An essential requirement of developing the launch vehicle system in 2013

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Abstract

The next generation launch technology requires developing and maturing innovative technologies based on overall propulsion, structure, vehicle systems and ground and flight operation. NASA plans to use existing elements for the boosters, crew, capsule and engines but the cryogenic stages are further element that requires significantly more design and development. Long term human space exploration depends on the development of a sustainable heavy lift launch vehicle (HLV). It must balance the technical and programmatic factors such as reliability, performance, cost, geographical configuration, logistic and assembly as well as in space issues such as mass and maintaining requirements for lunar and Mars mission and rendezvous and docking capability. In this paper examine the problem of designing sustainable heavy lift architecture in three ways. First recent advanced system architecture synthesized apply to HLV more precisely, as are the counter balancing the dynamics of adaptive and architecture lock in. Subsequent cases are studied to understand the evolution of system architecture is leading to the development of the last heavy lift vehicle. Finally, consider the vehicle capacity, aerodynamic effects and trajectory planning with fuel consumption in space requirement.

Keywords: Heavy Lift Vehicle, Launch Vehicle System, Nose cone type and configuration, and Trajectory Planning.

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1. Introduction

NASA's showed a keen interest in developing a heavy lift launch vehicle for future low and high earth orbital launch missions. The launch vehicle required a payload of 60 to 160 metric tons of low earth orbital missions and the payload masses to be adaptable, which may change from one mission to the other. This analysis is considered a conceptual architecture of the vehicle, including the number of stages, engine selection and aerodynamic considerations.

Other initial constraints were defined for the vehicle trajectory and aerodynamic considerations are limited by a maximum dynamic pressure. Developing such a design HLV requires the capability to model the vehicle aerodynamics and engine performance within an ascent trajectory. The trajectory simulation was developed with the help of Matlab computation and validates a trajectory as an initial value problem with boundary conditions. The initial values of velocity, the trajectory path angle, altitude, latitude, longitude, and heading were used to determine a solution to the final flight path angle, circular velocity, and position. Aerodynamic properties are to be evaluated by using the Indian Air Force missile aerodynamic simulator called Missile DATCOM. It gives accurate values which would be considered acceptable for this mission analysis.

2. History of Launch Vehicle

NASA is aggressively pursuing the challenge field of developing space vehicle and flight technologies in an effort to expand their presence of other nations. NASA's took the initiative of Space Launch Systems was introduced in February 2001 to develop technologies and identified options for future space transportation systems, performing the critical analysis necessary for NASA to eventually proceed with full-scale development of a new reusable launch vehicle system.

In November 2002, NASA revised the Integrated Space Transportation Plan to evolve the Space Launch Initiative to serve as a theme for two emerging programs. The first of these plans is intended to provide crew escape and crew transfer on the orbital space plane for International Space Station. The second was the Next Generation Launch Technology (NGLT) program, which requires the technology development for safe launching, routine space access for an exploration.

In 2004, the program was decided to proceed with a Next Generation Launch Vehicle advance phase, which included research and testing of large-scale tanks, engines, and structures. In 2009, NASA has decided to proceed with full-scale development of a specific vehicle enabled by the program's technological advances. A decision also reached in the next decade regarding future development of a reusable hypersonic launch vehicle, it is based on air-breathing propulsion system in progress.

The NGLT program combines elements of two previous research efforts: the original Space Launch Initiative program – which sought to reduce the risk associated with flying a second-generation reusable launch vehicle in the 2012 timeframe and NASA's former Advanced Space Transportation Program, which pursued launch, propulsion, and flight technologies intended to yield options for third-generation launch vehicle concepts capable of flight in the next two Decades.

Now, the Next Generation Launch Technology (NGLT) program seeks to develop and mature innovative technologies based on this procedure. The program is pursuing original research in the areas of propulsion, vehicle systems, structures, and ground operations. Overall, this program is focused on the development of innovative technologies that provide NASA the means of improving safety and lowering launch costs in future missions.

The High Lift Vehicle (HLV) program is pursuing four significant technology areas:

- Improvement of a reusable liquid-kerosene/ liquid-oxygen for rocket booster engine;

- Progress of Hypersonic, air-breathing propulsion and airframe systems;

- Development of crosscutting launch vehicle system technologies, proposed to support a wide-ranging of launch and flight vehicle architectures; and

Analysis activities to guide program investment and to ensure an appropriate fit, not just NASA's and other nations also jointly to develop the research in technology and determine the requirements to meet the entire nation's Hypersonic. Space launch and space technology needs. The most important goal of this research is to increase safety, reliability and to reduce overall costs, flying and maintaining the nation's next generation of space launch vehicle.

