Report

On

"Hypersonic Space Plane"

Submitted in fulfillment of the Requirements for the degree of

B. Tech. Aerospace Project

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Certificate

Certified that this B.Tech. Project Report titled "Hypersonic Space Plane" by "Nikhil Chauhan R180206033, Krishankant Soni R18020625 and Tarun Sharma R180206057" is approved by me for submission of report. Certified further that, to the best of my knowledge, the report represents work carried out by the students.

Date: 13.5.2010

Dr. Parag Mantri

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Abstract

The space vehicle being worked upon is an unmanned autonomous vehicle meant for carrying payloads to the circular low earth orbit to an altitude of 100nm or 185.2km with an orbital inclination 28.5 deg using air-breathing propulsion system. The target of the project is to reduce the gross takeoff weight of the vehicle by eliminating the need of carrying oxidizer onboard during takeoff and by using a better aerodynamic configuration. The vehicle must be design to carry the payload of 7500-8000kg. The vehicle will be designed to carry out operations from the major landing strips / airports around the globe and this capability makes it mandatory for the vehicle to take off horizontally. For the horizontal take off we can either have space vehicle fully capable of taking off horizontally on its own or we may have a vehicle that is piggy backed on large transport aircraft such as Boeing 747 or Antonov-225 and released in air at a point from where ramjet engine can take over thus it enables us to eliminate the use of separate turbine engines for subsonic flight regime. The gross takeoff weight can be reduced by increasing the specific impulse of the engine by improving the kinetic energy efficiency of the air-breather vehicle by selecting the appropriate ramp angle.

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1. Introduction

The soul motive of this project is to design a commercially viable and reusable space launch vehicle for space launches to place satellites in the low earth orbit at the minimum possible cost. The designing of the space vehicle revolves around the mission that must be successfully completed by the spacecraft. The must be able to place the satellite at an orbit about 100 nautical miles above the earth surface and safely return back to earth in single piece for being reused as a space launch vehicle. There is a need to eliminate the requirement of constructing specialized launch platforms as required in the case of the vertically launched space vehicles and that could be done by using a space vehicle which can be launched from regular airports around the world. In order to do this the space vehicle must be able to take off conventionally.

Since the orbital speeds of the LEO are quite high thus Hypersonic Space Vehicle is needed to complete the mission profile. The design of this space vehicle is specifically driven by the aero-thermodynamic requirements. The motivation of this project work is the ongoing project AVATAR of DRDO and ISRO, as per the information provided in an article ("The space transportation system in India: Present scenario and future directions") By B.N Suresh in "Journal of aerospace sciences and technologies (February 2009 issue)".

All the current launch systems in operations are entirely expendable or partially reusable as in case space shuttle operated by NASA, require years of preparation time, and are customized for specific payloads. At present launch cost is around $10,000/lb to LEO is an economic constraint for the space organizations around the world. The current generation of Evolved Expendable Launch Vehicles (EELV) does not meet the needs, but reusable launch vehicles (RLVs) have the potential to greatly surpass the abilities of expendable launch vehicles. National Aeronautics and Space Administration (NASA) and the U.S. Air Force have studied numerous RLVs since the beginning of spaceflight to find an affordable, routine, and operationally responsive launch system, but no program reached operational capability due to technological hurdles, political opposition, and large program costs. RLVs are more responsive since they can be designed for aircraft-like operations from existing airports and air force bases, especially if propelled by air-breathing engines. Reusability will reduce the operational costs and life-cycle costs of the system over that of expendable vehicles, if they can be designed to require maintenance practices close to regular aircraft.

1.1 Objective of this project

To design a fully reusable and unmanned space vehicle capable of carrying payloads to the circular low earth orbit to an altitude of 100nm or 185.2km having an orbit inclination of 28.5 deg using air-breathing propulsion system. The target is to reduce the gross takeoff weight of the vehicle by eliminating the need of carrying oxidizer onboard during takeoff. The vehicle must be design to carry the payload of 7500-8000kg. The vehicle must be designed for horizontal takeoff from conventional air strips at the major airports.

For the horizontal take off we can either have space vehicle fully capable of taking off horizontally on its own or we may have a vehicle that is piggy backed on large transport aircraft such as Boeing 747 or Antonov-225 and released in air at a point from where ramjet engine can take over thus it enables us to eliminate the use of separate turbine engines for subsonic flight regime.

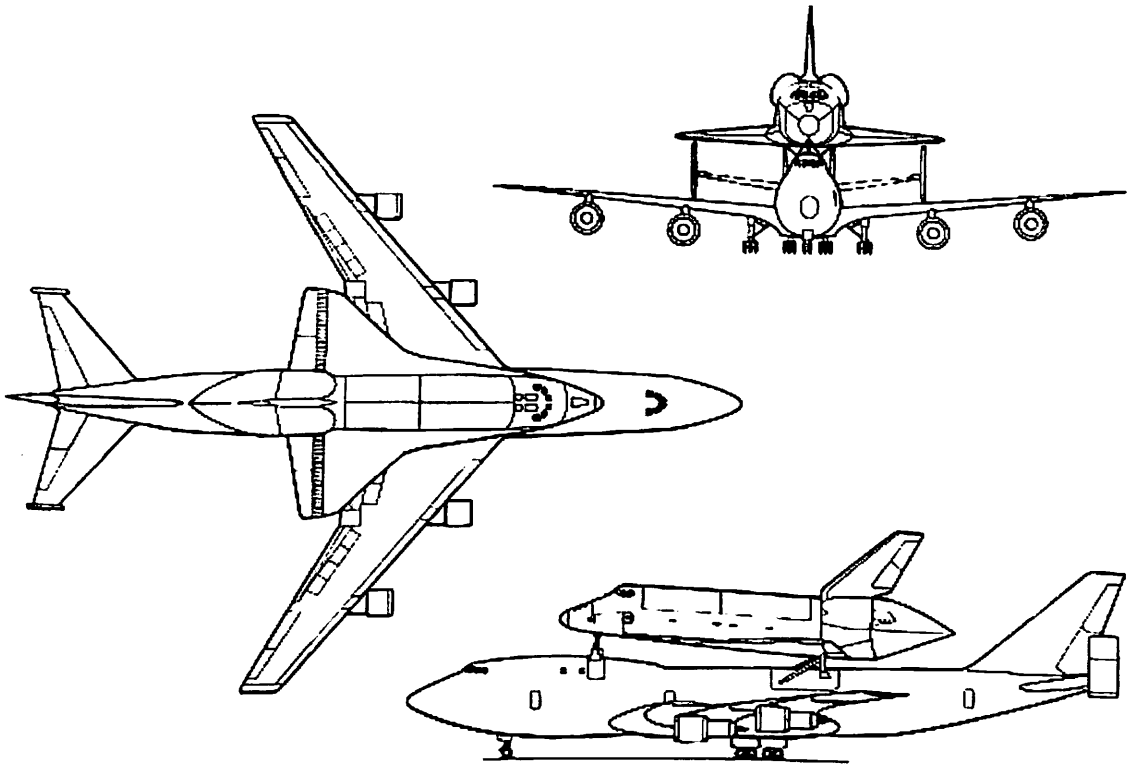
Specifications of Boeing 747 and Space shuttle orbiter

1.2 General characteristics of Boeing 747

- Crew: 4: pilot, co-pilot, 2 flight engineers
- Length: 231 ft 4 in (70.5 m)
- Wingspan: 195 ft 8 in (59.7 m)
- Height: 63 ft 5 in (19.3 m)
- Wing area: 5,500 ft^2 (510 m^2)
- Empty weight: 318,000 lb (144,200 kg)
- Maximum Takeoff Weight: 710,000 lb (322,000 kg)
- Cruise Speed: Mach 0.6 (397 knots, 457 mph, 735 km/h)
- Range: 1,150 mi (1,000 nm, 1,850 km) while carrying Shuttle
- Service Ceiling: 15,000 ft (4,500 m) (with Shuttle)

1.3General Characteristics of space shuttle orbiter

- Length: 122.17 ft (37.237 m)
- Wingspan: 78.06 ft (23.79 m)
- Height: 58.58 ft (17.86 m)
- Empty weight: 172,000 lb (78,000 kg)
- Gross liftoff weight: 240,000 lb (110,000 kg)
- Maximum landing weight: 230,000 lb (100,000 kg)
- Main engines: Three Rocket dyne Block II SSMEs, each with a sea level thrust of 393,800 lb f (1.752 MN)
- Maximum payload: 55,250 lb (25,060 kg)
- Payload bay dimensions: 15 by 59 ft (4.6 by 18 m)
- Speed: 7,743 m/s (27,870km/h; 17,320mph)
- Cross range: 1,085 nm (2,009 km; 1,249 mi)
- Crew: Varies. The earliest shuttle flights had the minimum crew of two; many later missions a crew of five.



