Study of Flow optimization in Mixed Compression Inlet with Cowl Deflection

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Abstract - *The supersonic inlet are the structures in supersonic power station systems which is able to reduce the supersonic flow to subsonic flow. The ramps designed for particular mach numbers are employed to get the specified flow. The proposed the numerical studies are conducted to ascertain an optimum design of a supersonic intake. The most goal of optimization was to maximization of the entire pressure recovery, and to attenuate the shock Interactions. For the planning conditions, the entire pressure recovery is maximized in step with the optimization criterion, and therefore, the dimensions of the inlet in terms of ratios to the engine face diameter are calculated. The calculated dimensions are 2-D modelled in Solid works 2016 and Further the model is meshed and can be analyzed in Ansys 18.1. The intake was designed for an oblong cross section, mixed compression and has three oblique shock waves, and a terminal normal shock followed by a transitioning diffuser with circular cross section at the exit plane, which provides a subsonic flow to the engine compressor. The turbine engine requires a supply of uniform high total pressure recovery air permanently performance and operation, thus the standard of the airflow at the engine face will significantly affect the performance of the engine, especially the entire pressure loss which affects the engine thrust and consequently the fuel consumption. For 1% total pressure loss, the engine will suffer a minimum of 1% thrust loss. Therefore, it's important to maximize the entire pressure recovery at the engine face.*

***Key Words*: Supersonic inlet, Ramp, Oblique shock, Shock interaction.**

1. INTRODUCTION

On a supersonic airliner a variable geometry engine air intake has to be employed to optimize performance throughout the flight envelope. This geometry has to be accurately controlled to give maximum fuel economy and it also needs a rapid response to atmospheric and engine transients. An intake ramp is designed to generate a number of shocks waves to aid the inlet compression process at supersonic speeds. The ramp sits at acute angle to deflect the intake air from the longitudinal direction. At supersonic flight speeds, the deflection of the air stream creates a number of oblique shock waves at each change of gradient along at the ramp. Air crossing each shock wave suddenly slows to a lower Mach number, thus increasing pressure. The interaction of shocks is analyzed during three condition such as constant ramped position, deflected upper and lower ramped position in order to reduce the formation of shocks by employing cowl deflection, hence the structural damage can be optimized. A typical supersonic intake has two sections, the first one is a supersonic diffuser section and other one is a subsonic diffuser section. The supersonic diffuser section compresses air from a supersonic state to the sonic state, and then subsonic diffuser compresses it to a moderate subsonic speed. Such compressed air is then supplied to the engine compressor. Total pressure recovery is the ratio of the average total pressure at the exit of the inlet to the free stream total pressure. A higher-pressure recovery indicates a better performance inlet with the advancement in modern CFD techniques it has become quite easy to design, analyze and optimize everything. Out of all the components involved in supersonic flight, engine intake is the most important one whose operation affects the overall performance of the engine to the great extent. The basic use of it, simply being to slow down the incoming flow or increasing the pressure of the flow to make it eligible for combustion. Simple as it sounds, it is done by a neat arrangement of shockwaves which are formed by the supersonic flow interacting with the ramp and the cowl tip. The shockwaves further keep reflecting after interacting with the cowl tip to form a final normal shock downstream in the back-pressure condition. When the incoming flow interacts with these shockwaves, the flow properties change abruptly. The changes which we are concentrating on are, increase on pressure and decrease in velocity, which is exactly what is needed for a good combustion.

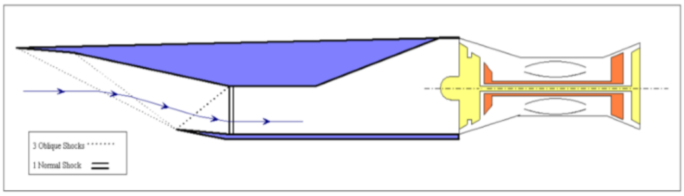

Fig.1 Mixed compression inlet with engine

1.1 Theoretical Study 1

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.

FORMULAS USED

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

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

γ = Specific heat ratio, M_o = Inlet Mach number, θ_1 = 1st Wave angle, M_1=Mach number after the 1st shock wave, δ_1=wedge angle with flow direction angle

$$Tan\theta = 2 * cot\beta * \frac{M_1^2 * Sin^2\beta - 1}{M^2*(\gamma+cos2\beta+2)}$$

$$M_{n_1} = M_1 \sin\beta \quad M_2 = \frac{M_{n_2}}{\sin(\beta-\theta)}$$

CONDITION 1 (+3 Degree Ramp Angle)

From the below equation:
For Mach 2.2

1st Point of Interaction

$\theta = 10^0$ M = 2.2 β = 35.7854 M_2=?