3. Aerodynamic Considerations and Limitations

Orbital launch vehicle design is complicated by the influence of an atmosphere. An atmosphere is described as a layer of gases that surrounds a celestial body as a result of the gravity of the specified body. These present gases from a combination of fluids that can hinder or assist the movement of any object within the atmosphere. This movement through an atmosphere is affected by a number of different aerodynamic quantities such as lift, drag, friction, dynamic pressure, heating, acoustics, and stability. These properties are generated as a reaction of an object to the surrounding fluid and must be evaluated to account correctly for an acceptable prediction of any simulated trajectory. In this chapter focus the aerodynamic effects of the vehicle system.

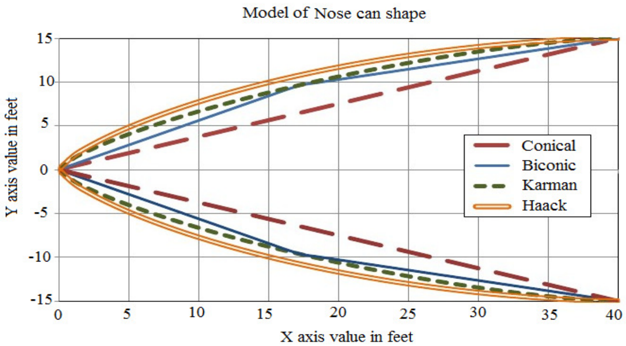


Fig.1 Selection of nose cones.

The figure 1 represents the selection of the nose cone. There are a few factors that generally contribute to design the nose cone configuration. The contributing factors are mass, volume, ease of manufacturing, aero acoustics, and aerodynamic properties. There are various types of nose cones, and each one of them has its own advantages and disadvantages associated with its specification and configuration, .Hence it is an important key role for design of the launch vehicle.

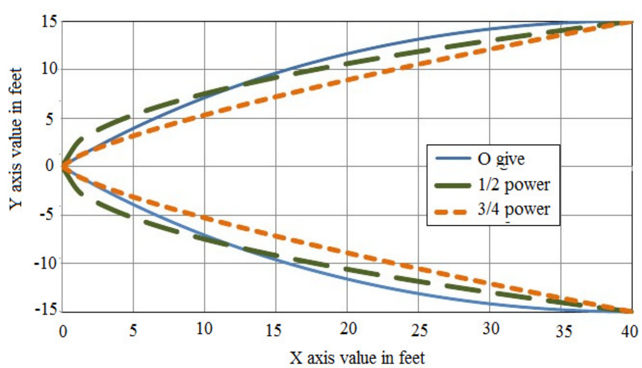


Fig.2 Distinctive nose cone types

An aerodynamic properties are translated to a minimized drag coefficient. The drag coefficient is affected by nose cone edges, curves, area, surface roughness, and the property of the surrounding fluid. This analysis will look at a few different configurations that vary across these parameters. For that simplicity, the surface roughness and the fluid properties (as a function of altitude) are held constant. This is a fair assumption as the surface roughness can be scaled if necessary, and the flight path is restricted to Earth's atmosphere.

While designing a launch vehicle, the major objective is to minimize the ratio of initial to final mass. It can be done by minimizing the mass of all components and in this case minimizing the quivering mass. The mass of a three dimensional shapes is a function of the product of surface area and thickness. When the surface area is minimized, so is the mass. The conical nose cone has the lowest surface area and would be the optimal case if the shroud were chosen based on mass alone. Does the conical nose cone have favorable aerodynamics and aero acoustics, and what is a good balance between minimizing mass and aerodynamic properties? These are questions in the next few sections will address and evaluate.

Volume available within the cover is an important factor because it dictates how much space is available for payload or avionics equipment. A cross section profile is shown in Figure1 and 2 compares distinctive nose cone shapes. The x axis represents the axial position with zero being the nose cone tip, and the y axis illustrates the radial position of each nose cone as a function of the axial position. Commonly, Volume is an essential consideration, but some mission in NASA did not include any bounds in terms of a minimum shroud volume, and as such volume is not a deciding factor for the final nose cone.

Power configurations have the best transonic region drag coefficient but then rise to follow the ¾ power design as a close second. The transonic region is a region of high concern as atmospheric effects are a maximum at this point. By the time a launch vehicle reaches Mach 3-5, drag effects have become negligible as the dynamic pressure has become law.

