

Module-8: Hypersonic vehicle design consideration

Lecture: 39: Hypersonic vehicle cooling strategies

39.1. Design Considerations of Hypersonic Vehicles

39.1.1. Major concerns of hypersonic flight

The design of a typical hypersonic vehicle can be undertaken by a successful experimental test in a ground based facility. However, design criteria of any flight vehicle are governed by the regime of the flow, which the vehicle is going to encounter. ‘Streamlined body’ is the criterion for design of a subsonic vehicle while ‘reduction of wave drag’ is the criterion for design of a supersonic vehicle. Earlier civil or fighter aircraft and missiles designed all over the globe were based on these themes. Excessive heating is the greatest concern in the design of ballistic missiles and spacecraft, since it could melt their surface. Temperature at the nose of the hypersonic vehicles, flying with the Apollo reentry speed, will be around 11,000 K. Hence, design of hypersonic vehicle is dominated by aerodynamic surface heating where ‘reduction of heat transfer rate’ plays an important role. Allen Julian showed that stagnation point aerodynamic heating varies inversely to the square root of the nose radius. This means the aerodynamic drag coefficient is inversely proportional to the heat load. Therefore, the blunt body configuration, which has a detached shock wave, experiences less heating than the traditional shape with its attached shock wave, is the choice of the hypersonic vehicle. Spacecrafts for the Mercury, Gemini, and Apollo programs were designed using this concept. However the maximum temperature that a space vehicle experiences in its hypersonic flight is far above the maximum sustainable temperature of any material. Hence, a proper heat shield should be designed to withstand the heating loads. Consequently nose bluntness increase the aerodynamic drag experienced by the body. Increase in wave drag is useful during reentry of the spacecraft for aero breaking. Moreover, it is disadvantageous during the ascent stage for a vehicle since increased wave drag demands for more fuel. Therefore, different heat transfer and drag reduction techniques are devised for safer and cheaper hypersonic flight.

39.2 Cooling techniques

The cooling techniques to remove thermal energy from the surface of spacecraft are broadly classified into two categories such as active cooling and passive cooling techniques. ‘Radiative shielding’ (e.g. Molybdenum and Zirconium) and ‘Insulation’ (e.g. Dynaquartz) cooling are the most widely used techniques in the area of passive cooling. A schematic of these techniques is shown in **Fig. 39.1** **Fig. 39.2** respectively. These techniques are used to avoid the oncoming heat to the vehicle and therefore they are called passive cooling techniques. ‘Convective cooling’, ‘ablative cooling’, ‘transpiration cooling’ and ‘film cooling’ are classified as ‘active cooling techniques’. These techniques are directly used to cool the vehicle surface.

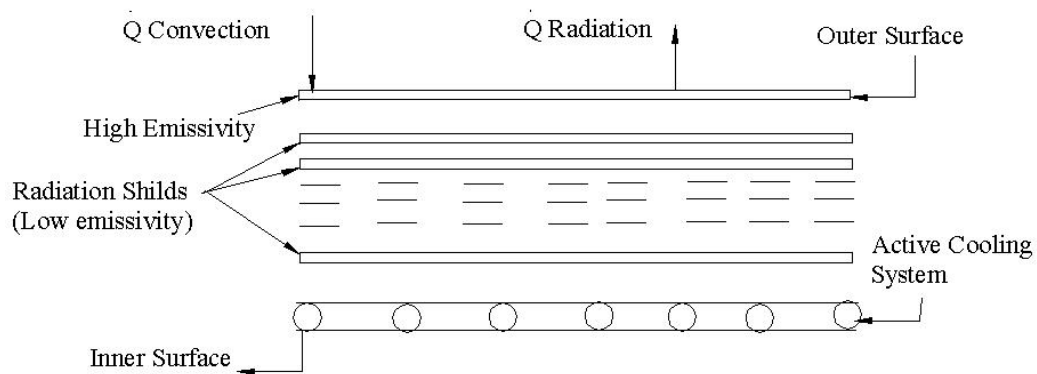


Fig.39.1 Typical Radiative cooling system.