$M_{n_1} = M_1 \sin\beta$ = 2.2 × sin (35.7854) = 1.2864

M_{n_2} = 0.7929

$$M_2 = \frac{M_{n_2}}{\sin(\beta-\theta)} = \frac{0.7929}{\sin(35.7854-10)} = 1.8228$$

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)\times M_1^2 \times (sin^2\beta)}{2+(\gamma-1)\times M_1^2\times(sin^2\beta)} =$$

$$\frac{(1.4+1)\times 2.2^2 \times (sin^2 35.7854)}{2+(1.4-1)\times 2.2^2 \times (sin^2 35.7854)} = 1.4920$$

$$\frac{p_2}{p_1} = \frac{2\times\gamma\times M_1^2\times(sin^2\beta)-(\gamma-1)}{(\gamma+1)} =$$

$$\frac{2\times1.4\times2.2^2\times(sin^2 35.7854)-(1.4-1)}{(1.4+1)} = 1.7641$$

$$\frac{T_2}{T_1} = \frac{p_2}{p_1}\times\frac{\rho_1}{\rho_2} = 1.1823$$

$$\frac{p_{02}}{p_{01}} = \frac{p_2}{p_1}\left(\frac{1+\frac{\gamma-1}{2}M_2^2}{1+\frac{\gamma-1}{2}M_1^2}\right)^{\frac{\gamma}{(\gamma-1)}} =$$

$$1.7641\left(\frac{1+\frac{1.4-1}{2}1.8228^2}{1+\frac{1.4-1}{2}2.2^2}\right)^{\frac{1.4}{(1.4-1)}} = 0.9816$$

2nd Point of Interaction

$\theta = 11.16^0$ M = 1.8228

β = 44.8885 To find = M_3

$M_{n_2} = M_2 \sin\beta$ = 1.8228 × sin(44.8885) = 1.2864

M_{n_3} = 0.7929

$$M_3 = \frac{M_{n_3}}{\sin(\beta-\theta)} = \frac{0.7929}{\sin(44.8885-11.16)} = 1.4280$$

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)\times M_1^2 \times (sin^2\beta)}{2+(\gamma-1)\times M_1^2\times(sin^2\beta)} =$$

$$\frac{(1.4+1)\times 1.8228^2 \times (sin^2 44.8885)}{2+(1.4-1)\times 1.8228^2 \times (sin^2 44.8885)} = 1.4920$$

$$\frac{p_2}{p_1} = \frac{2\times\gamma\times M_1^2\times(sin^2\beta)-(\gamma-1)}{(\gamma+1)} =$$

$$\frac{2\times1.4\times1.8228^2\times(sin^2 44.8885)-(1.4-1)}{(1.4+1)} = 1.7639$$

$$\frac{T_2}{T_1} = \frac{p_2}{p_1}\times\frac{\rho_1}{\rho_2} = 1.1822$$

$$\frac{p_{02}}{p_{01}} = \frac{p_2}{p_1}\left(\frac{1+\frac{\gamma-1}{2}M_3{}^2}{1+\frac{\gamma-1}{2}M_2{}^2}\right)^{\frac{\gamma}{(\gamma-1)}} =$$

$$1.7639\left(\frac{1+\frac{1.4-1}{2}1.4280^2}{1+\frac{1.4-1}{2}1.8228^2}\right)^{\frac{1.4}{(1.4-1)}} = \mathbf{0.9816}$$

3rd Point of Interaction

θ = 10.28⁰ M = 1.43

No shock at this condition

4th Point of Interaction

θ = 10.28⁰ M = 1.43

No shock at this condition

Pressure Recovery

C-1 * C-2 * C-3 * C4

$$\frac{p_{02}}{p_{01}} * \frac{p_{02}}{p_{01}} * \frac{p_{02}}{p_{01}} * \frac{p_{02}}{p_{01}} = \text{Total pressure recovery}$$

0.9816 * 0.9816 * 1 * 1 = 0.9636

1.2 Calculation for Inlet Dimension at Subsonic Diffuser

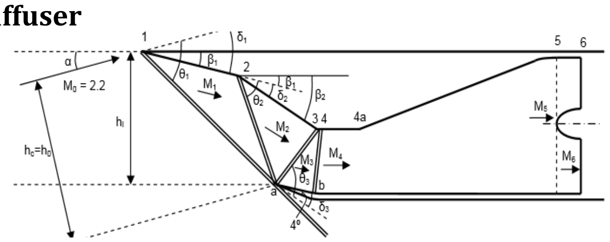

Fig 1. Sketch of the whole inlet system with shock positions M_o = Inlet Mach number, β= Wedge angle, α = Angle of attack, θ_1 = 1st Wave angle , δ_1=wedge angle with flow direction angle, M_1=Mach number after the 1st shock wave, θ_2 = 2nd Wave angle , δ_2= 2nd Wedge angle, M_2 = Mach number at position 2, M_3 = Mach number at position 3, θ_3 = 3rd wave angle

Assume the distance between station point 3 and 4 is very small and can be ignored.