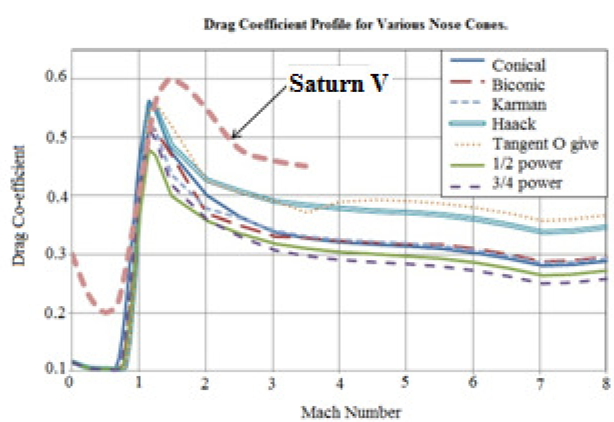


Fig.3 Drag Coefficient Profile for Various Nose Cones.

To aid in validating the Missile Datcom estimation for discrete nose cone shapes is shown in Figure 3 and Figure 4 shows NACA generated fore drag coefficients for three nose cone shapes. The fore drag represents the total drag minus the base drag, and as a result, the fore drag coefficient is less than the total drag coefficient. Even though the magnitude of drag coefficients differs between the two figures, the profiles are comparable. As expected the conical shape to start off with a high drag coefficient and decreases below the Ogive and Haack shapes as Mach number increases. The Ogive and Haack configurations are very similar to the Missile model results, which have to take from DATCOM Analysis. According to the NACA data, these two shapes start off low and then increase to a higher value. It is possible that this inconsistency is associated with the reference length that is added onto the shapes in Figure 3.

4. Nose Cone Types

There are five different types of nose cone shapes in which the designers choose the suitable options. They are e conical, ogive, power series, Karman, and Haack configurations. Each of them is described here to illustrate their shape and some of the basic advantages and disadvantages. Refer back to Figure 1 and 2 for a visual representation. The nose cones are modeled with equations that evaluate the local radius (y) as a function of regional axial position (x), overall nose cone length (L), and the base radius (R).

The conical configuration is the simplest nose cone and as the name indicates, is simply a cone. It is relatively easy to manufacture and is described by Eq. As mentioned before, this nose cone is important because it provides a minimum reference value for cover mass. Any other nose cone with the same base radius and length will have a greater mass. The conical class configuration can be modified to increase the number of conic sections. This modified shape is called the biconic and consists of a cone stacked on top of the frustum of another cone. It is still relatively easy to construct and has potential aerodynamic benefits as well as a larger volume.

The sharp corner has introduced some problems acoustically as there can be oscillating shock waves with supersonic flow. Additionally this shape can be modified further by adding more conic sections resulting in triconic, 4-conic, and 5-conic shapes.

The next nose cone is called an ogive and has a shape formed from a segment of a circle which smoothly meets with the rocket body. It is used because the base of the ogive shroud meets smoothly with the main body of the rocket. Simply put, this eliminates any discontinuities that would otherwise exist from a sharp edge generation where the shroud and rocket body meets. The radius (y) can be represented at any point as in (2)

The power series is a nose cone type that can be described by rotating a parabolic shape around an axis. This shape can be modified by changing the exponent of the parabola (n) from 0 to 1 as in equation (3) Increasing n towards one decreases the bluntness while decreasing n to 0 turns the shroud tip into a point. There is a small discontinuity in the rocket body and cone interface, but the n power can be modified to minimize these effects. The ½ and ¾ powerful nose cones are compared in this analysis.

Haack and Karman nose cones are different because they are not constructed geometrically but they designed mathematically to minimize the drag. These minimizations are constrained by two different factors. In a Haack configuration, the constraints are the length and volume while the Karman configuration is constrained by the length and diameter.

Table1 comparision of fairing and nose cones

	Nose cone and fairing				
Mission	Type	Diameter	Length (mm)	Mass (Kg)	Payload (Kg)
Delta IV Heavy	Unknown	5.1	19.1	3520	21892
Atlas V 500	Unknown	5.4	23.4	4649	17590
Ariane 5		5.4	17	2900	20000
Titan IVB		5.08	26.2	6300	21680
Ares V		10	21.68	13736	130000
Saturn V		6.6	18.8	-	119000

Table 1 shows a comparison of fairing and nose cones used on current and historical launch vehicles. These six cases serve to provide a historical basis when choosing the final nose cone type. The first two uses an unspecified nose cone profile, but based on published pictures, it is expected they use either a ¾ power or an ogive curve. The Titan IVB is approximately the same size as the previous cases but uses the biconic nose cone.