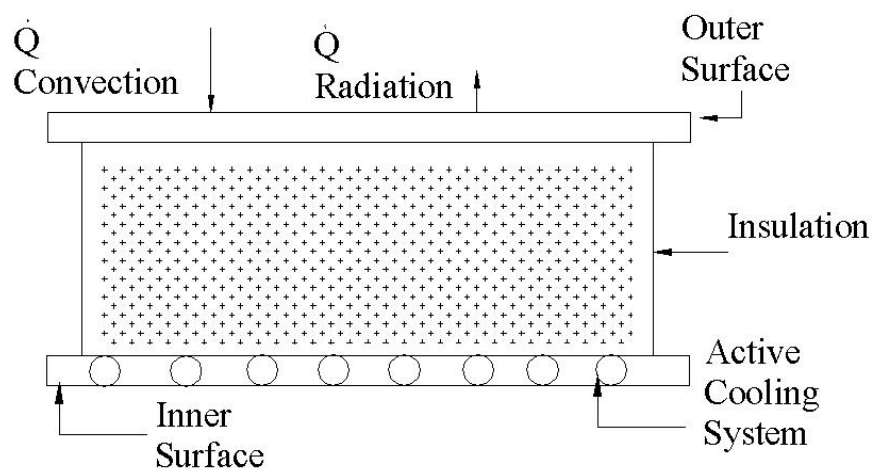


Fig.39.2 Typical insulation cooling system.

The ‘convective cooling’ can be further divided into ‘direct convective cooling’ and ‘indirect convective cooling’. In both the techniques, the heat absorbed by the coolant is used to raise its sensible heat or to change coolants phase. Typical convective cooling technique is shown schematically in the **Fig. 39.3**. In the direct convective cooling, the coolant is passed directly through the surface which needs to be cooled, while in the indirect convective cooling, one or many heat transfer loops are incorporated.

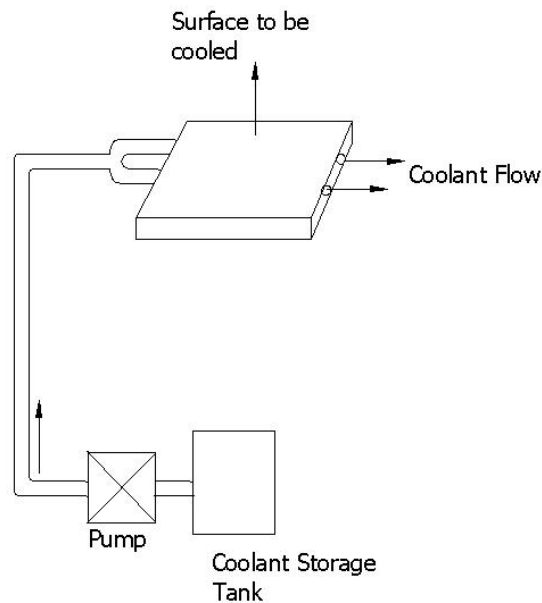


Fig. 39.3. Typical Convective cooling system.

In ‘ablation’ type active cooling, a layer of ablative material is coated over the surface to be protected. The protective layer melts and vaporizes due to heat load absorbing large amount of thermal energy. A typical ‘ablation cooling’ system is shown in **Fig. 39.4**. Graphite and phenolic materials are currently popular ones for ablative materials. It is also found that the vehicle can be cooled efficiently by transferring coolant mass in the boundary layer from its surface.

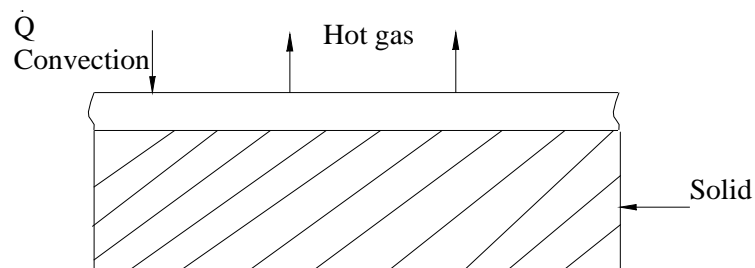


Fig. 39.4 Ablation cooling system.

Mass transfer cooling is an evolving solution to the aerodynamic heating problem. Mass transfer cooling is a type of active cooling technique. The field of mass transfer cooling can be further classified into ‘transpiration cooling’ and ‘film cooling’. In transpiration cooling, coolant mass is injected into the boundary layer from a porous surface of the vehicle so that the coolant comes out as a continuous mass stream and not as an individual jet. Typical transpiration cooling method is shown in the **Fig. 39.5**. The film cooling technique uses discrete hole injection of coolant to come out of a large number of jets, as shown in **Fig. 39.6**. Liquid, chemically inert or active gas can be used as coolant in the case of mass transfer cooling. Liquids are generally not preferred due to formation of blockages. Light molecular weight inert gas like helium is most preferred for mass transfer cooling.