Dryden Flight Research Center February 1998
Space Shuttle mated to 747 Shuttle Carrier Aircraft (SCA) 3-view

Source (www.nasa.gov)

1.4 SSTO vs. TSTO

We can either have a single stage or two stage orbit launch vehicles. Both of these configurations have got their share of pros and cons. Two-stage-to-orbit launch vehicles reduce mass during ascent by discarding propellant and structure. The point at which the vehicle expends a portion of its structure is called staging. Single-stage-to-orbit launch vehicles only discard propellant on their way to orbit. For a SSTO, there is an exact trade-off between structural mass and payload mass. SSTO vehicles are very sensitive to vehicle dry mass it is important to reduce the dry mass of the vehicle. By staging, a TSTO vehicle reduces its structural mass during the last phases of flight. This opens the margin of performance to a feasible level for attaining orbit. New advances in propulsion and material science have increased the efficiency of engines and allowed for smaller structural mass fractions. SSTO vehicles have some potential benefits over multi-stage launch systems is that they are more operationally flexible since they do not require the assembly of multiple vehicle components. SSTO vehicles have smaller wetted areas, and that the area using thermal protection system in turn reducing the number of maintenance hours required to turn around a RLV after returning from orbit. SSTO vehicles are very sensitive to the payload masses.

2. Brief review of past reusable launches vehicle projects

The USAF and NASA have been working on reusable launch vehicles since the very beginning of the spaceage; some of the past projects are as follows:

2.1 Dynamic Soarer (X-20A)

This project was started in response of Soviet launch of Sputnik I in 1957. It was designed as a military vehicle for USAF and was intended to be launched using Titan III booster. It measured 10.7m (35 ft) in length in addition to the Titan III and booster. It was called off in 1963. The lifting-body designs of X-20 inspired future X-planes and spacecraft designs.

Fig.1 DynaSoar X-20 (source, www.astronautix.com)

2.2 The Space Shuttle

The Space Transportation System (STS) project was started in1968. It was meant to provide a low cost solution for the space launches; various configurations were investigated by NASA and TSTO, vertical-takeoff horizontal-landing (VTHL) concept was finalized in 1970, the first prototype was completed in 1976. Designated *Enterprise*, the prototype demonstrated the gliding capabilities of the lifting-body design. Using both solid rocket boosters (SRBs), liquid-fuelled rockets and an External Tank (ET), the first operational Shuttle was launched in 1981. The maximum launches ever achieved in one year were eleven. There were many reasons why the Shuttle's launch rate was limited. **However, the most significant factor was the unexpected amount of man-hours required to service and turn-around an orbiter's Thermal Protection System (TPS).** The cost of maintaining the Shuttle, in addition to the cost associated with a manned vehicle, inhibited the program from reducing the cost of launching payloads into orbit.

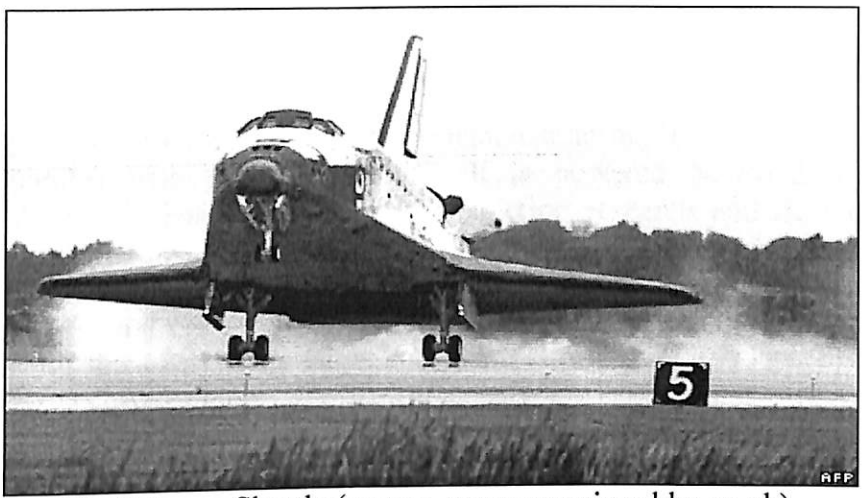


Fig.2 Space Shuttle (source, www.newsimg.bbc.co.uk)

2.3 National Aerospace Plane (X-30)

National Aerospace Plane (NASP) proposed to offer a civilian means of transportation that could, "take off from Dulles Airport and accelerate up to twenty-five times the speed of sound, attaining low earth orbit or flying to Tokyo within two hours." Designated the X-30 by the military, the NASP was a Phase II follow-on to a classified Defense Advanced Research Project Agency (DARPA) program during the early 1980's. The conceptual design consisted of a scramjet-powered, SSTO craft that took off and landed horizontally. The horizontal configuration was necessary for the craft to be used from regular airports. The NASP actively incorporated cooled surfaces. This design system and process pumped cold fuel under surfaces that experienced extreme heating from drag in hypersonic flight before injecting the fuel into the engine. This design process enables higher speeds and increases the efficiency of the combustion in the engine. External rockets would be needed to achieve orbit speeds.

Fig.3 NASP(Source,www.ae.msstate.edu/.../aircraft/x30e.gif)

2.4 Hyper X (X-43)

This a joint program of NASA and USAF to demonstrate air breathing engine capability to power SSTO vehicles with sizable payloads. It is powered NASA-developed hydrogen scramjets. The U. S. Air Force is currently conducting research and development into the production of a scramjet using hydrocarbon fuel. The X-43 is an unmanned experimental hypersonic aircraft design. A winged booster rocket with the X-43 itself at the tip, called a "stack", is launched from a carrier plane. After the booster rocket a modified first stage of the Pegasus rocket brings the stack to the target speed and altitude and it is later discarded followed by X-43 flight using its own engine.

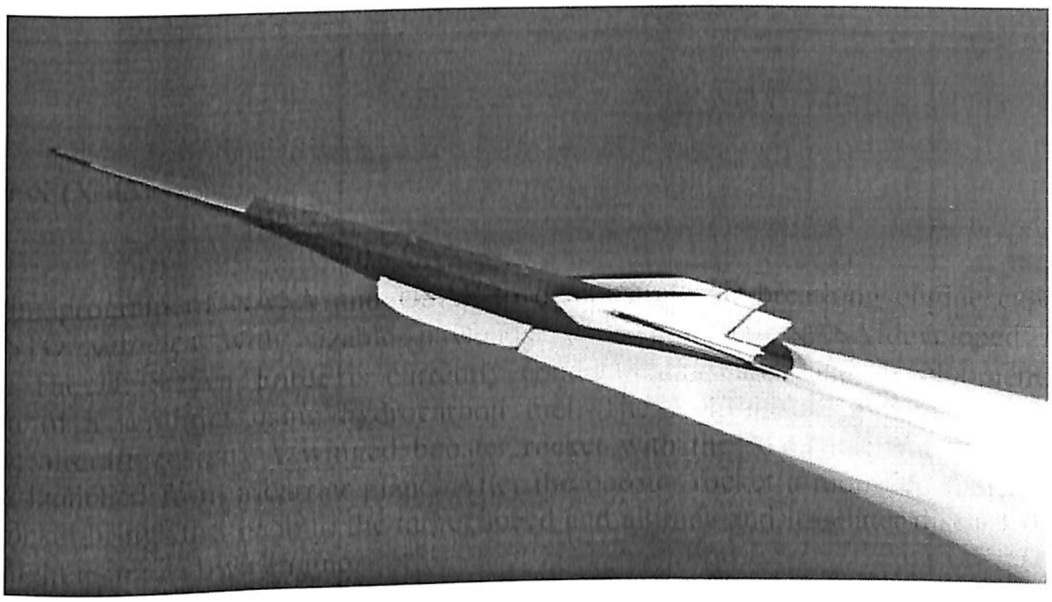


Fig.4 X-43 (source, NASA.gov)

2.5 Following are some more current and past reusable space vehicles

1. EADS's Astrium suborbital HTOL vehicle is a space plane designed to carry four passengers 100 km into space, where they will experience three minutes or more of weightlessness during the 90 minute trip. The Astrium space jet will take off and land conventionally from a standard airport using its jet engines. At about 12 km altitude, a rocket engine takes over to boost the vehicle's altitude to approximately 100 km. After atmospheric re-entry, the jet engines are again restarted for landing.