4.1 Properties of Nose Cone

Analysis of Missile DATCOM of the different nose cones with a reference length is shown in Figure 3. These curves illustrate the drag coefficient as a function of Mach number. Angle of attack in this flight regime is in the order of 0.1 and as such, is assumed to have no effect on the drag profile. Immediately the ogive and Haack shape differentiate themselves aerodynamically. The drag coefficients associated with these two are often 10% - 20% higher than the other shapes. The conical shape has a very high CD peak in the transonic region. This is likely due to the sharp tip, and must be carefully considered as these are typically a region of high dynamic pressure, drag, and acoustic effects. Notice the biconic shroud provides an improvement in drag over the conical shape from the transonic to Hypersonic flight regime. Adding an inflection point to the conical shape increases the tip angle which in turn has a positive effect on the drag profile.

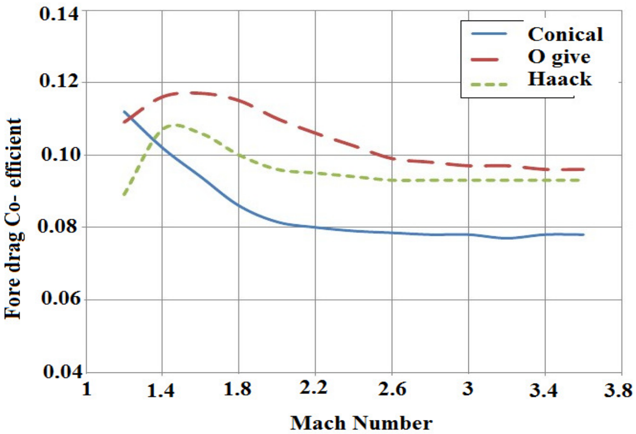


Figure 4 NACA Nose Cone Foredrag Coefficients. For L/D = 3

4.2 Aerodynamic Coefficient Estimation

This section discusses a comparison between the NASA developed aerodynamic data and Missile DATCOM to determine if it is a usable aerodynamic tool. Figures 5 and 6 compare both the drag and lift coefficient data for the two different aerodynamic sources. To validate Missile Datcom as an aerodynamic tool, it was capable of modeling heavy lift launch vehicles. The dimensions and properties of this vehicle are not revealed here due to ITAR regulations. Notice that Missile DATCOM models the wind tunnel drag coefficient profile in Figure 5 moderately well. It creates a curve that follows a profile similar to the data from the wind tunnel experiments. The largest difference is in the Hypersonic region from approximately Mach 4 all the way to the burnout condition. This region is of little importance as the density is low and aerodynamic effects are approximately negligible. The error of this approximation most likely originates in attempting to recreate the vehicle model with boosters or atmospheric conditions within the input deck of Missile DATCOM. It is difficult to exactly match the dimensions and properties of the vehicle because of the limited capability and complicated input files associated with this preliminary analysis tool. There could be some differences between the flow properties used in the wind tunnel and those simulated in Missile DATCOM. In spite of these errors, DATCOM does prove to be an acceptable tool when approximating experimental data for the drag coefficient.

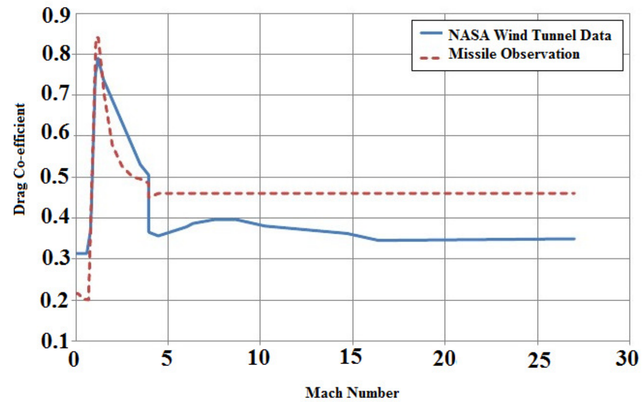


Figure 5 Missile DATCOM Drag Coefficient Estimation is comparable with NASA wind tunnel test data.