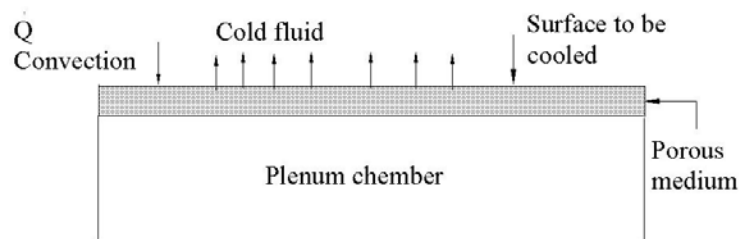


Fig.39.5 Transpiration cooling system.

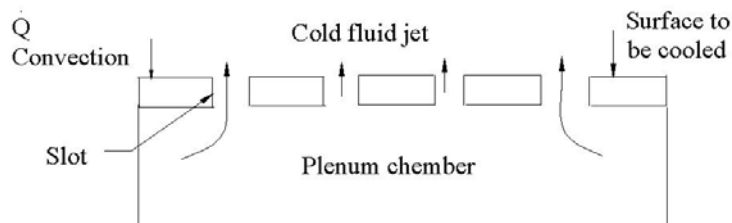


Fig.39.6 Film cooling system.

Lecture: 40: Drag reduction methods

40.1 Drag reduction techniques

Imposed bluntness at the nose of the hypersonic vehicle is necessary to alleviate the oncoming heat load. However, increased wave drag is the immediate consequence of the forced bluntness. The required fuel of the propulsive vehicle increases due to a large drag force. It is also observed that a marginal change in the drag force produces drastic change in the range of the missile or payload of the vehicle. Therefore, numerous wave drag reduction techniques are devised.

40.2 Aero-Spike Based Drag Reduction Studies

The idea of structural aerodynamic spike is to offer an effective passive means to reduce the wave drag. The spike recasts the original bow shock into a conical shock and induces a low pressure recirculation region, which together forms an aerodynamic configuration similar to that of a slender body ahead of blunt body. The schematic of the large angle blunt cone with aero-spike is shown in **Fig. 40.1**. The adverse pressure gradient incurred due to the presence of blunt body and friction offered by the spike jointly separate the flow near the stagnation region in the wake of the aerospike. The consequence of separation is the formation of a low pressure and low temperature recirculation region or separation bubble near the stagnation region. Presence of the low pressure and low energy flow ahead of the body reduces the wave drag. Separated flow reattaches at the blunt body and the location of this reattachment point depends on the spike length, spike configuration and freestream conditions. A reattachment shock appears at the reattachment point to turn the flow accordingly. Drag reduction for a blunt cone model using the aero-spike has been a topic of research from the last five to six decades. Numerical and experimental investigations have been carried out for a variety of objectives. Different lengths and geometries of spike, Reynolds number, Mach number and different test configurations have been considered as typical parameters during these studies.

