot structures.

The silica fibers are mixed with water and chemicals, and the mixture is poured into molds, which are zapped in microwave ovens at 2,350 degrees to fuse the silica fibers.

Tiles are too brittle to attach to the orbiter directly. The shuttle's skin contracts slightly while in orbit, then expands during reentry. In addition, the stresses of launch and reentry cause the skin to flex and bend. Such motions could easily crack the tiles or shake them off. To keep them in place, workers glue the tiles to flexible felt-like pads, then glue the pads to the orbiter. The primary tiles used are given one of two coatings. The tiles exposed to reentry temperatures of up to 2,300 degrees Fahrenheit, such as those on portions of the belly, are given a protective coating of black glass. Black tiles work by reflecting about 90 percent of the heat they're exposed to back into the atmosphere, while the tiles' interior absorbs the rest. The tiles' interiors radiate absorbed heat so slowly that after landing, the tiles take hours to cool

4.Thermal Protection Materials

i) Carbon/Carbon Composite: used in nozzle throat, nosecone (SiC Coating).

ii) Silica tiles: (Silica Fiber in Si gel) used in space shuttle and SRE (capsule) missions.

iii) Polyurethane foam insulation for cryogenic propellant tanks and subsystems.

Polyurethane foam insulation is a modern and highly effective method of the thermal insulation of buildings. When applied by spraying, it reaches even the smallest of gaps. It provides the permanent and damage-resistant insulation of a house, improving living comfort, and reducing heating bills. The strong point of polyurethane foam is its quick and easy application (of course, by a specialized team). PUR foam is applied using the spray method — it increases its volume several-dozen times and cures at a rapid pace. Spray insulation, therefore, takes much-less time than insulation with other insulation materials.

The advantage of polyurethane foam is its easy application in the loft. The material adapts perfectly to oblique and difficult surfaces. Thanks to its expansion, it penetrates into the tiniest of gaps, preventing the formation of thermal bridges. For slanted ceilings and cramped spaces, it is best to choose polyurethane foam.

iv) Flexible Reusable Surface Insulation-FRSI Blanket / felt with Teflon coating used in the leeward surface of RLV.

10.2.5. Special Materials

i) Soft magnetic alloy: Fe-Co-V: used in torque motor in spacecraft.

The Fe-Co-V ternary system in particular is known to contain a permanent magnet called Vic alloy with nominal composition of Fe40Co52V8. Fe–Co–V alloy with large grains displays lower coercivity and higher permeability than the commercial Fe–Co–V alloy. Even though its yield strength at 600 °C is lower than the commercial Fe–Co–V, the creep strains of the Fe–Co–V alloy with large grains are only 1/10–1/2 of that for the commercial alloy in the initial and middle periods of the creep test performed at 600 °C under 150 MPa

ii) HED permanent magnet: Sm-Co: used in gyroscope, inertial navigation, guidance & control system, reaction momentum wheel in
spacecraft.

iii) Bimetallics: (SS-Al and SS-Ti) used in Cryosystem for assembly of dissimilar items.

iv) Opto-electronics: used in photo sensors and photodiodes and photovoltaic devices for power generation in satellites.

v) Electro-optics: used in High resolution imaging in Spacecrafts.

Aluminum-lithium alloys, now introduced on a large scale in the Space Shuttle External Tank, represent a significant improvement in strength-to-weight ratio over conventional aluminum alloys. Composite propellant tanks can offer further gains in mass efficiency with judicious design, but the need for robust joints and minimization of permeation after fatigue remain significant roadblocks to the use of composites in pressurized structure. However, filament-wound composite solid rocket motor cases are a mature and widespread technology.

Table 10.1: Materials used in different components

S.NO	MATERIAL USED	COMPONENTS
1	15CDV6: Low carbon steel	Rocket motor case
2	Titanium alloy: (Ti-6Al-4V)	High pressure gas cylinders
3	Aluminum Alloy	Engine components,
4	Magnesium/ Mg-Lithium alloys	Payload adopter, avionic decks, equipment bay structure
5	Carbon F / Kevlar F	Pressure vessel, interstages
6	Carbon C / Silica C	Ablative liners, nozzle throat inserts
7	Carbon Fiber/ Silicon Carbide Ceramic Matrix	Nose-cap of the heat-shield, leading edge and control surface of RLV
8	SiC	Nozzle throat, nosecone
9	FRSI Blanket / felt with Teflon coating	Leeward surface of RLV
10	Polyurethane foam insulation	Propellant tanks
11	Fe-Co-V	Torque motor in spacecraft
12	: Sm-Co	Gyroscope, inertial navigation, guidance, control system, reaction momentum wheel
13	SS-Al and SS-Ti	Assembly of dissimilar items
14	Opto-electronics	Photo sensors and photodiodes and photovoltaic devices
15	Electro-optics	High resolution imaging

CHAPTER 11
ELECTRONICS

11.1 Electronics Bay

The electronics bay (ebay) is the central housing unit for all electronic and power systems for the High-powered model rockets. The main electronics in the ebay are the altimeter, power source and gyro that record data and trigger events during the flight.