The lift comparison is much less reliable in Figure 6 The missile prediction is approximately in order off magnitude of and is a poor representation. It is limited in its ability to calculate lift co-efficient for a such a large vehicle and cannot be used to estimate the lift co-efficient for the configuration in this particular analysis. An attempt was made to locate historical lift data for a vehicle similar to those in the optimization. They resemble the vehicle NASA used in the wind tunnel to generate the lift data, and they can use the same aerodynamic lift data.

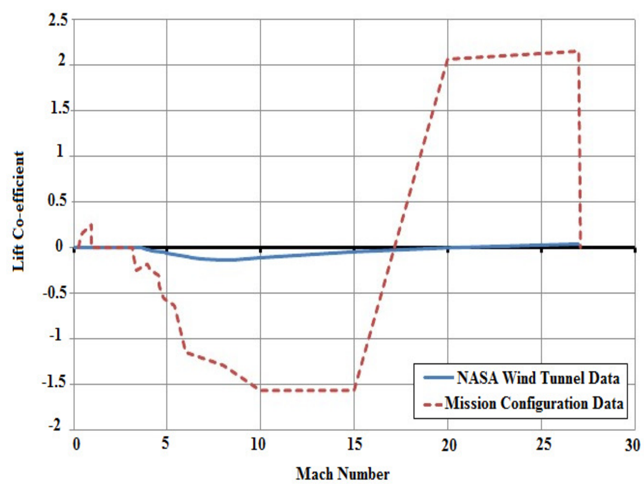


Figure 6 Missile DATCOM Lift Coefficient Estimation is compared in the above shown figure.

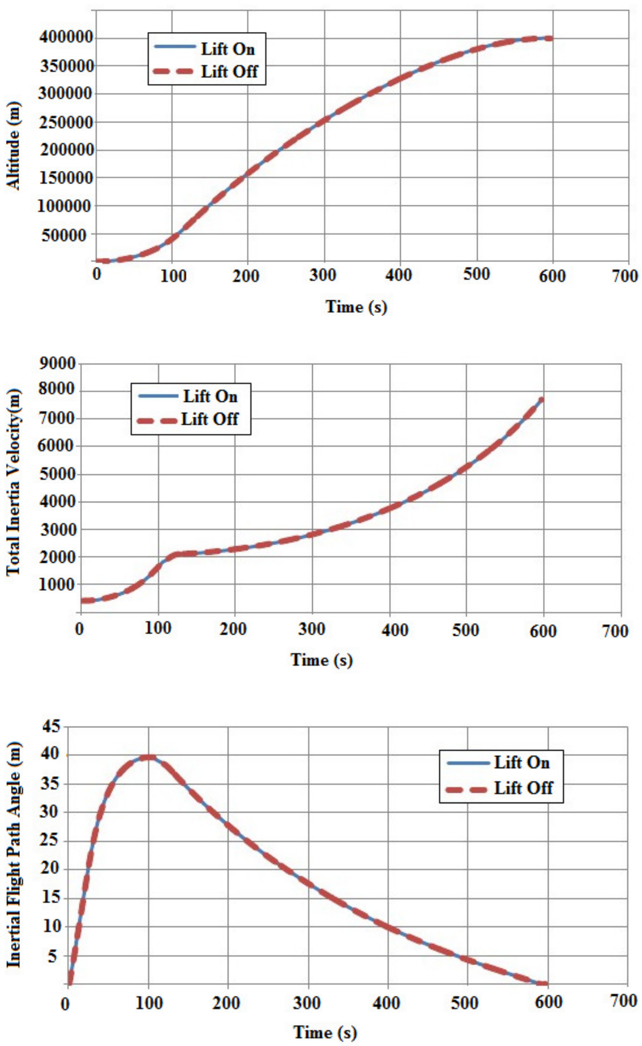


Figure 7 Altitudes, Inertial Velocity, and Inertial Flight Path Angle Comparison.

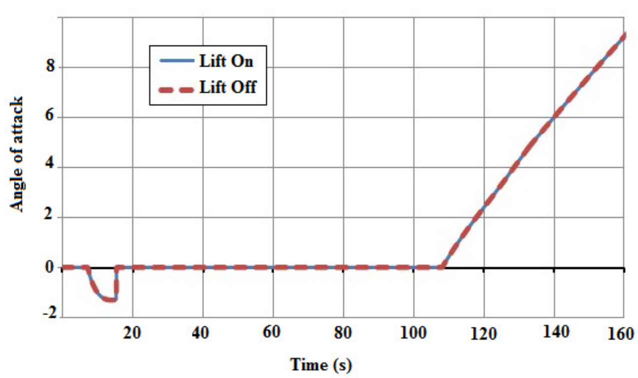


Figure 8 Pitch Induced Angle of Attack and corresponds with lift.