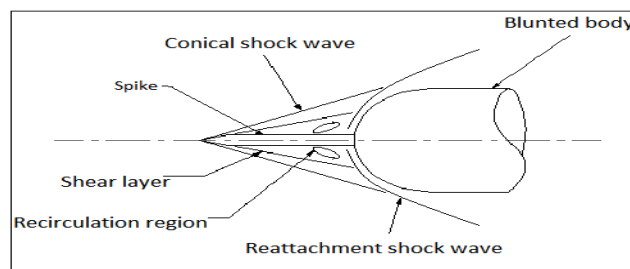


Fig.40.1. Typical Spiked configuration for drag reduction

40.3 Counterflow Drag Reduction Technique

Injection of a supersonic jet from the stagnation point of a blunt body changes the pressure and temperature distribution over the configuration of interest. The expected flow field around the blunt cone with counter flow injection is shown in **Fig. 40.2**. When the gas of total pressure, P_{0j} , and total temperature, T_{0j} , is injected from the stagnation point then the formation of jet and its properties at the exit of the orifice depend on the ratio, P_{0j}/P_{02} , where P_{02} is the freestream pitot pressure. If P_{0j} is very high in comparison with P_{02} , to achieve the choked flow at the exit, then the jet separates at the edge of the orifice and moves forward. Separation of the jet at the edge forms a toroidal recirculation region near the stagnation point. Expansion of the jet from the orifice continues until it passes through a terminal shock. An interface gets formed after the terminal shock at the stagnation point where jet meets the freestream decelerated by the bow shock. This stagnation point is normally called as ‘free stagnation point’, since position of this stagnation point depends on P_{0j} for the given freestream conditions. Heat, mass and momentum transfers are expected to take place between freestream and jet across the interface. This jet deflects from the free stagnation point and reattachment of the jet layer on the body takes place with a turning shock. Fluid in the recirculation region gets entrained in the jet layer forming the shear layer. Formation of such a low pressure and low temperature recirculation region gives rise to reduction of wave drag and heat transfer.

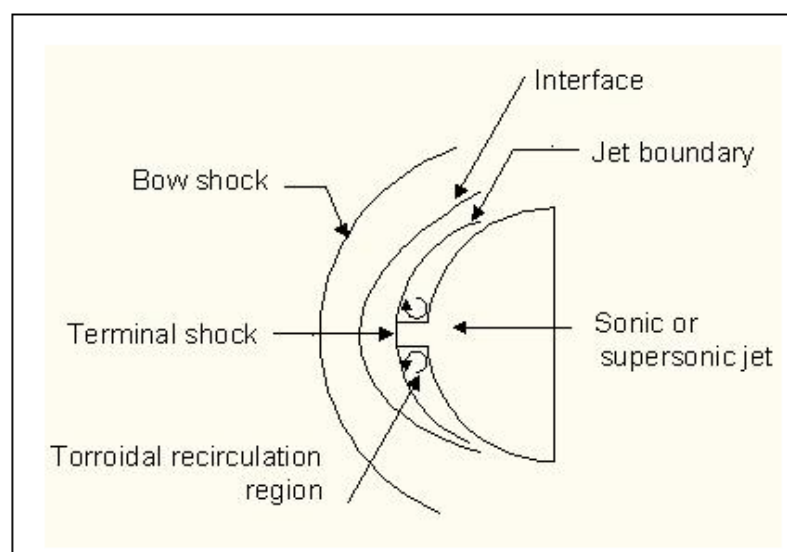


Fig. 40.2. Typical flowfield with counterflow injection based drag reduction technique

40.4. Energy Deposition and Drag Reduction

Localized energy deposition upstream of the configuration of interest is one of the potential options for reduction of wave drag. This novel technique is promising and attracting various researchers to contribute due to the flexibility it offers in terms of implementation. Any suitable source of energy can be used here for flow control by energy deposition. Energy deposition at the suitable location, upstream of the desired space-craft, induces early separation and provides low pressure region ahead the vehicle which leads to drag reduction. The schematic of the flowfield in the presence of upstream energy deposition is shown in **Fig. 40.3**. Basics of energy addition based flow field alteration can be understood using unsteady interaction of blast wave generated due to addition of energy and the shock anchored ahead of the body. This interaction produces a vortex which intern forms a low pressure region on the body. Amount of drag reduction obtainable through energy deposition depends upon location of deposition relative to the vehicle and amount of energy deposited. Therefore this method can be effectively used not only to reduce the wave drag but also to replace the control surfaces and to optimize the performance of scramjet inlet. Substantial amount of contributions have been reported in the open literature about drag reduction using concentrated energy deposition.

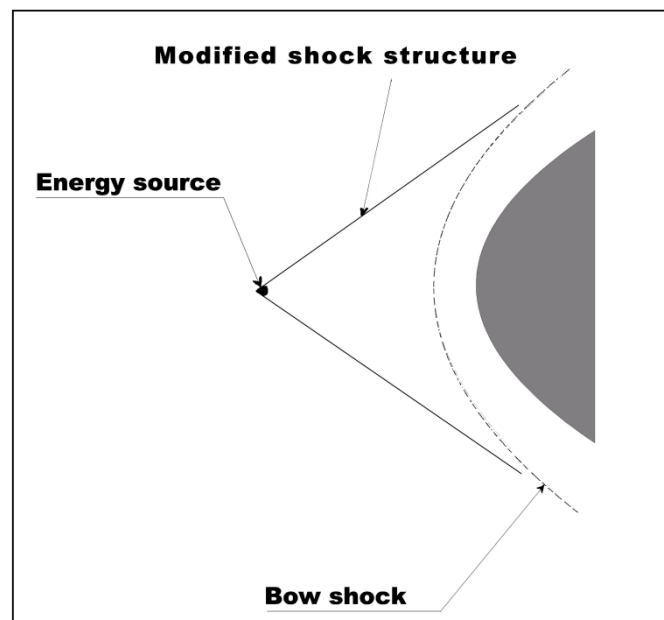


Fig. 40.3. Schematic of expected flowfield of energy addition drag reduction technique