2. BSC spaceship a Vertical Takeoff and Horizontal Landing VTHL sub oribtal space tourism vehicle proposed by the Benson Space Company. The design is partly inspired by X-2 and X-15 of NASA. Its powered by hybrid rocket motors sourced by Space Dev Inc, and will launch vertically to an minimum altitude of 65 miles.
3. Xerus is a two-person reusable spaceplane proposed by XCOR aerospace, which takes off and lands like a conventional aircraft, capable of climbing sub-orbitally to 100km using a cluster of reusable rocket engines developed by XCOR to use non-toxic propellants. After a short period of weightless free fall the vehicle then re-enters the atmosphere.
4. The Alpha Project is a fully reusable two stage vehicle proposed by World Aerospace Inc, This is a three-person space plane designed to take one pilot and two mission specialists into space. Intended to use off-the-shelf technology where possible, the CX-1A may also be launched from the ground as a suborbital spacecraft.
5. Aeroshell Proposed by Star Raker Associates, Star-Raker is a single stage to orbit horizontal take off and landing vehicle intended to launch payloads of upto 50,000 lbs and 200,000 lbs in its standard and largest configurations to LEO. Low altitude engines comprise of ten supersonic multi-cycle air breather ramjets, based on current existing technology, that lift the vehicle to 100,000ft at a speed of Mach 6 from take off at a conventional commercial airport, at which point rocket propulsion takes over. The aeroshell is a tri-delta form with Whitcomb airfoil lifting sections.
6. SKYLON is being developed by Reaction Engines Ltd. It is an unpiloted fully reusable aircraft-like vehicle capable of transporting 12 tonnes of cargo into space and is intended as a replacement for expensive expendable launchers in the commercial market.
7. AVATAR(Aerobic Vehicle for hypersonic Aerospace Transportation) is a single-stage reusable rocket plane capable of horizontal takeoff and landing, being developed by India's Defense Research and Development Organization along with Indian Space Research Organization and other research institutions; it could be used for cheaper military and civilian satellite launches to Lower Earth Orbit.

3. Propulsion System

3.1 Basic Propulsion Options

All propulsion methods are effectively based upon Newton's third law of motion. They produce thrust by expelling mass, or propellant in the form of a gas out from the nozzle section. There are two basic types of propulsion that can be used for launch vehicle and they are rocket and air breathing engines.

3.2 Air breathing Propulsion

Air breathing engines don't carry oxidizer on board and use atmospheric air and this property has the potential to make the vehicle lighter, but it restricts the zone of operation of the engines only where ambient oxygen is available, thus rendering them incapability to operate beyond atmosphere. These engines have higher specific impulses compared to rocket engines. During ascent of RLVs maximum time is spent in the atmosphere where there is sufficient oxygen to effectively accelerate the vehicle to a point from where rocket propulsion can take over to provide the launch vehicle with the orbital speed.

Ramjets and Scramjets are most apt options for SSTO RLVs. Ramjets are simplest because of no moving parts. Air entering engine is compressed by a series of shock waves to sub-sonic speeds at which fuel is added to the flow ad combusted followed by acceleration of this flow out of nozzle. Ramjets require the forward velocity of the vehicle to compress the air thus they cannot operate alone, deceleration of incoming air to subsonic speeds prevents ramjets from being effective above Mach 6.

3.3 Scramjet Engines

Scramjet engine is based on a modified Brayton Cycle. First of all air is compressed followed by injecting and mixing of fuel to burn and increase the temperature and pressure; finally all these combustion products are expanded. The compression of air in scramjet engine is a result of the forward motion of the vehicle compresses the air, and this is followed by burning of fuel injected into the compressed air. Combustion products are expanded through the nozzle generating the thrust to push the vehicle forward and this is caused by the increased kinetic energy between the initial and final states of the working fluid. This is called a modified cycle because the final state in the scramjet nozzle is not ambient. Specific impulse is the thrust (N) produced per unit mass flow (Kg/s) of propellant used. Rocket propellant includes fuel and oxidizer but for the air breather, fuel is only the propellant carried. Scramjets are more efficient than rockets. Scramjet engine has the ability to move way beyond traditional aircraft's limits of operation. As a result of subsonic combustion in ramjets high static pressure and temperature and high heat loads are generated in the combustor section at high mach numbers placing an upper limit on ramjet operation between Mach 6 and 8. The scramjet overcomes this limit by making use of supersonic combustion. Nozzle throat is absent throat at the end of the combustor in scramjets, supersonic combustion takes place at significantly reduced static pressure and temperature in turn reducing the combustor wall heat loads. The upper limit of the scramjet operation is between Mach 13 and 15 while they can be operated below Mach 6 using mixed mode of combustion. The fuel is injected at the end of combustor the pressure rise because of combustion separates the incoming boundary layer disturbing the flow and forming a recirculation region. This compresses the flow to a reduced speed of Mach 1, which sustains through the initial combustion region, and it accelerates supersonically through the rest of combustor and nozzle. Whenever the combustion takes place in the recirculation and supersonic flow, it is defined as mixed mode combustion. With the capability to operate both is ramjet and scramjet mode these engines are also called dual-mode-scramjet. A Dual Mode Ramjet/Scramjet can operate in both subsonic and supersonic combustion modes. If the overall range of speed of operation is too wide i.e. 1.5-12 specially towards the lower end it becomes necessary to use variable geometry inlet.

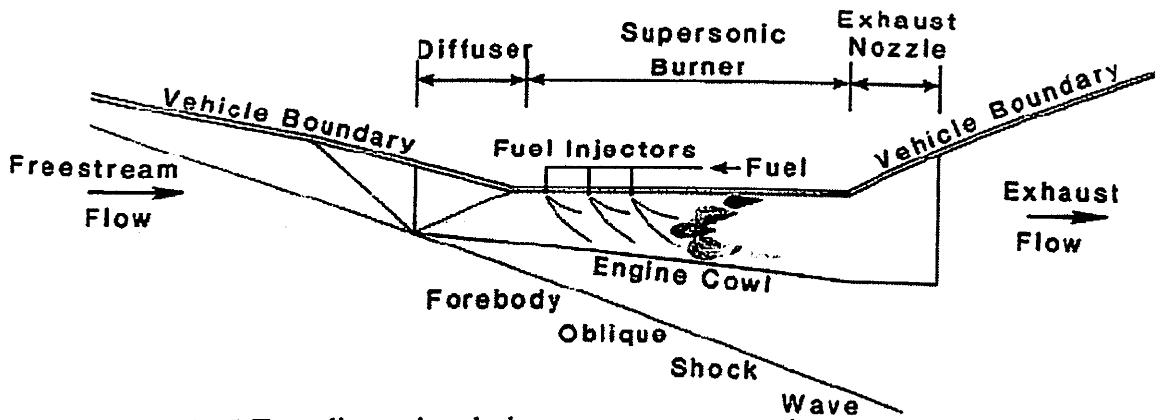


Fig.5 Two dimensional planar geometry scramjet engine([])

3.4 Need of a variable geometry concept

In dual mode scramjet we are required to address the need of operation within Mach 1.5-12 which makes it necessary to use variable geometry inlet in order to provide best acceleration in the air breathing mode. For a fixed geometry combustion chamber with a variable capture area air inlet, the fixed minimum section of the air inlet for the fixed section of the combustion chamber entrance limits the thrust at low Mach number since the incoming air is blocked.

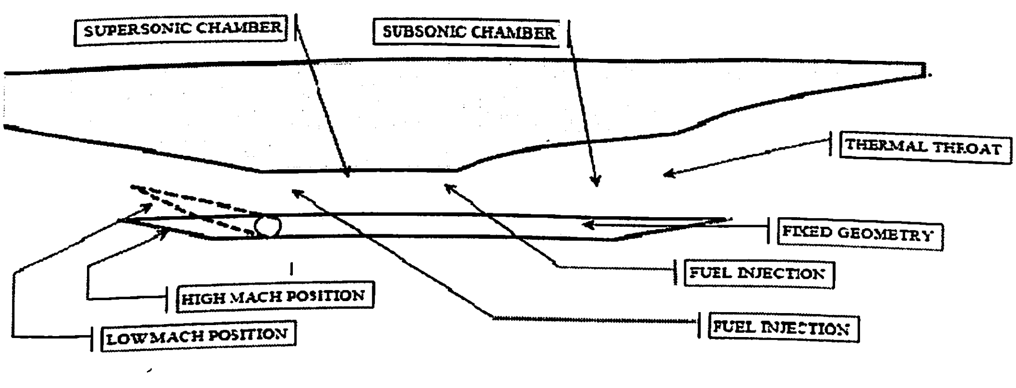


Fig.6 Variable intake configuration of scramjet engine

3.5 Advantages of air breathing propulsion system

1. They have got higher specific impulse compared to rocket engines and are less sensitive to increase in inert mass and are safer. As they utilize the atmospheric oxygen they are not required to carry oxidizer in the vehicle and this helps in reductions of gross take off weights.
2. Rocket usually has a specific impulse ranging between 300 and 500 seconds. Whereas air breathing engines are capable of reaching specific impulses of 7000 seconds. Thus air breathing engines can produce the same thrust as rocket engine but using less propellant. Since they require less propellant per mass of structure and payload, they are less susceptible to vehicle growth due to increases in inert mass. Susceptibility to weight growth is a good indicator of the quality of a vehicle's design. It is important to reduce sensitivity to weight growth to design a vehicle successful vehicle.