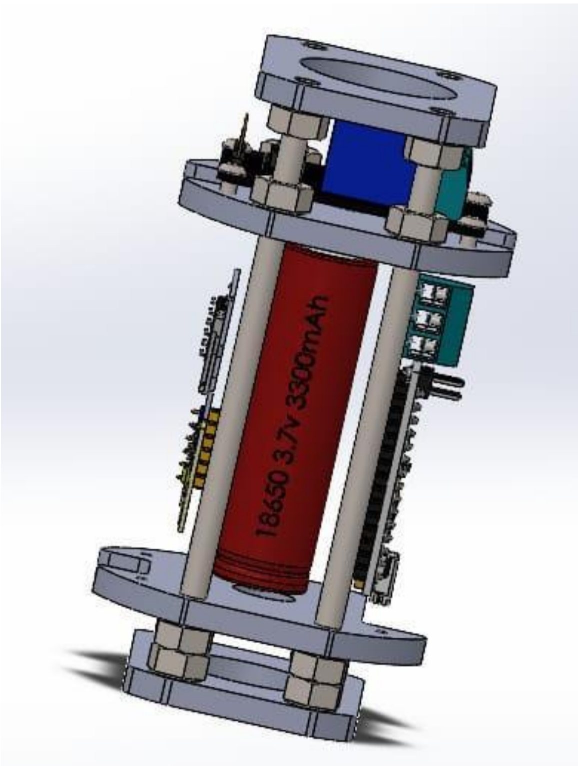


Figure.11.1: Avionics in a rocket

11.2 Avionics

A literal blend of the terms "aviation" and "electronics," the avionics installed in an aircraft or spacecraft can include engine controls, flight control systems, navigation, communications, flight recorders, lighting systems, threat detection, fuel systems, electro-optic (EO/IR) systems, weather radar, performance monitors, and systems that carry out hundreds of other mission and flight management tasks.

Every modern aircraft, spacecraft, and artificial satellite uses electronic systems of varying types to perform a range of functions pertinent to their purpose and mission. Generally, the more complex the craft or mission is, the more complicated the electronic systems are that they employ.

—Include rugged flight computers, GPS systems for location monitoring, inertial measurement units (IMUs) for reporting the vehicle's specific force, angular rate, position location and maneuvering, and proprietary control units for propulsion, separation, valve, payload interfacing, and pressurization, and a transponder (C-band) for range safety tracking.
(Linux and dual-core x86 processors).

11.3 Navigation & Guidance Systems

The guidance system of a rocket includes very sophisticated sensors, radars, and communication equipment. The guidance system has two main roles during the launch of a rocket; to provide stability for the rocket, and to control the rocket during maneuvers.

Navigation/guidance systems use modified versions of algorithms. The guidance system adjusts the target trajectory accordingly as capsules separate from one other during first stage burn. (GPS)