5. Launch Vehicle Systems

There are seven heavy lift launch configurations in this analysis, and this topic details the procedure for optimization as well as the results for each case. The optimization is not as simple as searching through cases to determine which best satisfy the target burnout conditions. It requires a process that is very purposeful. The next few sections will explain further and present the optimized results for each configuration.

5.1 Configuration Types

A payload to orbit of 130 Mt and 100 Mt are considered for each of these configurations. Each configuration is initially optimized to the maximum payload. If successful, each case is then optimized to the smaller payload requirements. Not all of them will be capable of handling the different payload requirements. This analysis reveals the difference between those which are suited towards bringing the maximum payload to orbit, those that have an all around capability, and those that are not suited at all for heavy lift launch.

5.2 Aerodynamics Coefficient

Figures 9 and 10 represent the aerodynamic data for vehicles associated with 2.5 stage RSRM configurations. The lift coefficient profile in Figure 9 is only used for the 130 Mt. The drag coefficient profile for the different payload configurations is shown in Figure 10. The 130 mt payload configuration uses a higher drag coefficient due to the attached boosters. The size of 100 mt payload configurations is expected to be similar, and as a result, the drag coefficient profiles are assumed to be the same

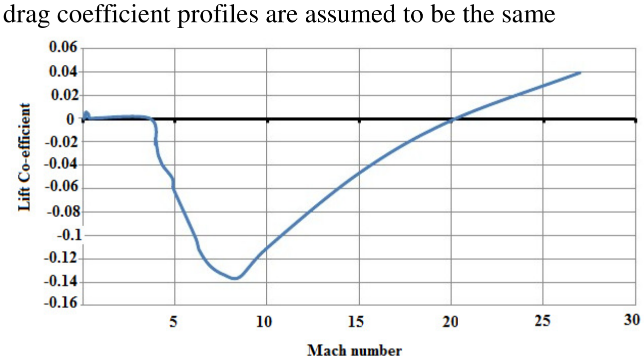


Figure 9 RSRM Configuration Lift Coefficient Profile

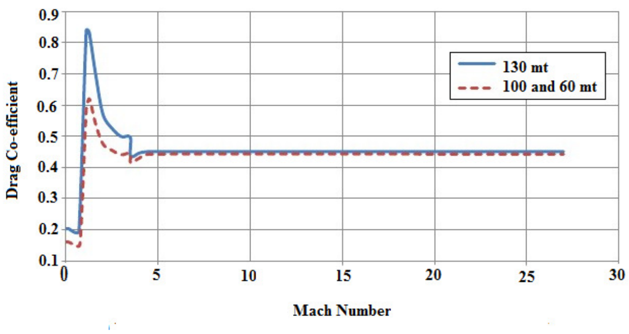


Figure 10 RSRM Configuration Drag Coefficient Profile

This surface shows that the vehicle is somewhat stable in terms of this ΔV property. The location is indicated by the yellow dot in the figure 11. The axis labeled Stage A, dV signifies the portion of the launch were the core engines and boosters burn together. Stage B, dV represents the contribution of the core after the boosters have separated. The position of the dot indicates that a modification of these stage contribution fractions will have a slight but not significant effect on GLOW. Figure 12 verifies that this configuration did not surpass any limits for the constraints of angle of attack, dynamic pressure, and angle of attack*dynamic pressure.