3. The design of air breathing systems makes them more reliable than rocket-based systems. Since they operate at lower chamber pressures they have got greater reliability and service life. Safety of the engines is important to maintain overall system reliability. Air breathing engines are less prone to catastrophic failures and provide the crew with time to escape in case of a total failure of the propulsion system.

3.6 Air breathing Propulsion Disadvantages

1. The disadvantages of air breathing propulsion include technical complexity, limited operational zone and air speed, along with engine weight and other penalties.
2. Air breathing propulsion is not capable of taking vehicle to the orbit all by itself. They are not able to operate in oxygen deficient zone of the extreme upper atmosphere and thus they are restricted to the lower portions of a vehicle's trajectory.
3. A particular form of air breathing propulsion can operate over a specific speed range. Ramjets and scramjets require a mean to propel them to their take over speeds.
4. Air breathing vehicles have got more empty mass compared to the rockets and inturn they have a reduced thrust to weight ratio.
5. They are meant to spent most of the time of their mission profile in dense air thus they require a robust thermal protection system. The shape of the vehicle is not very effective in terms of drag reduction.

3.7 Combination/Combined Cycle Engines

Dual-mode scramjet can operate from Mach 3 to Mach 15. It is required to accelerate the vehicle to scramjet takeover speeds. Alternate power source is necessary to for efficient operation below Mach 3-4 for take-off, acceleration, and deceleration for powered landing. The scramjet use at high Mach provides increased payload capability but if the liquid oxygen is used below scramjet takeover Mach number reduces the payload capability. Liquid air cycle engine is an ejector-scramjet has real-time liquid-air collection and compression feeding system. Air breathing launch vehicles requires an additional propulsion system to reach up to orbital velocity and rockets are best suited option for orbit insertion in to the low earth orbit. We also have a combined cycle known as rocket-based combined cycle or RBCC and the major design challenges in this case are placement of rocket and the impact of the rocket engine on the performance of the scramjet. RBCC operates on air augmented rocket mode from Mach 0-3. Once the speed reaches beyond Mach 3 rocket mode is turned off and dual-mode scramjet is activated. Finally the rocket mode begins at Mach 10 with the air-inlets closed before the vehicle leaves the atmosphere.

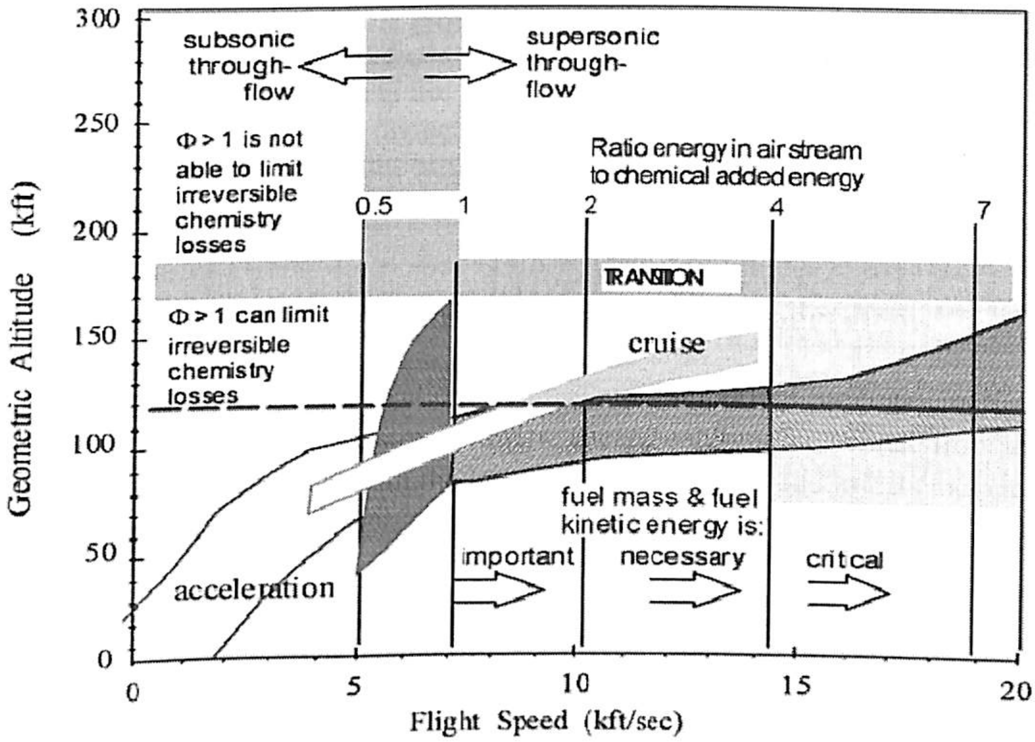
3.8 Challenges [1]

The problems that are repeatedly encountered in hypersonic air breathing propulsion are:

- Operating efficiently and reliably over a varied flight envelope having large range of Mach numbers from 0 to 25 (orbital speed) and moving from sea level to the orbit.
- The practical aerospace plane configurations allow only one inlet section for the engine. Also ramjets and scramjets produce no thrust while standing still or when the atmosphere is too thin, and is further complicated from the point of ordinary "aerodynamic" effects such as lift and drag forces and stability and control moments.
- Accomplishing stable, efficient mixing and combustion in a supersonic flow within a burner of reasonable size.
- Providing the structural integrity necessary for a reusable system despite the extremely hostile environmental conditions.
- Developing the analytical tools that enable confident control over the engine design and reliable prediction of the actual behavior.
- Proving that the aerospace plane and engine are ready for routine operations by means of analysis, ground testing, and flight testing of experimental vehicles.

4. Air breathing Vehicle's operation spectrum

With the increasing speed energy conservation becomes more important for engine performance rather than combustion chemistry. Free stream kinetic energy entering the vehicle from the inlet becomes more significant and critical with the increase in flight speed and affects the operating limits of the air breather vehicle.



The figure above presents the flight corridor followed by the vehicle in order to reach the orbital speed. The regime is decided by the dynamic pressure limits of the air breather with the lower limits a function of skin temperatures and weight of structure. Whereas the upper limit is based on the thrust available for acceleration to orbital speed. The cruising zone as specified on the figure is needed to be followed by the air breather for the maximum possible range .The transition of flow takes place between 5000-7000 fts/sec. To the left of the transition zone compression ratio is too high making the engine less efficient during high speed operation and it is required to limit compression enthalpy by limiting the diffusion in order to keep the speeds in the supersonic zone ahead of the combustion chamber. Kinetic energy is an important source to overcome internal drag and mixing loses. At higher speeds even the energy addition becomes critical. Till date no operational vehicle has ever ventured into the speed regime of energy ratio higher than 4. The thrust to drag ratio decreases with the increasing energy ratios to a point beyond which rocket uses less propellant compared to the scramjet. Thus at this very the transition from air breather to rocket mode is inevitable and this has a practical limit of 4.33 km/sec.

5. RLV Fueling Options

Propellant forms the major component of the mass and volume of a launch vehicle, thus the fuel used can have a major impact on the vehicle's design. The two most commonly used fuels are liquid hydrogen (LH2) and hydrocarbon fuels, such as RP-1. Hydrogen fuel releases around 51,570 btu / lbm (119,954 KJ/kg) and has a density of (70.973 kg/m3), and RP-1 releases around 18,400 btu/lbm (42800 KJ/kg) and has a density of 50.56 lbm/ft3 (810 kg/m3).

Hydrogen fuel is cryogenic and thus it is required to be stored at low temperatures, around 20 K (-424 deg F) to be a liquid whereas Hydrocarbon fuels can be stored even at room temperatures and standard atmospheric pressure. Vehicles using hydrogen fuel instead of liquid hydrocarbon fuel require heavier plumbing, larger diameter pipes due to the low density, and insulation to prevent boiling, all of which increases the structural mass of hydrogen fueled vehicles.

