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Design and Performance Analysis of Supersonic Inlets Using Computational Fluid Dynamics

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INTRODUCTION

CONE INLET

Although centre-body diffusers and intakes are more describes at high supersonic Mach numbers than a pitot-type intakes on account of the improvement in pressure recovery, they are often prone to violent prone to violent flows oscillations for part or all of the mass flow oscillations is complex but two main types can be recognized. One we shall call the larger oscillation and the other the small or 'organ pipe' oscillations.

According to the explanations offered the large oscillations is determined by the relative positions of the centre-body and cowl, and can be avoided by judicious positioning of one the other. The small oscillation is caused by resonance of air in diffuser and can also be avoided by paying attention to the dimensions of the cowl and centre-body. An explanation of the mechanisms which cause these flow oscillations, and suggestions for avoiding them, will be given in this note. Unfortunately in many cases stable flow over a large range of masses flows can only be obtained by incurring a rise in spillage drag. The conclusions regarding the flow oscillation were reached as a result of series of wind-tunnel tests design solely for the purpose of studying the oscillations. Most of the tests were made at M=1.8 with two- and three-dimensional ducted bodies. Some relevant schlieren photographs of observations of these tests are presented to substantiate the explanations of the flow instability. The photographs were taken with an exposure time of 1/100 sec and oscillations unstable flows show up as blurred images, the only clearly defined shock positions being the limits of the oscillation.

MECHANISMS OF THE LARGE FLOW OSCILLATION

Below figure shows a typical center-body diffuser designed to have the conical shock on the cowl lip at full mass flow for a free-stream Mach number M; there will then be no spillage drag and the flow is quiet stable. As the mass flow through the diffuser is reduced, the flow configuration at the entry has to adjust itself to satisfy the new requirement. Fig below shows such a configuration with air being spilled round the outside of the cowl with a three-shock configuration ahead of the diffuser.

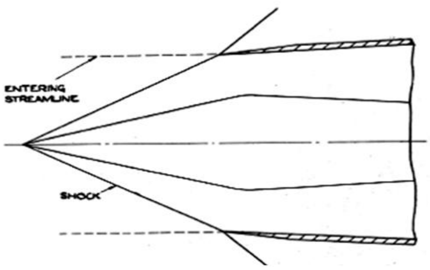

Fig1. Centre-body diffuser Full mass flow

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The flow is modified slightly by the presence of a boundary layer on the Centre-body which will separate some extent depending on the strength of the shock intersecting it and also on the state of the boundary layer itself. As the mass flow is reduced the curved shock moves forward to allow more and more spillage but a point is reached at which the shock can move no further forward.

This is the shock position which corresponds to the detached shock which would from ahead of the cowl if it were a solid body. To make a further reduction in mass flow through the diffuser entry the flow configuration ahead of the entry must now change to some other form.

MECHANISM OF THE SMALL FLOW OSCILLATION

The previous sections have defined the design of a diffuser which will be stable as far as the large oscillation is concerned. Such a diffuser however is still likely to be susceptible to a small oscillation. This small oscillation appears to be caused by resonance of the column of air in the diffuser which vibrates as in an organ pipe causing fluctuations of pressure at the month of the diffuser. This oscillation is illustrated by the schlieren of fig. the diffuser is operating at no mass flow in both cases and the only difference in construction is that the air space between the Centre body and cowl diffuser but the flow oscillations ahead of the other in fig1.4 the point of separation of the from midway along the wedge is clearly defined. In fig1.4 this separation is not visible. In fact the point of separation is oscillations over some length of the wedge and the resultant picture is accordingly blurred. It may be noticed that this oscillation did, but this distinction need not always apply.

The oscillations can be cured by altering the dimensions of the air space in the diffuser either in length or breadth which suggests that the exciting force will only excite a given frequency. The exact nature of the exciting force is not fully understood at present but the following explanation is suggested as being probable. In organ pipes the resonance of the air column is sensitive to the length of the air jet playing on the edge of the pipe, the resonance being produced by the ‘edge tone’ thus produced. In a similar way the edge tone of the diffuser cowl may be excited by the separated boundary layer acting as a jet, or when the vortex sheet from the intersection of the three shock configuration plays on the lip of the cowl. The length of the jet in these causes would be the length of the separated boundary layer or vortex sheet.

Brief tests were made to see if the oscillation could be excited by natural frequency oscillations of the cowl or model supported but in both cases altering the stiffness did not stabilize the flow.