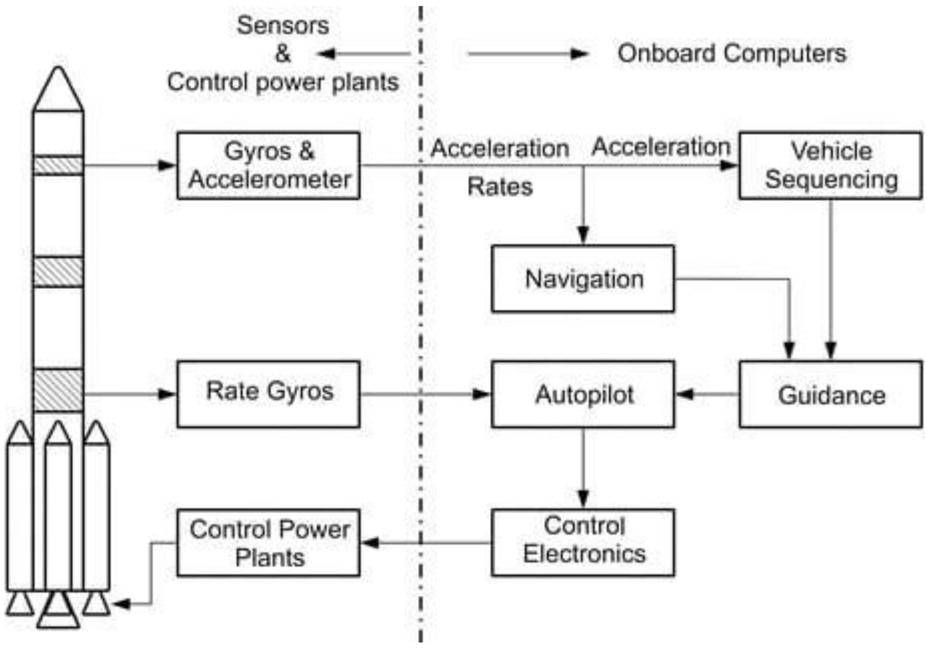


Figure.11.2: Schematic of navigation and guidance systems in a rocket

11.4 Flight Termination Systems (FTS)

A Flight Termination System (FTS) consists of the components onboard a vehicle that provide the ability to end that vehicle's flight in a controlled manner.

1. An FTS may comprise different types of equipment or procedure, including: a self-destruct system for remote initiation, a separation system, a parachute release system or any other systems or components onboard the launched vehicle that may be used to terminate flight.
2. In order to increase flight safety, these kinds of systems must be completely independent. For this reason, an FTS must have its own dedicated communications datalink. In case of autopilot or communications error, the FTS will therefore still be capable of terminating the flight.

—The rocket is equipped with a standard FTS comprising dual redundant receiver/encoder strings, DC batteries, Electro-mechanical Safe & Arm Devices (ESADs), and Spacecraft Ordinance Systems (which provide initiation protocols and prevent inadvertent firing) with electrostatic discharge immunity.

11.5 Electrical Power Supply

The primary electrical power for operating the electronic equipment is obtained from solar cells. Individual cells can generate only small amounts of power, and therefore, arrays of cells in series-parallel connection are required. The power supply most readily available is the battery, which converts chemical to electrical energy.

Table 11.1: Performance of different types of cells

Type of cell	Limiting theoretical performance (watt-hours per pound)	Currently available performance Watt-hours per pound	Currently available performance Watt-hours per cubic inch
Lead-acid	75	15	1.0
Nickel-cadmium	92	15	1.0
Zinc-silver	176	40	3.0
Hydrogen-oxygen	1,700	300	4.6

— The electrical power supply in the launch can vary depending on requirements by a payload provider who is also responsible for supplying the necessary cabling to interface ground support equipment with payload processing room power. Typical power values in this section are 110V AC and 208V AC

11.6 Radio Frequency Systems

The communication system is an essential part of a spacecraft, enabling spacecraft to transmit data and telemetry to Earth. The three functions of a communications system are receiving commands from Earth (uplink), transmitting data down to Earth (downlink) and transmitting or receiving information from another satellite (crosslink or inter-satellite link). There are two types of communication systems: radio frequency (RF) and free space optical (FSO) also referred to as laser communications (lasercom). Most spacecraft communications systems are radio frequency based. They are conducted in the Institute of Electrical and Electronics Engineers (IEEE) radio bands of 300 MHz to 40 GHz.

The Stage 1 launch vehicle transmits RF signals in the S-band at a 2221.5 MHz using PCM/FM modulation with data rates at 1.8 Mbps and power output of 10W. Stage 2 uses the same configuration as the Stage 1 but uses a frequency of 2231.5 power output of 20W. (derived from falcon 9)

11.7 Radar Transponder

In a communications satellite, the equipment which provides the connecting link between the satellite's transmit and receive antennas is referred to as the transponder. The transponder forms one of the main sections of the payload, the other being the antenna subsystems.

Two C-band transponders with receivers operating at frequencies of 5690 and 5690 MHz, pulse modulation with data rates at 2000 and 3000 pps and maximum power output of 400W.