Fuel	Heat of reaction h_{pr} kJ/kg fuel
Hydrogen ,H2	119954
Methane, CH4	50010
Ethane, C2H6	47484
Hexane, C6H14	45100

Heat of reaction of typical fuels used[1]

The storage requirements for liquid hydrogen affects the vehicle design because it possess very low density making it necessary to have large storage tanks and that affects our vehicle design as well as the ground support equipment. While hydrocarbon fuels are much easier to handle than hydrogen and can be stored at room temperatures in normal fuel tanks. Hydrogen fuel is also more expensive than hydrocarbon fuel. The facilities required for hydrogen-fueled vehicles are more extensive and expensive than the facilities for hydrocarbon-fueled vehicles.

For SSTO vehicles, hydrogen provides an overall specific impulse better than hydrocarbon-based fuels because of the higher energy density and provides a source for active cooling of the airframe. The fast chemical kinetics of hydrogen helps reducing the combustion time in the scramjet mode operation. Density of hydrogen can be increased using advanced gelled hydrogen or slush hydrogen. Slush hydrogen yields a 15%[6] increase in density compared to liquid hydrogen and, additionally, it provides 20% greater thermal sink. This is important, particularly in the liquid-air cycle engine (LACE) where hydrogen "recycling" i.e., returning some hydrogen to the slush hydrogen tank for re-cooling, can increase the engine performance, hence improving the vehicle mass properties.

The cooling capacity of the hydrocarbon fuels is less compared to hydrogen and thus they cannot ensure cooling of the combustion chamber of the dual-mode ramjet which is necessary at very high mach numbers to ensure thermal resistance of the combustion chamber and second also to improve mixing and combustion process and maximize the net thrust. In order to improve the cooling capacity the endothermic properties of liquid hydrogen is used by making the fuel components lighter in order to increase the heat absorption capacity.

6. Aerodynamics of the vehicle

Using space shuttle orbiter data, we approximate length and width of our space vehicle so that it could be carried on the back of Boeing 747.

The approximated length and width are 40 and 25 meters respectively. The weight is 70000Kgs approximately. Generating the values of S for various aspect ratio on excel, we have got the following values.

We are using NACA 2415 for our wing design airfoil data taken appendix 3

Aspect Ratio	Wing Span(m)	Span Area(m^2)
3	25	208.33
3	28	261.33
3	30	300
4	25	156.25
4	28	196
4	30	225
5	25	125
5	28	156.8
5	30	180
6	25	104.1667
6	28	130.667
6	30	150

Wing Span (m)	Aspect ratio	Span Area (m^2)	Velocity(m/s)	L(N)
25	3	208.33	340	1701219
28	3	261.33	340	2134017
30	3	300	340	2449795
25	4	156.25	340	1275935
28	4	196.0	340	1600533
30	4	225	340	1837346
25	5	125	340	1020748
28	5	156.8	340	1280426
30	5	180	340	1469877
25	6	104.166	340	850623.6
28	6	130.66	340	1067022
30	6	150	340	1224858

Our estimated weight for space vehicle was 700000N. As to account for flexibility in weight for the variation of our payload and fuel weight .We would consider wing is producing a lift of 1200000N.

Now with our current weight consideration we can have

Aspect ratio=3, B=25m, S=268.33m^2, OR
Aspect ratio=4, B=25m, S=156.25m^2, OR
Aspect ratio=5, B=28m, S=156.80m^2.

Now obtaining data from for the above stated wing configuration form the excel sheet where calculations has been carried out.

AR	S	λ	B	C_{root}	C_{tip}	Λ_{LE}	$\Lambda_{C/4}$	C_{mean}
3	208.33	.15	25	14.49	2.17	25	12.4	8.981
3	208.33	.16	25	14.36	2.25	30	18.56	8.89
3	208.33	.17	25	14.24	2.42	45	37.36	8.81
3	208.33	.18	25	14.12	2.54	50	48.83	8.73
3	208.33	.19	25	14.00	2.66	55	50.22	8.65
3	208.33	.20	25	13.88	2.79	70	68.39	8.57
4	156.25	.15	25	10.86	1.63	25	15.722	6.56
4	156.25	.16	25	10.77	1.72	30	21.61	6.50
4	156.25	.17	25	10.68	1.81	45	39.44	6.44
4	156.25	.18	25	10.59	1.90	50	45.54	6.38
4	156.25	.19	25	10.50	1.95	55	51.55	6.31
4	156.25	.20	25	10.41	2.08	70	68.81	6.25
5	156.80	.15	28	9.739	1.46	25	17.66	5.81
5	156.80	.16	28	9.655	1.54	30	23.38	5.75
5	156.80	.17	28	9.572	1.62	45	40.63	5.69
5	156.80	.18	28	9.49	1.70	50	46.47	5.64
5	156.80	.19	28	9.41	1.78	55	52.25	5.58
5	156.80	.20	28	9.33	1.86	70	69.062	5.53

From the above table we select our geometry for the wing as follows

AR=3,

S=208.33,

$\Lambda LE = 50$,

λ=0.18,

Calculating for the dimension of the wing we have

C_{mean}=8.73m,

C_{root} = 14.12m,
C_{tip} =2.54m,

There is a specific reason for choosing a low aspect ratio for our wing since at high altitudes the density of air is very less which makes it mandatory for the vehicle to be flown at very high mach numbers in order to obtain sufficient lift for our vehicle (refer to appendix 5) on observing the values given in the table that is provided in the appendix we can easily make out that why is that so.

6.1 Calculation of drag polar

$$C_D = C_{D_0} + kC_L^2$$

$$k = \frac{1}{\pi eAR}$$

Where $C_{D0}$the parasite drag component, e is is the Oswald's efficiency for the wing and AR is the aspect ratio that has been taken from the previous section.