Further tests are required to examine more closely this type of oscillation and the mentioned by which it is excited. In devising these experiments it is likely that the natural frequency of the air in the diffuser will be sensitive to temperature changes and will therefore depend on whether burning is taking place.

RAMP INLET

This paper provides a method of preliminary design for a two dimensional, mixed compression, two ramp supersonic inlets to maximize total pressure recovery and match the mass flow demand of the engine. For an on-design condition, the total pressure recovery is maximized according to the optimization criterion, and the

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dimensions of the inlet in terms of ratios to the engine face diameter are calculated. The optimization criterion is defined such that in a system of (n-1) oblique shocks and one normal shock in two dimensions, the maximum shock pressure recovery is obtained when the shocks are of equal strength. This paper also provides a method to estimate the total pressure recovery for an off-design condition for the specified inlet configuration. For an off- design condition can be estimated. To match the mass flow demand of the engine, the second ramp angle is adjusted and the open/close schedule of a bypass door is determined. The effects of boundary layer are not considered for the supersonic part of the inlet, however friction and expansion losses are considered for the subsonic diffuser.

The inlet is a duct before the engine. Its basic function is to capture a certain amount of air from the free stream and supply it to the engine. Most gas turbine requires the Mach number at engine face at a moderate subsonic speed, to be about Mach 0.4. Therefore, for supersonic aircraft with a gas turbine engine, the inlet will reduce the supersonic free stream to subsonic speed, and provided a matched air mass flow rate to the engine.

The aerodynamic design of a supersonic intake becomes a critical issue to estimate the overall performance of an air-breathing propulsion system which operates at supersonic to hypersonic speeds and captures the incoming air to supply to combustor of main engine after compression. Combined cycle engines have the advantage of having a single flow passage, where compression could be achieved through a series of oblique shocks generated through compression ramps and internal contraction. This leads to formation of series of shock waves and expansion waves inside such intakes. The advantage of such a system is the simple geometry and possibility of adopting variable geometry for efficient operation of engine depending upon the flight operating conditions. A schematic of flow field for a typical combined cycle intake is presented in Fig1.5. At the design condition, the series of compression shocks generated by the ramps gets reflected at the tip of the cowl and leads to further compression inside the intake with the formation of terminal shock at the throat of the intake after passing through a series of shocks. Due to the interaction of shock wave and boundary layer, there exists the possibility of flow separation inside the intake and it is likely to reduce the overall performance of the intake. There also exists the possibility that intake may not start or intake buzz may occur due to possible shock oscillations inside the intake. All these flow phenomena might lead to loss of performance or damage to the structures. To alleviate these problems, attempts are being made by adopting various methods like bleeding, variable geometry, side wall compression, perforations, isolators, length of diffuser etc, to improve the performance of engine. Each of the methods has its own merits and demerits as it involves incorporation of additional system e.g., installation of bleed system or movement and control of system, cooling system, etc, for efficient operation over wide range of operations of intake. Neale and Lamb1-2 demonstrated the effect of various geometrical parameters like ramp angle, side wall, geometry variation, diffuser length, Reynolds number, etc on an intake designed for Mach number 2.2, through extensive and systematic experiments.

The gas turbine engine requires a supply of uniform high total pressure recovery air for good performance and operation, thus the quality of the

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airflow at the engine face will significantly affect the performance of the engine, especially the total pressure loss which affects the engine thrust and consequently the fuel consumption. For1% total pressure loss, the engine will suffer at least 1% thrust loss. Therefore, it is important to maximize the total pressure recovery at the engine face. The total pressure recovery is the ratio of the total pressure of the airflow at the engine face to that of the free stream.

A supersonic inlet has two parts. The supersonic diffuser for supersonic and the subsonic diffuser, it is long and heavy. The designer needs to know the size of the inlet in order to properly account for it during the conceptual and preliminary design stage. The designer also needs to estimate the total pressure recovery at the engine face in order to estimate the performance of the engine and the whole aircraft. Therefore, a method is needed to estimate the size of the supersonic inlet and the total pressure recovery in the early design stages. This paper provides such a method for a 2D supersonic inlet.