Lecture: 41: Hypersonic flight parameters and stability

41.1 Design Parameters of Flight

Consider a gliding re-entry of a spacecraft as shown in Fig.41.1. Thrust for the vehicle is zero due to the gliding consideration. Hence the forces acting on spacecraft are drag (D), lift (L) and weight (W) or gravity. Let's apply Newton's second for the motion of the vehicle shown in Fig. 41.1.

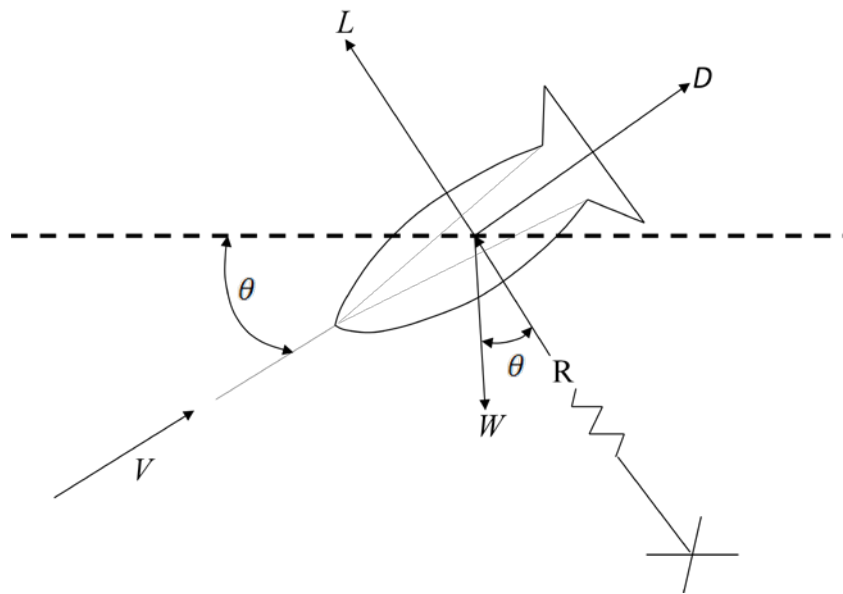


Fig. 41.1. Free-body diagram for a spacecraft [1]

For the forces acting along the axis of the body or along the longitudinal direction in the flight path, we have,

$$W \sin \theta - D = m \frac{dV}{dt}$$

For the forces acting along normal to the axis of the body or along the lateral direction in the flight path, we have,

$$L - W \cos \theta = -m \frac{V^2}{R}$$

Above equations of the motion can be simplified for the actual flight conditions where θ is very small. This approximation leads to $\sin(\theta) \sim 0$ and $\cos(\theta) \sim 1$. Therefore,

$$-D = \frac{W}{g} \frac{dV}{dt}$$

$$L - W = -\frac{W}{g} \frac{V^2}{R}$$

Drag and Lift force can be expressed in terms of non-dimensional coefficients as,

$$C_D = \frac{D}{(1/2)\rho V^2 S} \text{ and } C_L = \frac{L}{(1/2)\rho V^2 S}$$

Here S is the projected or reference area. Hence the equation of motion can be further re-written for axial direction as,

$$-\frac{1}{2}\rho V^2 S C_D = \frac{W}{g} \frac{dV}{dt}$$

$$-\frac{1}{g} \frac{dV}{dt} = \left(\frac{W}{C_D S}\right)^{-1} \frac{\rho V^2}{2} \quad (41.1)$$

The similar expression for the normal direction is as,

$$\frac{1}{2}\rho V^2 S C_L - W = -\frac{W}{g} \frac{V^2}{R}$$

$$1 - \frac{1}{g} \frac{V^2}{R} = \left(\frac{W}{C_L S}\right)^{-1} \frac{\rho V^2}{2} \quad (41.2)$$

These equations (41.1) and (41.2) assert that the parameters $\frac{W}{C_D S}$ and $\frac{W}{C_L S}$ are very important for the estimation of the flight path of a re-entry vehicle. Therefore velocity-altitude map is generally plotted for any flight. Vehicles with higher value of these coefficients encounter higher penetration in the active atmosphere before larger

deceleration. The parameter $\frac{W}{C_D S}$ is found to be important for ballistic re-entry while

$\frac{W}{C_L S}$ turns out to be an important parameter for lifting re-entry.