Substituting the value of

AR=3 for wing

$\Lambda_{LE} = 50°$

Mean aerodynamic chord L=8.73m {for wing }

Formula used for Oswald's efficiency

$$e = \{4.61(1 - 0.045AR^{0.68})(Cos\ \Lambda_{LE})\} - 3.1$$

e=4.61(1-(0.045*2.1107)0.9358-3.1=0.8043

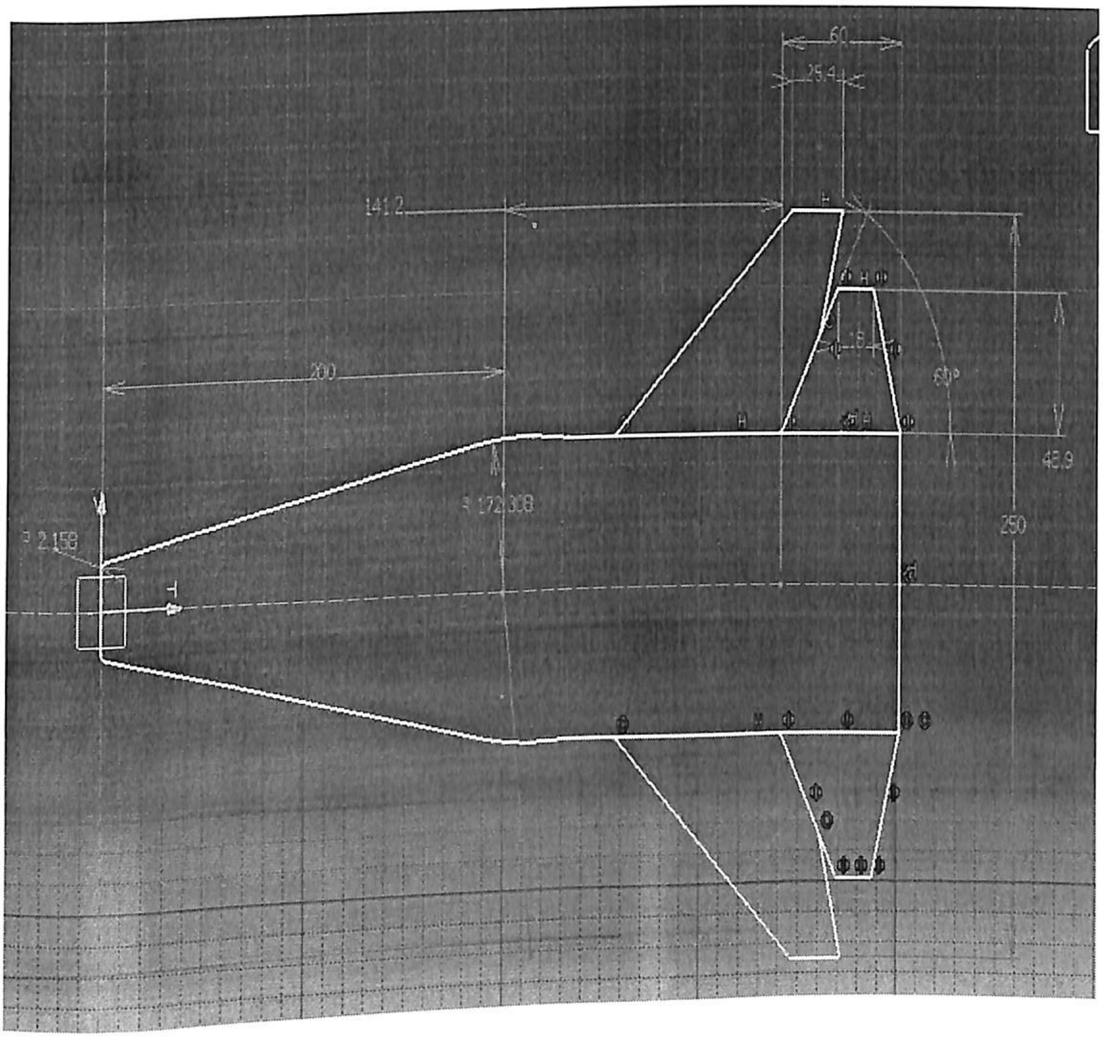
e=0.8043

$$C_{D_0} = \frac{\Sigma(^D/_q)components}{S(planform\ area}$$

$$\left(\frac{D}{q}\right) = C_f \cdot S_{wet}$$

$$C_f = \frac{0.455}{(log_{10}R_e)(1 + 0.144M^2)^{.68}}$$

$$R_e = \frac{\rho v l}{\mu}$$



$$\mu = \mu_o + \alpha t + \beta t^2$$

$\mu_o = 1.7 \times 10^{-5}$

$\alpha = 0.587 \times 10^{-7}$

$\beta = 0.1189 \times 10^{-9}$

t= 288.2K (for M=1)

$\mu = 2.40413 \times 10^{-5}\, Ns/m^2$

For Reynolds's no. calculation of wing:-

$\rho = 1.2256 kg/m^3$

$v = 340.3 m/s$

$l = 8.73 m$

$R_e = 1.5144 \times 10^8$

$C_f = 1.8413 \times 10^{-3}$

$S_{wet} = 898.3\ m^2$

$\left(\frac{D}{q}\right)_{wing} = 1.654$

For M=10

$\rho = 1.0269 \times 10^{-3}$

$t = 270.70626 K$

$\mu = 3.20186 \times 10^{-5}\, {Ns}/{m^2}$

$R_e = 1.0853 \times 10^6$

$C_f = 7.4451 \times 10^{-4}$

$\frac{D}{q} = 0.6688$

For Tail

At M=1;

$\Lambda_{LE} = 30°$;

L=3.5;

e=0.5131

$\mu = 2.40413 \times 10^{-5}\ Ns/_{m^2}$

$R_e = 6.0718 \times 10^7$

$C_f = 2.093 \times 10^{-3}$

$^D/_q = 0.7983$

At M=10

$\mu = 3.2018 \times 10^{-5}\ Ns/_{m^2}$

$R_e = 3.7022 \times 10^5$

$C_f = 7.7188 \times 10^{-4}$

$^D/_q = 2.9439$

$Base\ drag = [0.139 + .419(M - 0.161)^2]$ $Area_{base}$

Base drag = 0.4 Base drag (since in case of air breathers, some spillage was assumed to occur inside the nozzle. The value for base area drag for air breathers was reduced by a factor of 0.6)[ ]
Base drag= 409.64

For fuselage

At M=1;

$$^D/_q = 3.5555$$

At M=10;

$$^D/_q = 6.52148$$

$$(^D/_q)_{total} = 415.6478 \quad (for\ M = 1)$$

$$planform\ Area_{total} = 3639.7m^2$$

$C_{D_o} = 0.1141$ (for M=1)

$$({}^{D}/_{q})_{total} = 419.7775 \qquad (for\ M = 10)$$

$C_{D_o} = .1153$ (for M=10)

$$k = 0.132$$

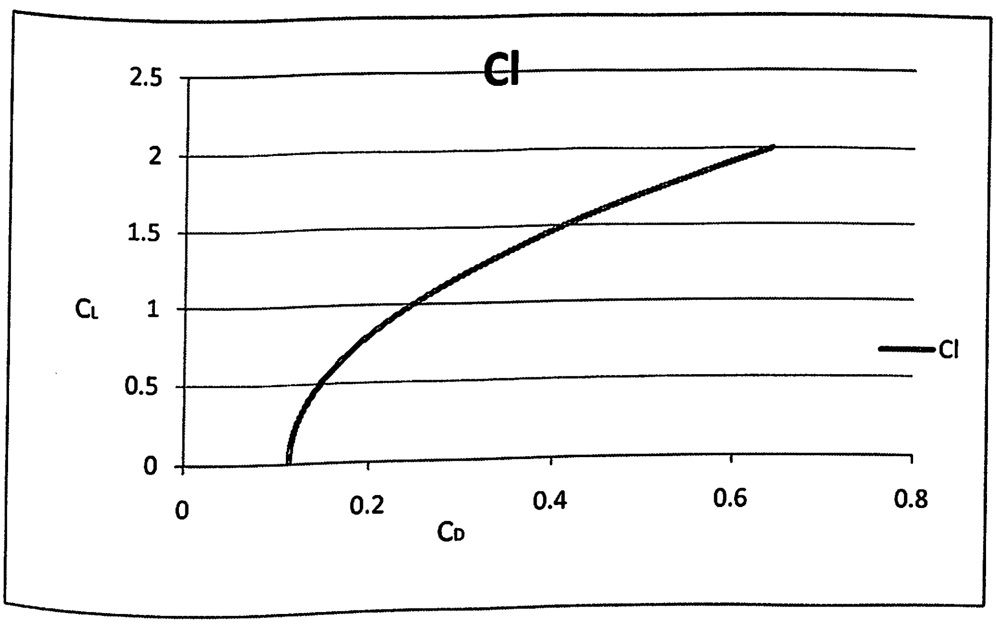
$$C_D = C_{D_o} + kC_L^2$$

At M=1;

$$C_D = 0.1141 + 0.132C_L^2$$

At M=10;

$$C_D = 0.1153 + 0.132C_L^2$$



$$({}^{L}/_{D})_{max} = 4.157$$

$$({}^{L}/_{D})_{cruise} = 0.866 * 4.157 = 3.6$$

For hypersonic configurations our $(L/D)_{max}$ is limited by the following formula

$$(L/_D)_{max} = \frac{4(M_\infty + 3)}{M_\infty}$$

For M=10

$(L/_D)_{max}$ Comes out to be 5.2 while our calculated value of $(L/_D)_{max}$ comes out to be 4.157 i.e. well within the limits of (L/D) as specified by the formula given above.

7. Calculation of fuel mass fraction and empty mass fraction for the vehicle

At the point of release when the vehicle is released from Boeing 747 at M=0.9 the vehicle is needed to accelerate up to M=1.5 giving the value of Δv as 0.204 km/sec and we have got the value of u, effective exhaust velocity for liquid hydrogen from appendix 2

$m_{bo} = m_0 e^{-(\Delta v + g_o t)/u}$

$$Mass\ of\ Propellant = m_o\left\{1 - e^{-\frac{\Delta v + g_o t}{u}}\right\}$$

$$= m_o\left\{1 - e^{-\left(\frac{0.197}{2.342}\right)}\right\}$$

$$= m_o\{1 - 0.9193\}$$

$$= 0.0806 m_o$$

$$Mass\ of\ Propellant \quad = 0.0806 m_o kg$$

Where m_{bo} is the mass of the system at the end of the burnout, Δv is the change in velocity to be obtained by the burning of fuel and 'u' is the effective exhaust velocity of the fuel this value is dependent on the specific impulse of the fuel.

If we assume the gross takeoff weight of our vehicle to be equal to 70000kg the mass of propellant required for the initial boost phase comes out to be 5838 kg.