PROPOSED METHOD AND IMPLEMENTATION

COMPUTATIONAL METHODOLOGY

The computations are performed using commercial software CFX which adopts finite volume approach to solve compressible Reynolds Averaged Navier Stokes equations with standard turbulence models. Present computations have been made adopting k-w turbulence model. The standard k-w model in CFX is based on the Wilcox k-w model, which is designed to be applied throughout the boundary layer and is applicable to wall bounded flows as well as free-shear flows. “k-w” turbulent simulations over air intake reported by Reinartz8, et. al. and Coratekin13, et al. gave a good comparison with experimental results at supersonic Mach numbers. In the present tests, compressibility corrections were applied and the default model constants were set. Explicit coupled solver with upwind discretization scheme for flow and transport equations was adopted. For faster convergence, 4-stage multi grid was used. The computational domain was restricted to the internal duct section enclosed by ramp surface and the cowl internal surface only with appropriate boundary conditions to reduce the computational time. Boundary conditions at inlet boundary were specified by stagnation and static pressures corresponding to supersonic flow of Mach 2.2 with a small turbulent intensity and viscosity ratio. At the exit, pressure outlet boundary condition was assigned. For supersonic outflow, the variables were extrapolated from the interior cells and for subsonic outflow, a back pressure was enforced. No-slip boundary conditions were enforced at all the solid walls. Computations were made for free flow (I.e., no back pressure) and with a back pressure specified by appropriate subsonic out flow condition.

The preliminary design of the inlet is divided into the following five subtasks. The first subtask is the selection of the inlet configuration including selection of the cross sectional shape of the supersonic part, selection of compression method, selection of the number of ramps or oblique shocks, and selection of the subsonic diffuser. The second subtask is the determination of the optimization criteria to maximize the total pressure recovery. Third is the method to design the inlet according to the on-design conditions, including the estimation of the total pressure recovery and geometric sizes of the inlet. The fourth subtask compares the optimum on design

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result with the experimental data and CFD simulation result. The final subtask is to estimate the total pressure recovery of the inlet under off design conditions. The following section outlines the five subtasks, completion of which provides the preliminary design of a 2D supersonic inlet and information on the size of the inlet and information on the size of the inlet, the maximum total pressure recovery of the on design condition, and the estimations of total pressure recovery of off-design conditions.

SELECTION OF INLET CONFIGURATION

For a supersonic inlet, the free stream is decelerated to the subsonic engine face entry speed through a suitable shock system and a subsonic diffuser. The shock system will decelerate the flow to a subsonic number, and the diffuser will further reduce the flow speed to the engine face entry speed. The design criterion is to maximize total pressure recovery. At a flight Mach number of 2.2, a practical number of 95% total pressure recovery is desired for a long duration cruise transportation aircraft.

During design of this supersonic inlet configuration, several sections or tradeoffs have to be made, including selection of the cross section shape of the supersonic part, selection of number of ramp or oblique shocks, and selection of the subsonic diffuser.

The cross section of the supersonic diffuser can be annular or rectangular. In general, an annular supersonic diffuser will have higher total pressure recovery as long as the free stream flow is aligned with the centre body axis. However, if the direction of the flow is at an angle to the axis, this type of diffuser will be more likely to have such flow distortion that the engine compressor may operate close to surge line. In contrast, the two-dimensional supersonic diffuser is much more insensitive to non-symmetric flow. Also, a two-dimensional supersonic diffuser is much more insensitive to non-symmetric flow. Also, a two-dimensional supersonic diffuser can provide a larger variation in inlet flow and is obviously much simpler in design. For transportation aircraft, safety is the paramount consideration, therefore, the two-dimensional cross section shape is selected for the supersonic diffuser.

Compression selection was based on the idea that for supersonic Mach numbers up to 1.4-1.6,a pitot type inlet with a normal shock is considered as the best choice considering tradeoffs between total pressure recovery, inlet length and inlet weight. At high flight Mach numbers up to approximately 2.0, an external compression multi-ramp system is usually the best choice, again considering total pressure recovery, length, and weight. With an external compression inlet, there will be one or more oblique shocks followed by normal shocks, which remains outside of the cowl lip. Finally, for flight Mach numbers above 2.0, a mixed compression multi-ramp system is considered the best choice.