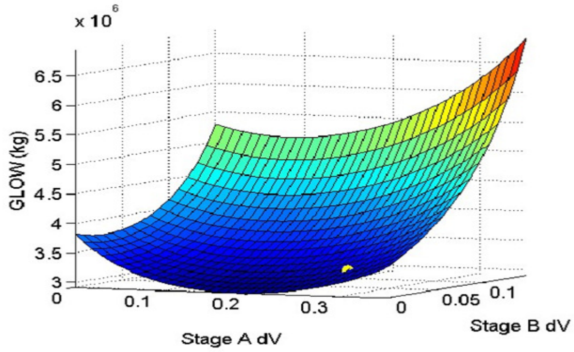
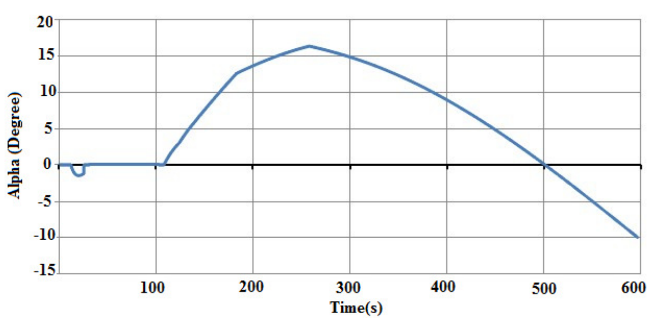
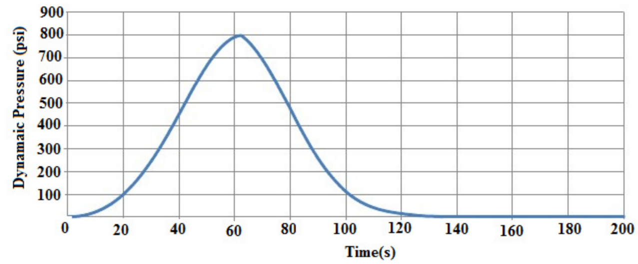


Fig 11. Stability Surface for 100 mt Configuration





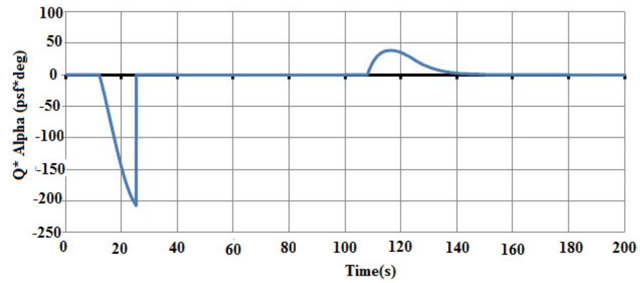


Fig 12. 100 mt Payload Analysis

The next step is to determine if this configuration can be modified to fulfill payload requirements of 130 Mt. These configurations with boosters are very capable as the boosters are well suited to bringing a heavy payload to orbit. This capability is due to the high thrust and large amounts of propellant associated with RSRM boosters. These same qualities can also serve as a hindrance when attempting to scale the higher payload version to that of 100 Mt. Considering the size of these particular boosters, it is a safe assumption to state that any configuration with a payload smaller than 130 mt will most likely fly with two or no boosters. As it is impractical to fly with one booster, these are the only two options

The requirement for this analysis is to determine a vehicle with adaptable payload capabilities that has optimized the 130 mt configuration. In other words, can the capabilities of the vehicle be adjusted without making many significant design changes to the original model? This concept is an adjective because minimizing changes can decrease costs. For example, if the same size propellant tank for stage one can be used on both the 100 mt and 130 mt configurations, the entire process of design and development is simplified. Engineers only have to design one tank, and parts for two different vehicles can be manufactured together. It is beneficial to employ this concept in the smaller payload configurations instead of attempting to minimize the GLOW. This is done for all 100 Mt and 130mt configurations.

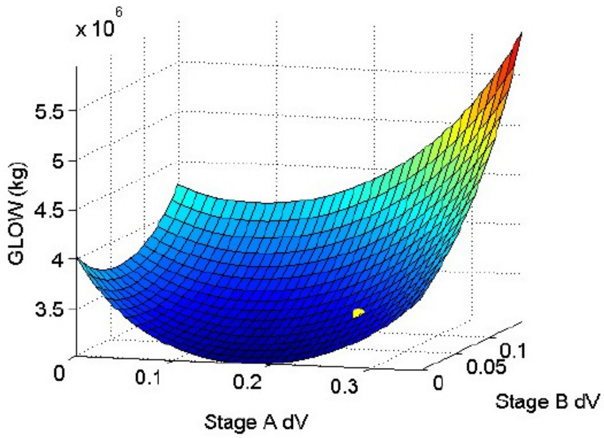


Fig.13 Stability surface on 130 mt configuration

It is important to consider the possible impracticalities of this situation. The majority of 2.5 stage vehicles go through three steps in the propulsion system. The first step consists of the boosters and the first stage engines burning together. Step two also includes the first stage engines but considers them only after the boosters have separated from the vehicle. Lastly, the third step consists of the second stage engines firing until the propellants are exhausted. This complicates the design as it is necessary to allow the boosters to burn in coincidence with the first and second stage. There are ways to modify the vehicle so that this capability can be met, but that will not be discussed here. It is instead important to recognize that any attempt to minimize GLOW for this vehicle increases the complexity of the vehicle as a result of the booster coincident burn. Consideration of this increased complexity should be taken into account when deciding on the final top configurations