Fuel mass fraction for the cruising range

For ramjet

$$\pi_f = 1 - e^{-\left[\frac{g_0 R}{\eta_0 h_{pr}(1-\phi e)\frac{C_L}{C_d}}\right]} \quad [1]$$

$$\pi_f = 1 - e^{-\left[\frac{9.81R}{\eta_0(119{,}954)(1-\phi e)\frac{C_L}{C_d}}\right]}$$

$$\pi_f = 1 - e^{-\left[\frac{9.81R}{\eta_0(119{,}954)\left(\frac{L}{D}\right)_{max}}\right]}$$

$$\pi_f = 1 - e^{-\left[\frac{9.81R}{0.22(119{,}954)\left(\frac{L}{D}\right)_{max}}\right]}$$

$$\pi_f = 1 - e^{-\left[\frac{3.717*10^{-4}R}{\left(\frac{L}{D}\right)_{max}}\right]}$$

For ramjet operation the value of η_0 is assumed to be equal to 0.22[1]

For the calculation of fuel mass fraction assume the cruise range to be equal to 100km and L/D max for cruise came out to be 3.6[obtained from calculations shown in the previous section]

Fuel mass fraction = 0.01038 calculated from the formula in equation

For scramjet range the overall efficiency $\eta_0 = 0.36$[1]

$$\pi_f = 1 - e^{-\left[\frac{9.81R}{0.36(119{,}954)\left(\frac{L}{D}\right)_{max}}\right]}$$ [1]

Fuel mass fraction for the range of 6000km comes out to be equal to 0.3786

Thus total fuel mass fraction becomes equal to 0.4724

Inverse of initial mass ratio is known as the payload mass fraction which is the figure of merit in commercial aviation in terms of design.

Calculation of the tank volume

Weight of fuel =0.4724*70000kg =33068kg-4836kg (weight of oxidizer) = 28232 kg

Density of fuel = 70.973 kg/m^3

Volume of tank = 28232 / 70.973=397.78 m^3

7.1 Scope of Improvement in terms of design.....

Specific impulse is the best measure of efficiency of a scramjet engine. The engine that has the maximum specific impulse is considered to be the best of all. The specific impulse of an engine depends upon a number of factors namely specific thrust of an installed engine that has taken engine drag also into account. In the following empirical relationship that is derived the stress is concentrated on the improvement of the intake of the engine in order to improve the specific impulse, while rest of factors are is assumed to be constant.

Assumptions for the following:

1. (f) i.e. fuel to air ratio(stoichiometric) =0.0291
2. η_{KEO} is the overall kinetic energy efficiency of the engine = $\eta_{KEC} x \eta_{KEb} x \eta_{KEe}$, here η_{KEb} and η_{KEe} are the kinetic energy efficiencies of combustion and expansion respectively and they are assumed to be 0.80 and 0.90 respectively[1].
3. Speed of sound at altitude equal to 50 km is 298.4 m/s and the mach number is assumed to be equal to 10.

$$I_{sp} = \frac{1}{g_0 f} * \frac{F}{\dot{m}_0} \quad [1]$$

$$= \frac{1}{g_0 f}\ a_0 M_0 \left[\sqrt{\eta_{KEo}(1+f)\left(1+\frac{\eta_b h_{PR} f}{C_p T_0\left(1+\frac{\gamma-1}{2}\right)M_0^2}\right)} - 1\right] - \left[\left(\frac{C_D A_x}{2A_0}\right) add + \left(\frac{C_D A_x}{2A_0}\right) ext.\right]$$

$$= \frac{1}{9.81*0.0291} * 2984 \left[\sqrt{\eta_{KEO}(1.0291)\left(\frac{10.8}{\left(1+\frac{1.28-1}{2}\right)M_0^2}\right)} - 1\right] - \left[\frac{2}{\gamma_0}\frac{\left(\frac{P}{P_0}\right)A_0}{M_0^2 A_0} + \frac{2}{\gamma_0 M_0^2}\left(\frac{P}{P_0} - 1\right)\frac{A_x}{A_0}\right]\gamma_0$$

$$= \frac{2984}{9.81*.0291}\left[\sqrt{\eta_{KEO}(1.0291)\left(1+\frac{10.8}{15}\right)} - 1\right] - \left[\left(\frac{2}{1.36*100}\frac{\left(\frac{P}{P_0}-1\right)A_x}{2A_0}\right) add + \left(\frac{2}{1.36*100}\frac{\left(\frac{P}{P_0}-1\right)A_x}{A_0}\right) ext.\right]\gamma_0$$

$$= 10452.9\left[1.3304\sqrt{\eta_{KEO}} - 1\right] - 7.3259 * 10^{-3}\left[\left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{2A_0}\right) add + \left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{A_0}\right) ext.\right]$$

$$= 10452.9 * 7.3259 * 10^{-3}\left[1.3304\sqrt{\eta_{KEC} * .8 * .96} - 1\right] - \left[\left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{2A_0}\right) add + \left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{A_0}\right) ext.\right]$$

$$= 76.85\left[\frac{1.1196}{7.3259*10^{-3}}\sqrt{\eta_{KEC}} - 1\right] - \left[\left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{2A_0}\right) add + \left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{A_0}\right) ext.\right]$$

$$= 76.85\left[152.266\sqrt{\eta_{KEC}} - 1\right] - \left[\left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{2A_0}\right) add + \left(\frac{\left(\frac{P}{P_0}-1\right)A_x}{A_0}\right) ext.\right]$$

On the simplification of the above given equation plugging in our assumed values we came to a conclusion that specific impulse of the installed is proportional to the square root of kinetic energy efficiency of compression stage, in case of scramjets the complete fore body of the vehicle acts as a compression device which compresses the flow with the help of shock waves developed because the ramp angles of the intake. Calculating the compression efficiency values for the different ramp angles we found out that the intake ramp angles have an evident effect on the efficiency values. The efficiency values at various angles are tabulated in the following table. The pressure ratios are calculated using the formula stated in the appendix 4

θ	β	P/P_0	η_c
14	19	12.064	0.6043
15	20	13.33	0.5778
16	21.1	14.784	0.573
17	23.1	17.588	0.5473

$$\pi_c = \left\{\frac{1}{\psi(1-\eta_c)+\eta_c}\right\}^{\frac{\gamma_c}{\gamma_c-1}}$$

$$\eta_{KEC} = 1 - \frac{2}{(\gamma_c-1)M_0^2}\left\{\frac{1}{\pi_c}^{\frac{\gamma_c-1}{\gamma_c}} - 1\right\}$$

Using the above stated formula we can find out the kinetic energy efficiency of the intake for the ramp angle whose compression efficiency is 0.6043 the η_{KEC} came out to be 0.9352.

8.Result

It has been found that at higher altitude we are required to use low aspect ratio wings if we are using a constant wingspan because only that can provide sufficient lift to support the vehicle at higher altitudes.

Dimensions of the wing are as follows:

AR=3,

S=208.33,

$\Lambda LE = 50$,

λ=0.18,

Cmean=8.73m

C_{root}= 14.12m,
C_{tip} =2.54m,

Weight of fuel required: 33068 kg for a vehicle having a gross take-off weight equal to 70000kg.

Volume of Tank = 397.785 m^3

The specific impulse of the engine can be increased by increasing the kinetic energy efficiency of the compression stage and that has been shown with the help of calculations which are carried out concentrating upon the kinetic energy efficiency for the compression stage of the intake of the scram jet. With a ramp angle of 14 deg we obtained the best value out of the angles under configuration and that came out to be equal to 0.9352

Appendix 1 Atmospheric Data[3]