A mixed compression method compression method has a combination of external and internal oblique shocks followed a normal shock at the inlet throat. Figure 1 shows three inlets a, b, and c designed to Mach 2.2 with the same pressure recovery, but using different compression methods: inlet A, all external; inlet b, 2 external oblique shocks,1 internal oblique shock, and 1 internal normal shock; inlet c, 1 external oblique shock, 2 internal oblique shocks, and 1 internal normal shock. Although the three inlets in fig below have the same pressure recovery, they are different in the following aspects:

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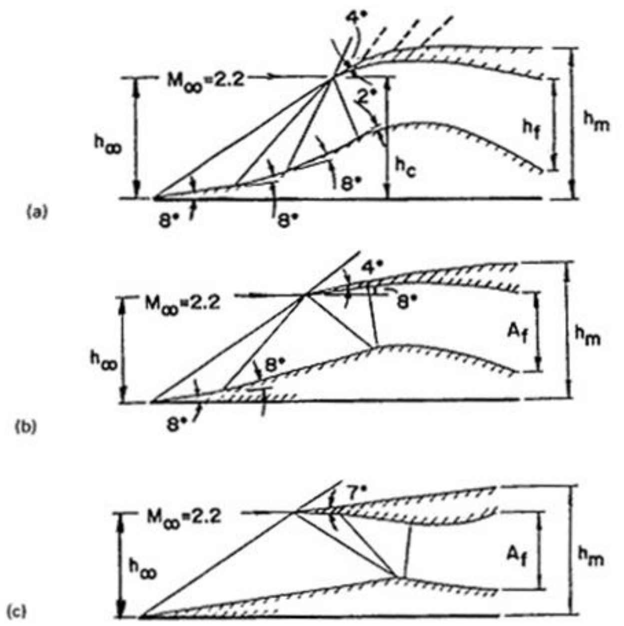

Fig1.5 types of compression

(1) Self-starting: inlet A is self-starting because of the external shock, neither B nor C is self- starting both b and c have to use movable ramps in order to establish the design shock system, and C is more difficult than B to establish the design shock system.

(2) Weight: as the degree of internal compression increases, the supersonic section of the inlet becomes longer, and hence heavier.

(3) Boundary layer effects: increased enclosure of shocks will make the boundary layer effect more severe.

(4) External drag: as the degree of internal compression increases, the external line to the cowl is finer and hence the external wave drag is less.

The function of the subsonic diffuser is to further reduce the flow speed after the normal shock to a lower subsonic Mach number at the engine face. Given the diffuser entry Mach number and the engine face speed, the geometric factors of the diffuser are mainly affected by the duct expansion angle. Here the geometric factors include the area ratio of engine face area to entry throat area, and length. While the supersonic section of the inlet is two-dimensional in shape, there is a transition from two –dimensional to circular on the subsonic section of the inlet is two-dimensional in shape, there is a transition from two-dimensional to circular in the subsonic diffuser. Also, a constant area region is needed to prevent boundary layer separation at the entrance of the subsonic diffuser. Integrating the above design decisions together, a sketch of the whole inlet system is given below in fig. the double lines originating from point 1, 2, 3 and 4 represent oblique or normal shocks.

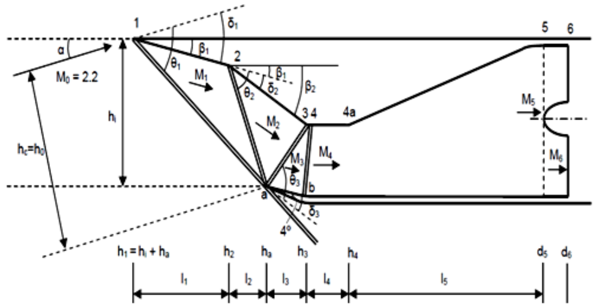

Fig1.6 schematic diagram of ramp inlet

THE OPTIMIZATION CREITERON TO MAXIIZE TOTAL PRESSURE RECOVERY

Instead of using an optimizer with an iterative procedure, an optimization criterion is used to determine the ramp angles of the oblique shocks for maximum pressure recovery of the supersonic section. The advantage of using optimization criteria is that it is faster, more accurate, and there is no need of an optimizer. The optimization criterion is proposed by Oswatitsch and is described as follows. The maximum shock pressure recovery is obtained when the shocks are of equal strength, i.e. The Mach numbers perpendicular to the individual shocks are equal.

$$M_1 \sin\theta_1 = M_2 sin = M_{n-1} sin\theta_{n-1} \quad (1)$$

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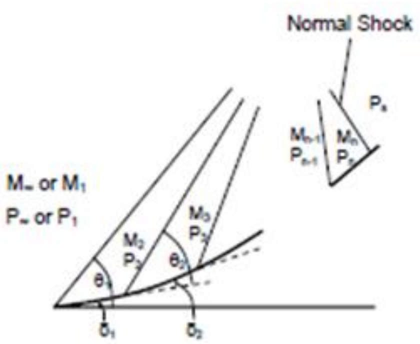