41.2 Stability and Stability Margin

Force incurred by the aircraft during motion can be considered as a non-uniform distribution of pressure (inviscid) and shear (viscous) forces. The resultant force of all these forces passes through and can be termed as the centre of pressure. The terminology ‘centre of pressure’ can be felt due to the dominance of the pressure force in the presence of shock wave. This resultant force is invariantly expressed in terms of its components. However two systems are followed while expressing the resultant in terms of the components as shown in **Fig. 41.2**. If we consider the body or object or space craft as the basis of the co-ordinate system then the components are called as axial (along the axis of the object) and normal (along the direction, normal to the axis of the object). However, if we consider the trajectory or flight path as the basis of the co-ordinate system then the components, then the resultant is decomposed with respect to the velocity. Thus represented component of force parallel to the velocity is called as drag force and normal to it is called as lift force.

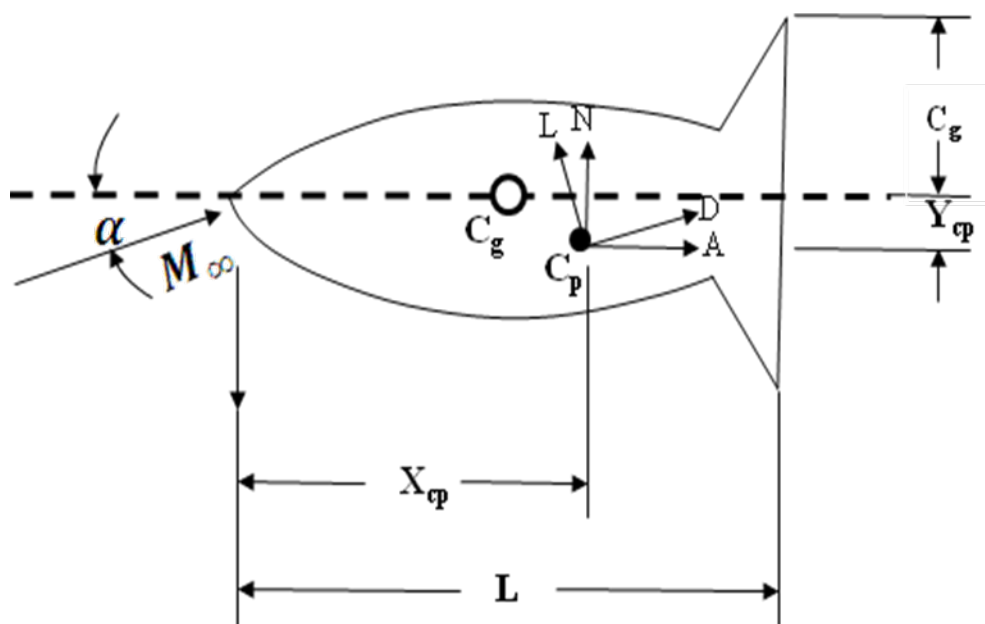


Fig. 41.2. Free-body diagram for a spacecraft in different co-ordinate systems.

Position of the ‘centre of pressure’ or ‘centre of force’ relative to the centre of gravity is an important parameter for stability of the flight and hence for the mission. Static margin of the flight is defined as,

$$\text{Static Margin} = \frac{x_{cp} - x_{cg}}{L}$$

Vehicle is said to be statically stable if centre of pressure and centre of gravity lie at the same distance from the leading edge of the vehicle i.e. $x_{cp} = x_{cg}$. Static margin is invariably kept positive by placing centre of pressure positioned as shown in **Fig.41.2** and is generally 3-5% of the characteristic length (L) of the aircraft. The imposed positivity of the static margin maintains stability by producing a restoring couple since both the components of the axial force provide the pitching moment. In case of increase in angle of attack of the vehicle in the presence of any disturbance, drag force increases. Thus increased drag force leads to increase in the anti-clock wise pitching moment. This increased pitching moment acts as the restoring moment by keep the nose or leading edge of the vehicle down.