CHAPTER 12
3-VIEW DIAGRAM OF ROCKET

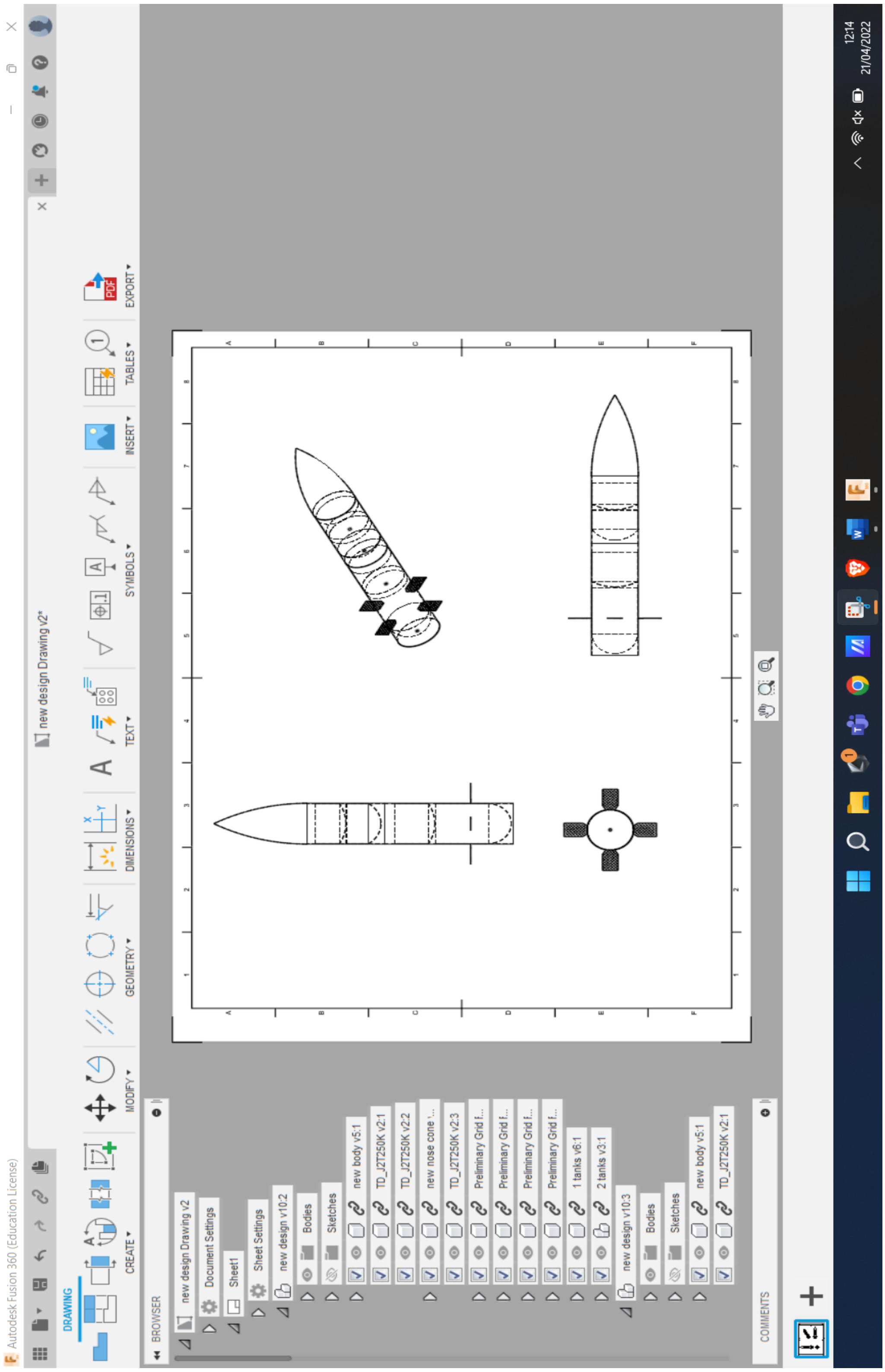


Figure.12.1: 3-View diagram of rocket design

CHAPTER 13

3-D DIAGRAM OF ROCKET

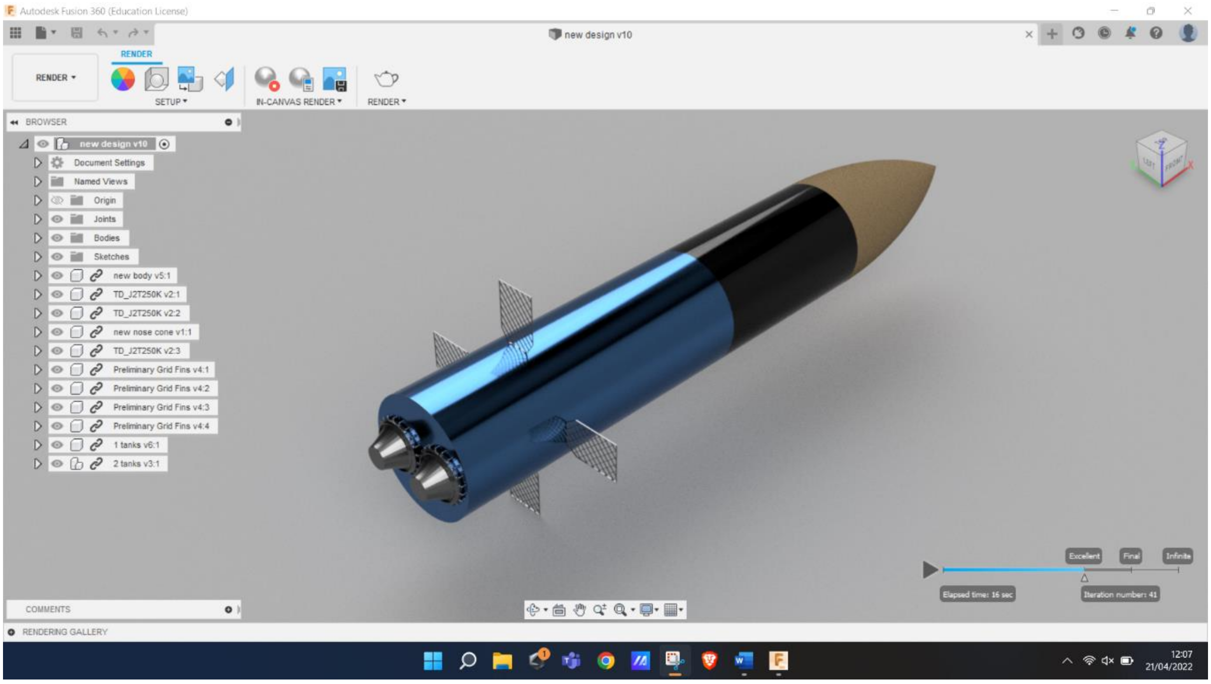


Fig.13.1: Isometric view

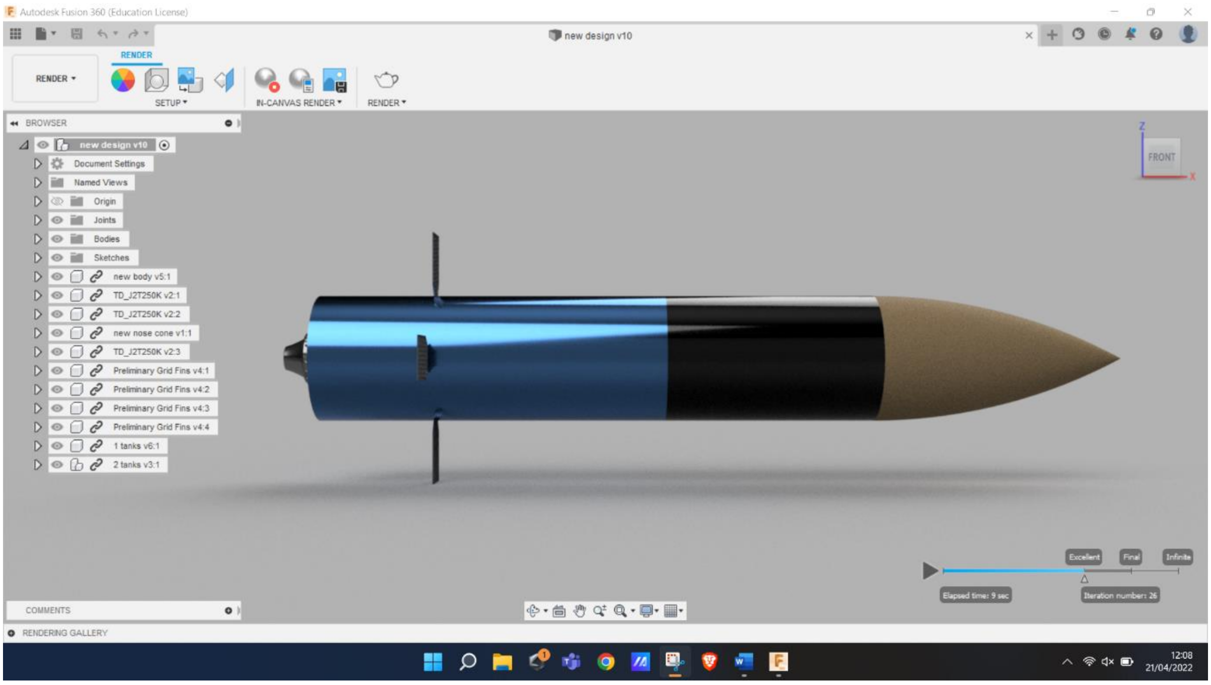


Fig.13.2: Side view

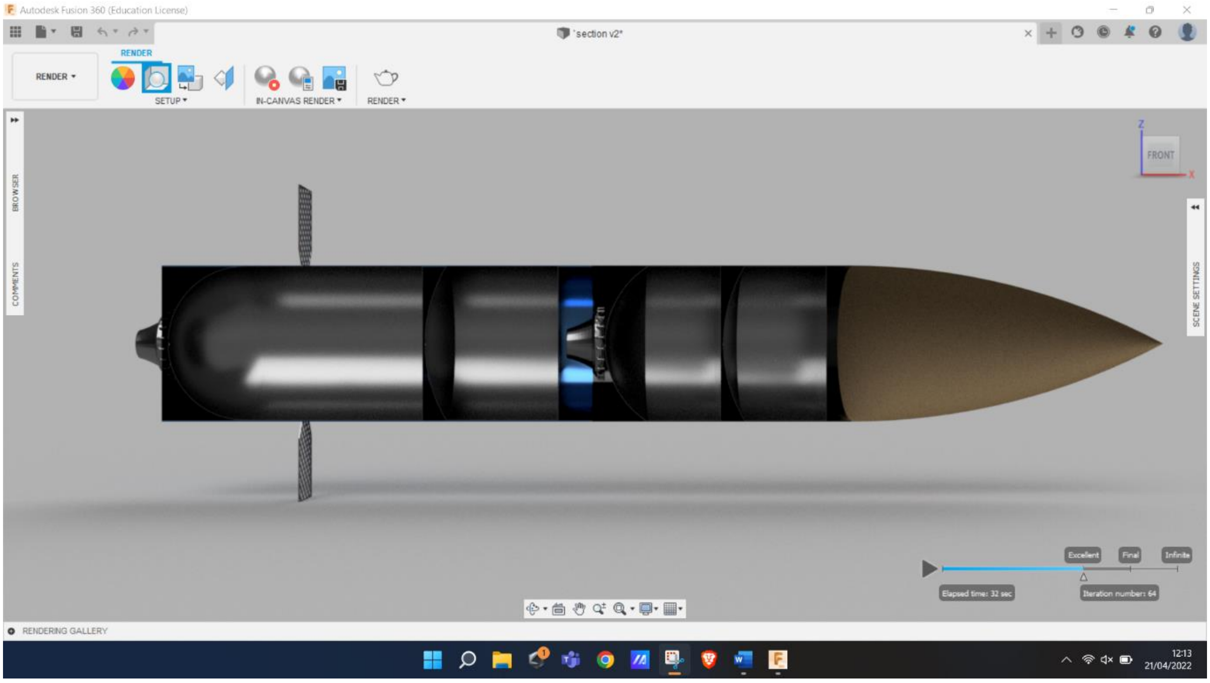


Fig.13.3: Cross section view

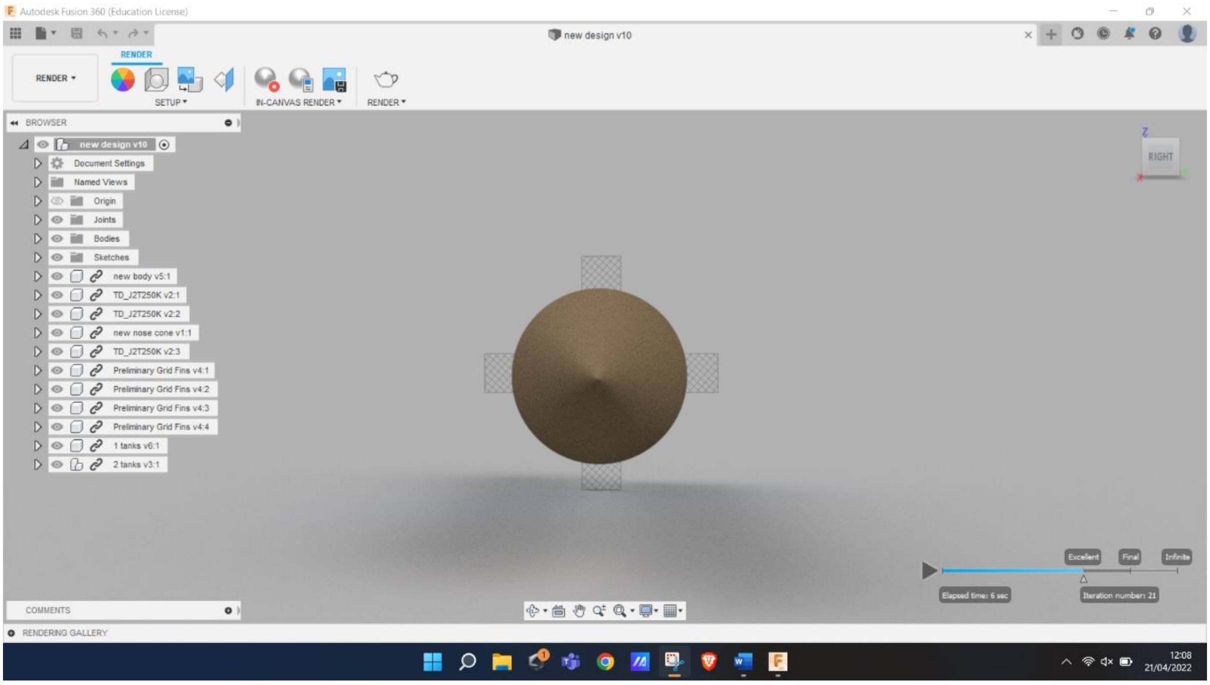


Fig.13.4: Top view

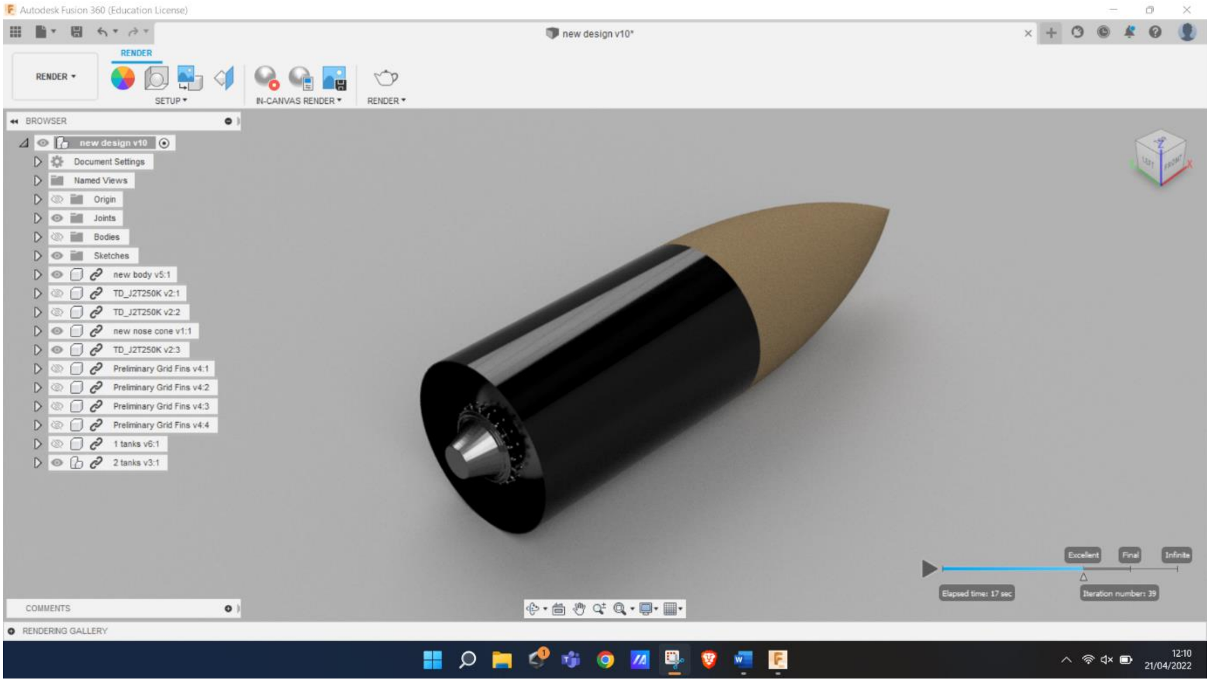


Fig.13.5: Isometric view of 2nd stage

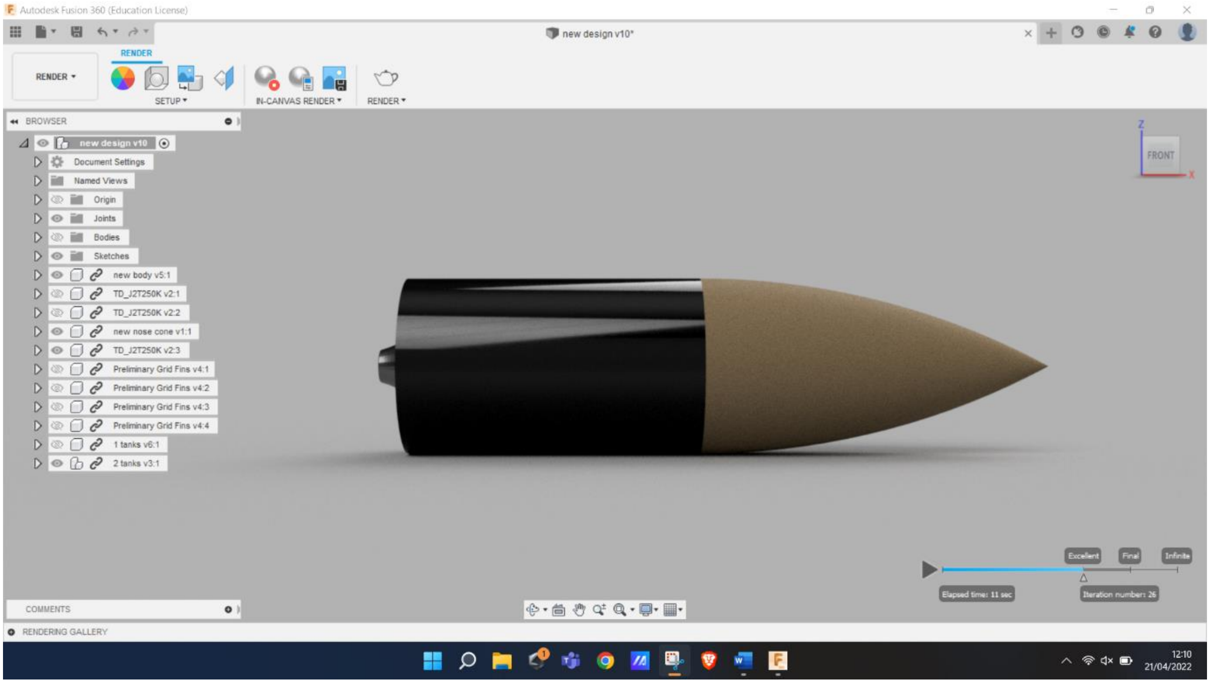