Fig1.7 .Multi shock compression for optimization

DESIGN OF THE INLET ACCORDING TO ON-DESIGN CONDITIONS

Given the free stream mach number M_0, angle of attack α, flight altitude H, the normal shock up-stream Mach numberM_4_up,the engine face hub-tip ratio h_t, the fan face entry Mach number M_6, and the ratio of supersonic diffuser width to engine face diameter w_d_6, the goal of the inlet system design is to determine ratios of lengths to fan face diameter l_1_d_6 – l_5_d_6, and ratios of heights to fan face diameter h_1_d_6 – h_5_d_6. The Mach number M_4_up is given in order to shock the system; otherwise the number of unknown variables is more than the number of equations.

Because higher M_4_up results in lower Mach number M_4 after the terminal normal shock and thus higher total pressure loss across, and lower m_4 results in lower total pressure loss in the subsonic diffuser, there is a value of m_4_up that will result in maximum total pressure recovery. This value of m_4_up is found to be about 1.265 for free stream 2.2 in the example given later.

SOLUTION OF THE MACH NUMBER AT DIFFERENT POSITIONS

The inlet is to be designed at the cruise conditions of flight Mach number 2.2 and flight altitude 55,000 ft. at the on design point, the oblique shock waves from the two external ramps intersect at the cowl leading edge, and the third oblique shock reflects upward to intersect the junction of the final ramp and the throat section. This is shown in above figure

$$M_1^2 = \frac{(\gamma+1)^2 M_0{}^4 sin^2\theta_1 - 4(M_0{}^2 sin^2\theta_1 - 1)(\gamma M_0{}^2 sin^2\theta_1 + 1)}{[2\gamma M_0{}^2 sin^2\theta_1 - (\gamma+1)][(\gamma-1)M_0{}^2 sin^2\theta_1 + 2]} \quad (2)$$

$$tan\delta_1 = \frac{2cot\theta_1 (M_0^2 sin^2\theta_1 - 1)}{2 + M_0^2(\gamma + 1 - 2sin^2\theta_1)} \quad (3)$$

$$M_2^2 = \frac{(\gamma+1)^2 M_1{}^4 sin^2\theta_2 - 4(M_1{}^2 sin^2\theta_2 - 1)(\gamma M_1{}^2 sin^2\theta_2 + 1)}{[2\gamma M_1{}^2 sin^2\theta_2 - (\gamma+1)][(\gamma-1)M_0{}^2 sin^2\theta_2 + 2]} \quad (4)$$

$$tan\delta_2 = \frac{2cot\theta_2 (M_1^2 sin^2\theta_2 - 1)}{2 + M_1^2(\gamma + 1 - 2sin^2\theta_2)} \quad (5)$$

$$M_3^2 = \frac{(\gamma+1)^2 M_2{}^4 sin^2\theta_3 - 4(M_2{}^2 sin^2\theta_3 - 1)(\gamma M_2{}^2 sin^2\theta_3 + 1)}{[2\gamma M_2{}^2 sin^2\theta_3 - (\gamma+1)][(\gamma-1)M_1{}^2 sin^2\theta_3 + 2]} \quad (6)$$

$$tan\delta_3 = \frac{2cot\theta_3 (M_2^2 sin^2\theta_3 - 1)}{2 + M_2^2(\gamma + 1 - 2sin^2\theta_3)} \quad (7)$$

The portion of the form Station point 4 to 4a is the transition zone that ensures the reattachment of the boundary layer after the normal shock. According to ref, the slope of this zone should be zero, the cross section area of this zone should be constant and the length is selected to be 2 times the height of this zone...

From station point 4a to 5, the cross section of the duct transits from a rectangular to a circle and expands according to ref the expansion angle 2α should be 6-12 degrees to have short length.

CONE PROFILE AND ITS DESIGN CONSIDERATIONS

Simulated Wind tunnel modeling: The present work is a wind tunnel modeling in ICEM CFD. A rectangular domain is designed to simulate on time conditions prevail at supersonic flow regimes. In ICEM CFD the geometry model is drafted and meshed. Now this model is an assemblage of elements, which is interconnected at nodal points or nodes and together represents the original body. The meshed model should be meshed with proper selection of mesh area, mesh sizes and mesh type. The mesh type we used I this work is TETRAHEDRAL type: which means

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there are only tetrahedral elements on the mesh. The tetrahedral elements give uniformity the structure and hence the distribution of forces is easy. The model is checked to ensure connectivity of adjacent members/elements in assemblage.