Geometric Altitude (km)	Pressure (p/p_{SL})	Temperature (K)	Density (ρ/ρ_{SL})	Viscosity (μ/μ_{SL})	Speed of Sound (m/s)
0	1.0000 E+00	288.150	1.0000 E+00	1.00000	340.29
1	8.8700 E−01	281.651	9.0748 E−01	0.98237	336.43
2	7.8461 E−01	275.154	8.2168 E−01	0.96456	332.53
3	6.9204 E−01	268.659	7.4225 E−01	0.94656	328.58
4	6.0854 E−01	262.166	6.6885 E−01	0.92836	324.59
5	5.3341 E−01	255.676	6.0117 E−01	0.90995	320.55
6	4.6600 E−01	249.187	5.3887 E−01	0.89133	316.45
7	4.0567 E−01	242.700	4.8165 E−01	0.87249	312.31
8	3.5185 E−01	236.215	4.2921 E−01	0.85343	308.11
9	3.0397 E−01	229.733	3.8128 E−01	0.83414	303.85
10	2.6153 E−01	223.252	3.3756 E−01	0.81461	299.53
11	2.2403 E−01	216.774	2.9780 E−01	0.79485	295.15
12	1.9145 E−01	216.650	2.5464 E−01	0.79447	295.07
13	1.6362 E−01	216.650	2.1763 E−01	0.79447	295.07
14	1.3985 E−01	216.650	1.8601 E−01	0.79447	295.07
15	1.1953 E−01	216.650	1.5898 E−01	0.79447	295.07
16	1.0217 E−01	216.650	1.3589 E−01	0.79447	295.07
17	8.7340 E−02	216.650	1.1616 E−01	0.79447	295.07
18	7.4663 E−02	216.650	9.9304 E−02	0.79447	295.07
19	6.3829 E−02	216.650	8.4894 E−02	0.79447	295.07
20	5.4570 E−02	216.650	7.2580 E−02	0.79447	295.07
21	4.6671 E−02	217.581	6.1808 E−02	0.79732	295.70
22	3.9945 E−02	218.574	5.2661 E−02	0.80037	296.38
23	3.4215 E−02	219.567	4.4903 E−02	0.80340	297.05
24	2.9328 E−02	220.560	3.8317 E−02	0.80643	297.72
25	2.5158 E−02	221.552	3.2722 E−02	0.80945	298.39
26	2.1597 E−02	222.544	2.7965 E−02	0.81247	299.06
27	1.8553 E−02	223.536	2.3917 E−02	0.81547	299.72
28	1.5950 E−02	224.527	2.0470 E−02	0.81847	300.39
29	1.3722 E−02	225.518	1.7533 E−02	0.82147	301.05
30	1.1813 E−02	226.509	1.5029 E−02	0.82446	301.71
31	1.0177 E−02	227.500	1.2891 E−02	0.82744	302.37
32	8.7743 E−03	228.490	1.1065 E−02	0.83041	303.02
33	7.5727 E−03	230.973	9.4474 E−03	0.83785	304.67
34	6.5473 E−03	233.743	8.0714 E−03	0.84610	306.49
35	5.6708 E−03	236.513	6.9089 E−03	0.85431	308.30
36	4.9200 E−03	239.282	5.9248 E−03	0.86247	310.10
37	4.2758 E−03	242.050	5.0902 E−03	0.87059	311.89
38	3.7220 E−03	244.818	4.3809 E−03	0.87866	313.67
39	3.2452 E−03	247.584	3.7769 E−03	0.88669	315.43
40	2.8338 E−03	250.350	3.2618 E−03	0.89468	317.19

Geometric Altitude (km)	Pressure (p/p_{SL})	Temperature (K)	Density (ρ/ρ_{SL})	Viscosity (μ/μ_{SL})	Speed of Sound (m/s)
41	2.4784 E−03	253.114	2.8216 E−03	0.90262	318.94
42	2.1709 E−03	255.878	2.4447 E−03	0.91052	320.67
43	1.9042 E−03	258.641	2.1216 E−03	0.91838	322.40
44	1.6728 E−03	261.403	1.8440 E−03	0.92620	324.12
45	1.4715 E−03	264.164	1.6051 E−03	0.93398	325.82
46	1.2962 E−03	266.925	1.3993 E−03	0.94172	327.52
47	1.1433 E−03	269.684	1.2217 E−03	0.94941	329.21
48	1.0095 E−03	270.650	1.0749 E−03	0.95210	329.80
49	8.9155 E−04	270.650	9.4920 E−04	0.95210	329.80
50	7.8735 E−04	270.650	8.3827 E−04	0.95210	329.80
55	4.1969 E−04	260.771	4.6376 E−04	0.92442	323.72
60	2.1671 E−04	247.021	2.5280 E−04	0.88506	315.07
65	1.0786 E−04	233.292	1.3323 E−04	0.84476	306.19
70	5.1526 E−05	219.585	6.7616 E−05	0.80346	297.06
75	2.3569 E−05	208.399	3.2589 E−05	0.76892	289.40
80	1.0387 E−05	198.639	1.5068 E−05	0.73813	282.54
85	4.3985 E−06	188.893	6.7099 E−06	0.70677	275.52

Reference values: $p_{SL} = 1.01325 \times 10^5$ N/m^2; $T_{SL} = 288.150$ K
$\rho_{SL} = 1.2250$ kg/m^3; $\mu_{SL} = 1.7894 \times 10^{-5}$ kg/s·m

Appendix 2 Rocket motor data [3]

Parameter	RD-180	SSME
Fuel	RP-1	H_2
Oxidizer	LOX	LOX
Mixture Ratio	2.6/1	6/1
T/W Ratio(rocket)	80	73.3
Nozzle Area Ratio	36.4	77.5
Chamber Pressure(psi)	3772	3260
Characteristic Velocity(fps)	5914	7684
I_{sp}-Sea Level(s)	311	370.8
I_{sp}-Vaccum(s)	337	454.4
Average Thrust-Sea Level(s)	860	418130
Average Thrust-Vacuum(s)	933000	512410
Weight(lbf)	11675	6990
Length(ft)	13	14
Diameter(ft)	9.8	8

Appendix 3[4]

NACA 2415

C_{lmax} = 1.281,

C_{lmax} angle = 11.5

Max L/D = 40.672,

Max L/D angle = 6.5

Stall angle = 11.5,

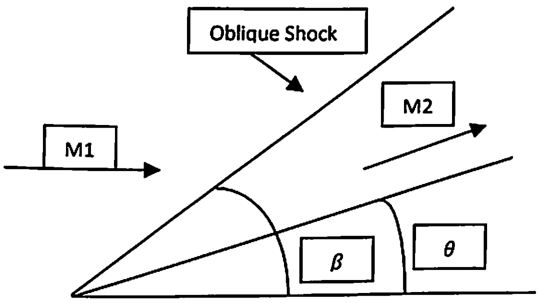
Max L/D C_l= 0.991,

Zero lift angle =-2.0

Appendix 4

TWO-DIMENSIONAL OBLIQUE SHOCK FUNCTIONS

For a two-dimensional oblique shock as shown in Figure the following property ratio apply across the shock.



Because a two-dimensional oblique shock acts as a normal shock perpendicular to the flow, the normal shock relations can be applied to oblique shocks. The normal shock relations of App. F apply, with $M_x$replaced by $M_1 sin\beta$ and M_y replaced by $M_2\sin(\beta-\theta)$. So, we can write:

$$\frac{P_2}{P_1} = f_1(M_1 sin\beta) = \frac{2\gamma}{\gamma+1}(M_1 sin\beta)^2 - \frac{\gamma-1}{\gamma+1}$$

$$\frac{\rho_2}{\rho_1} = f_2(M_1 sin\beta) = \frac{\frac{(\gamma+1)^2}{2(\gamma-1)}(M_1 sin\beta)^2\left[\frac{2\gamma}{\gamma+1}(M_1 sin\beta)^2 - \frac{\gamma-1}{\gamma+1}\right]}{\left[1+\frac{\gamma-1}{2}(M_1 sin\beta)^2\right]\left[\frac{2\gamma}{\gamma+1}(M_1 sin\beta)^2 - 1\right]}$$

$$\frac{T_2}{T_1} = f_3(M_1 sin\beta) = \frac{\left[1+\frac{\gamma-1}{2}(M_1 sin\beta)^2\right]\left[\frac{2\gamma}{\gamma+1}(M_1 sin\beta)^2 - 1\right]}{\frac{(\gamma+1)^2}{2(\gamma-1)}(M_1 sin\beta)^2}$$

Appendix 5

Values calculated at M=8 at an altitude of 50 km

B	AR	S m^2	M	V(m/s)	L(Newtons)
25	3	208.3333	8	2638.4	855467.886
28	3	261.3333	8	2638.4	1073098.92
30	3	300	8	2638.4	1231873.76
25	4	156.25	8	2638.4	641600.914
28	4	196	8	2638.4	804824.187
30	4	225	8	2638.4	923905.317
25	5	125	8	2638.4	513280.731
28	5	156.8	8	2638.4	643859.35
30	5	180	8	2638.4	739124.253
25	6	104.1667	8	2638.4	427733.943
28	6	130.6667	8	2638.4	536549.458
30	6	150	8	2638.4	615936.878
25	3	208.3333	9	2968.2	1082701.54
28	3	261.3333	9	2968.2	1358140.82
30	3	300	9	2968.2	1559090.22
25	4	156.25	9	2968.2	812026.157
28	4	196	9	2968.2	1018605.61
30	4	225	9	2968.2	1169317.67
25	5	125	9	2968.2	649620.926
28	5	156.8	9	2968.2	814884.489
30	5	180	9	2968.2	935454.133
25	6	104.1667	9	2968.2	541350.771
28	6	130.6667	9	2968.2	679070.408
30	6	150	9	2968.2	779545.111

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