Fig.13.6: Side view of 2nd stage

CONCLUSION

The rocket model was successfully implemented based on the specified parameters mentioned in the selection of tentative parameters from the conceptual study of fifteen rockets. The rocket should carry a payload of 5000 kg. Toroidal aerospike engine was used for this design. This engine has partial internal expansion in order to reduce the turning angle, a moderate expansion area ratio of 74.1, an outer diameter of 2.5m and a throat gap of 7mm. Aerospike nozzle has 90% overall better performance than the conventional bell-shaped nozzle. The tentative design parameters were found to be as: height = 38.5m, diameter = 7m. The total mass of the vehicle was found to be 5,00,000kg and the empty mass of 54,874kg. The apogee of the rocket was found to be 233.4km. Ogive nose cone of 100kg with 12m length and 7m diameter and 0.05m thickness is taken and a cylindrical body of 2233 kg with 26.5m height and 7m diameter and 0.01m thickness and grid fins of 90kg. Design variables were found where the optima occurred on extreme ends. For optima at extreme ends, limiting bounds were chosen from design constraints and practical limits. The main question is the use of aerospike engine which uses 25-30% less fuel at low altitudes, where most missions have the greatest need of thrust. Aerospike engines have been studied for several years and are the baseline engines for many single-stage-to-orbit (SSTO) designs and were also a strong contender for the Space Shuttle main engine. However, no such engine is in commercial production, although some large-scale aerospikes are in testing phases.

FUTURE WORKS

- The present design will be carried further for thermal, structural and performance analyses.
- Thus, obtained results will be validated and verified.
- These validated results will be used for design optimization.
- Overall mission and performance characteristics of the vehicle will be simulated.
- Various systems redundancies will be included based on the analysis and safety factors.
- Key components like aerospike nozzle, grid fins, etc. will be manufactured using 3D printing.

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