The meshed geometry is exported to CFX-PRE and on time conditions like temperature pressure and velocity conditions are assigned to the meshed geometry model. Later the flow conditions assigned meshed cone is simulated in CFX-POST to obtain results.

PARTS OF WIND TUNNEL MODEL

Domain inlet: It is the inlet for wind tunnel incoming supersonic flow conditions are assigned in this part.
Domain outlet: it is the outlet part of the wind tunnel.

Domain side walls: these are the side walls of the wind tunnel for which free slip conditions are assigned.

Center cone: This is the leading edge in the supersonic inlet which first meets the relative wind.

Inner cylinder: This is attached with outer cylinder multiple shock reflections and total pressure is obtained in this section.

Outer lip: Outer lip connects inner cylinder and outer cylinder it is designed to deviate expansion waves when subjected to supersonic flow.

Outer cylinder: Outer cylinder connects inner cylinder and outer lip generally outer cylinder is the outer casing of the supersonic inlets it is attached to the pylon of the aircraft.

MESHING

DESCRIPTION OF FINITE ELEMENT METHOD

Finite element method is a powerful numerical discretization technique for approximate solution of continuum mechanics problems or complex problems. Analytical method of solving many engineering problems is not possible as it involves mathematical expressions that give the value of desired unknown quantity at any location in a body and as a result the solution can be obtained only for certain simplified situations for example problems involving complex structures and boundary conditions, numerical methods provide approximate but acceptable solutions.

In numerical solutions, a body or a structure is divided into an equivalent system of smaller bodies or units with finite degrees of freedom, a process that is termed as discretization. The assemblage of such units represents the original body. Each unit is called element and these elements are interconnected at joints, which are called nodes or nodal points. Instead of solving the problem for the entire body in one operation, an approximate admissible solution is constructed for each element and the solution continuity is maintained at the inter element boundaries. The properties of elements are formulated and combined to obtain the solution for the entire structure. This forms the basis of finite element method.

RESULTS FOR CONE INLET

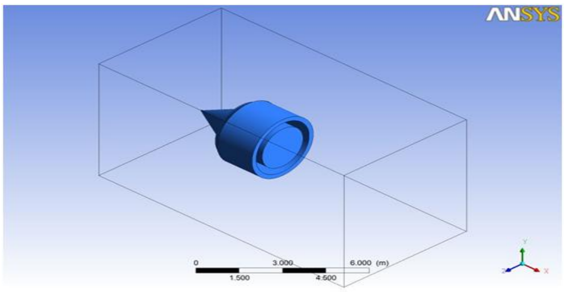

Fig3.1 cone inlet isometric view

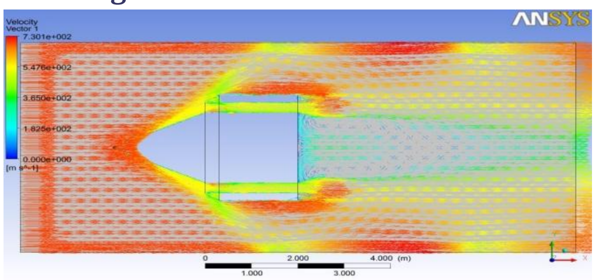

Fig3.2 cone inlet velocity vector plot

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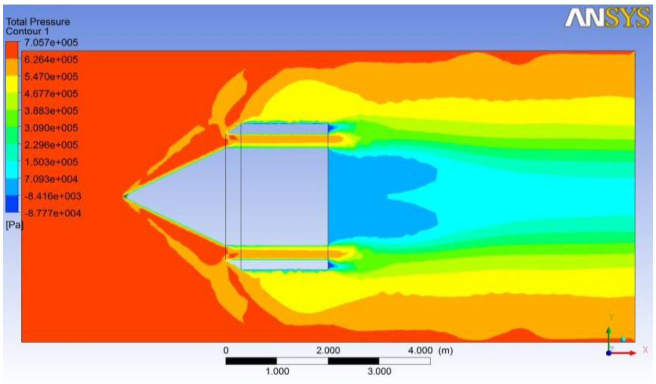