This velocity budget method is very important because it provides a visual representation as to how adjusting the adjusting the contribution of a particular stage to the total ΔV affects GLOW of the vehicle. These propellant masses are then simulated in THEO to determine if they are in fact capable of bringing the vehicle to desire orbital conditions

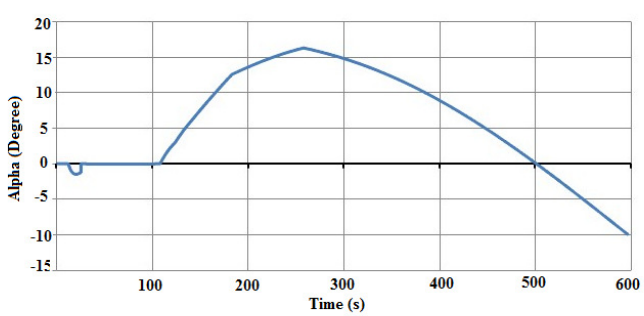


Fig.13(a) Alpha Vs Time of Payload 130 mt.

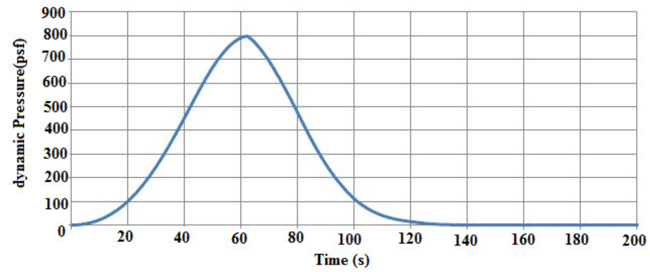
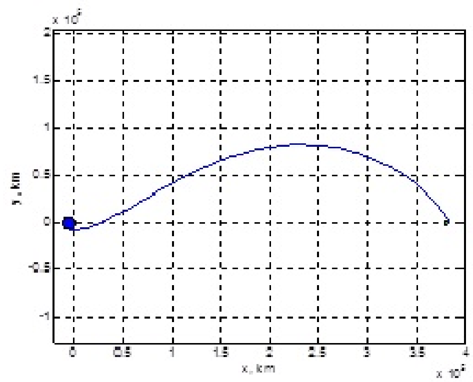


Fig.13(b) Dynamic Pressure Vs Time of Payload 130 mt.

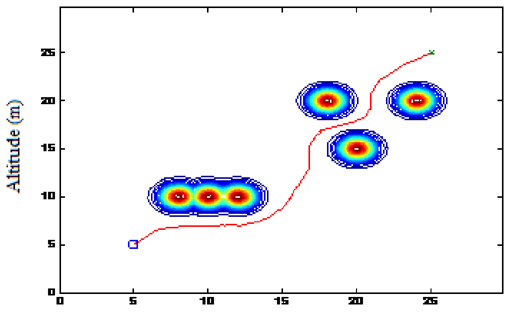
The effect of a ΔV variation on GLOW is shown in Figure 13. It is generated based on a total ΔV requirement of 9700 m/s, specific impulse for the engines, and the inert mass ratios specified in 0.06 and 0.08 for the first and second stage. Total ΔV is an estimation of the necessary energy, in terms of velocity, to overcome adverse effects that are a result of drag, steering, and gravity losses.

6. Trajectory Planning

Trajectory planning has been a topic of momentous research interest in the Aerospace Industry. In this study more helpful to develop a heavy launch vehicle system. A trajectory is the path of a moving object in space as a function of time. The object might be a satellite, or spacecraft, as it travels around a central mass. It's described by mathematically either by the geometry of the path, or the position of the object in time.



14. Trajectory model



The simulated mathematical model is accurate. It wholly depends upon the co-ordinate selection and will be varied when the inclination changes. It can help to improve fuel consumption and avoid the collision from space particle.

7. Conclusion

In this paper examined the problem of designing sustainable heavy lift architectures. The advance system architecture synthesized applied to HLV more precisely, considered aerodynamic characteristics and trajectories planning within the atmosphere as are the counter balancing the dynamics of adaptability and further analyses of launch vehicle systems.

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