Fig3.3 cone inlet total pressure contour

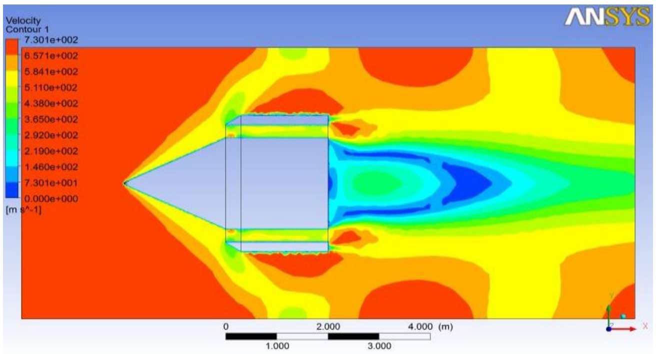

Fig 3.4 cone inlet velocity contour

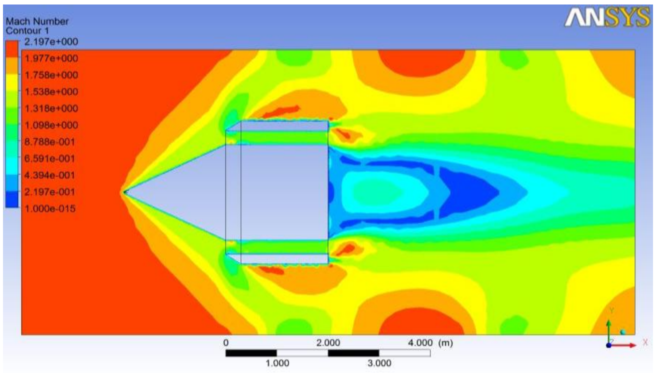

Fig 3.5 cone inlet Mach number contour

RESULTS FOR RAMP INLET

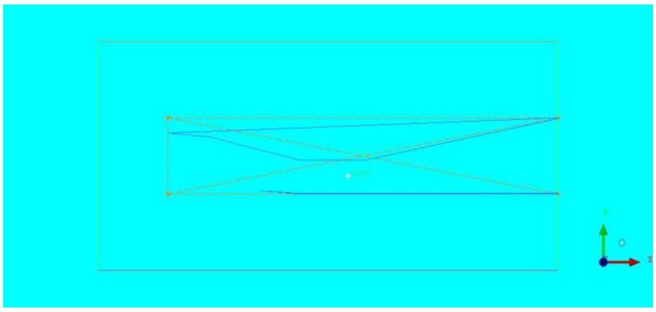

Fig 3.6 ramp inlet geometry

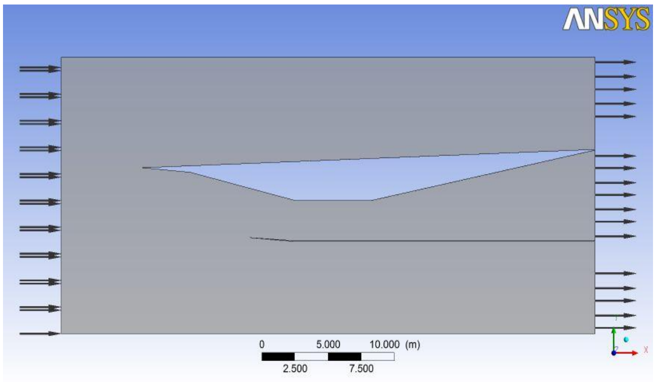

Fig 3.7 ramp inlet CFX-PRE

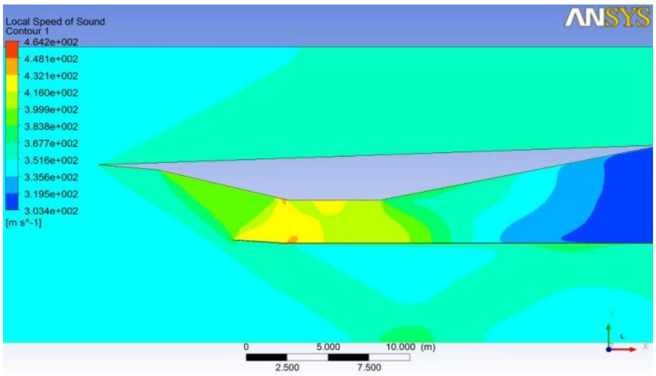

Fig 3.8 ramp inlet local speed of sound

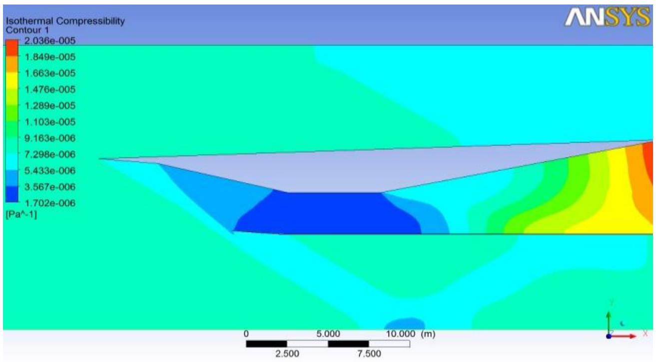

Fig 3.9 ramp inlet isothermal compressibility

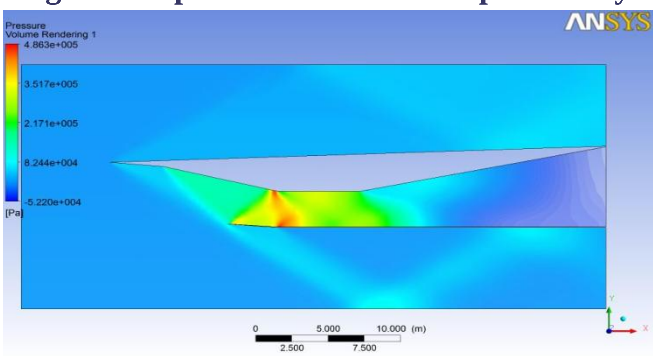

Fig 3.10 ramp inlet pressure volume rendering

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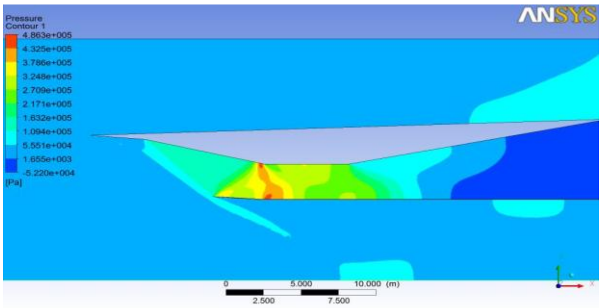

Fig3.11 ramp inlet pressure contour

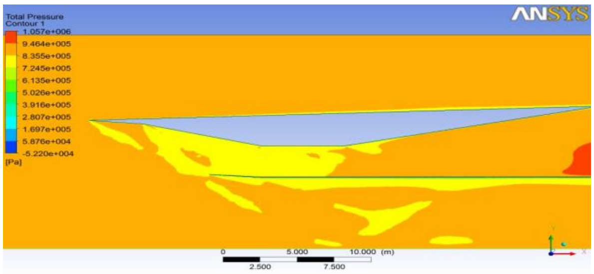

Fig3.12 ramp inlet total pressure

CONCLUSIONS

CONICAL INLET:

This report has offered an explanation of the large flow oscillations which can occur with Centre body diffusers at supersonic speeds and the results of some wind tunnel tests have been given in support.

With an understanding of the mechanism of the large flows oscillation it is possible to design a diffuser which will be stable or in which the limits of stable operation can be found. Although a diffuser may be stable as regards the large oscillation it may still be vulnerable to a small oscillation. The mechanism of this oscillation is not fully known but methods of avoiding it in practical cases are suggested.

RAMP INLET:

At high flight Mach numbers up to approximately 2.0, an external compression multi-ramp system is usually the best choice, again considering total pressure recovery, length, and weight. With an external compression inlet, there will be one or more oblique shocks followed by a normal shock, which remains outside of the cowl lip. Finally, for flight Mach numbers above 2.0, a mixed compression multi-ramp system is considered the best choice. A mixed compression method has a combination of external and internal oblique shocks followed by a normal shock at the inlet throat.

The maximum pressure recovery is obtained just after the aft position of the acceleration chamber. Due to the presence of normal shock at the station 4 the flow in the acceleration chamber is subsonic speeds we need to accelerate the flow to velocity range of 0.5-0.8 this results were achieved at the aft position of the throat section. The overall length of the supersonic diffuser is decreased. Thus the maximum pressure recovery is obtained at the aft of the throat section.

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