- The portion of the inlet from station point 4 to 4a is the transition zone that ensures the reattachment of the boundary layer after the normal shock, 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 rectangle to a circle and expands, the expansion angle dθ should be 6 to 12 degrees in order to obtain a high total pressure ratio.
- By Assuming the duct diameter is constant from station 5 to 6, we have the following relation for the airflow areas.

$$\frac{A_5}{A_6} = \frac{1}{1-h\ t^2} \quad \frac{A_5}{A_6} = \textbf{Area ratio}$$

h_t = **Hub to tip ratios**

From the area-Mach number relation we have

$$\left(\frac{A_6}{A^*}\right)^2 = \frac{1}{M_6{}^2}\left(\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M_6{}^2\right)\right)^{\frac{\gamma+1}{\gamma-1}}$$

$$\left(\frac{A_6}{A^*}\right)^2 =$$

$$\frac{1}{0.395^2}\left(\frac{2}{1.4+1}\left(1+\frac{1.4-1}{2}0.395^2\right)\right)^{\frac{1.4+1}{1.4-1}}$$

M_6= Mach number at position 6

$$\frac{A_6}{A^*} = \mathbf{1.4093} \qquad \frac{A_5}{A_6} = \frac{1}{1-0.3204}$$

A_5 =cross- sectional area at position 5

A_6 =cross- sectional area at position 6

A^*= The cross-section area of flow tube at throat where the flow is sonic

$$\left(\frac{A_5}{A^*}\right)^2 = \frac{1}{M_5{}^2}\left(\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M_5{}^2\right)\right)^{\frac{\gamma+1}{\gamma-1}}$$

$(1.409 * 1.1144)^2 =$

$$\frac{1}{M_5^2}\left(\frac{2}{1.4+1}\left(1+\frac{1.4-1}{2}M_5^2\right)\right)^{\frac{1.4+1}{1.4-1}}$$

$$M_5 = \mathbf{0.3123}$$

Statics	
Nodes	30258
Elements	29633

2. DESIGN AND ANALYSIS

2-D Geometry

The geometry has been made for 3 cases of analysis that is varied according to the mach no. and varying ramp angle during the operating conditions. The geometry specifications are as follows

The 1st geometry for 3° ramp angle for varying Mach no. 1.7, 2.2 & 2.8.

The 2nd geometry for 0° ramp angle for varying Mach no. 1.7, 2.2 & 2.8.

The 3rd geometry for -3° ramp angle for varying Mach no. 1.7, 2.2 & 2.8.

Contition-1 (3° Ramp angle)

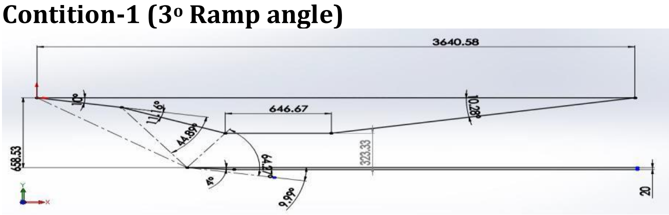

Fig 2. The 2-D design of the inlet for 3° ramp angle

3. MESHING

The required mesh size is defined and various steps are performed to produce a mesh of high quality, using the commands of Edge Sizing, Face Sizing and Inflation mesh grids near and around the ramp. High aspect-ratio elements are generated at near-wall areas to divide these regions efficiently and capture the boundary layer and recirculation zone accurately. The whole domain is divided into different sections so as to obtain differential mesh quality. That is achieved using Face split.

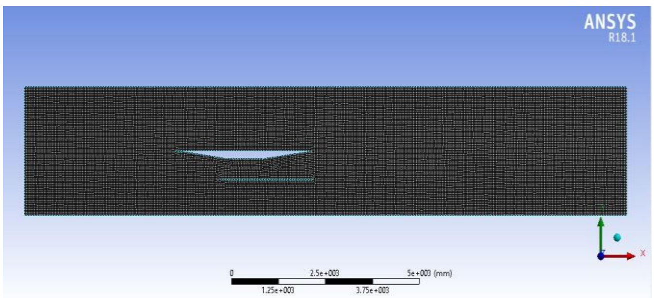

Fig 3. structured fine meshed domain

4. BOUNDARY CONDITIONS.

The analysis for a viscous case is done for 3 ramp angles from-3° degrees to +3° degrees as,-3°, 0° & +3° as well as for the Mach numbers of 1.7, 2.2 & 2.8.to start with, the case of 1.7 Mach for a -3°degree ramp angle is simulated under turbulence models of k-ω SST to determine which gives the best and most realistic results.

Domain	Air, Density-Based
Model	Viscous- k-ω SST
Energy equation	On
Material Density Coefficient Pressure Thermal Viscosity	Air Ideal Gas Piecewise Polynomial Kinetic Theory Sutherland
Boundary Condition	**Inlet**: Pressure inlet Total gauge Pressure: 717622 Pa Supersonic Pressure: 26436 Pa Temperature: 573 K **Pressure far field:** Mach Number: 2.2 Static gauge Pressure: 26436 Pa Temperature:225.13 K **Outlet**: Pressure Outlet Outlet Gauge Pressure: 26436 Pa Wall: No-Slip
olution Method Flux Gradient Flow Solution Initialization	Implicit AUSM Least square cell based Second order upwind Standard Initialization

3. RESULTS AND DISCUSSION

CFD analysis has been carried out for the 2-D case. Results of each case have been studied and plotted. They will further be discussed in detail.

VISCOUS MODEL - k-ω SST MODEL

The velocity and Static Pressure variation are captured under the k-ω SST model as follows.

Ramp Angle 3° for Mach No. 2.2

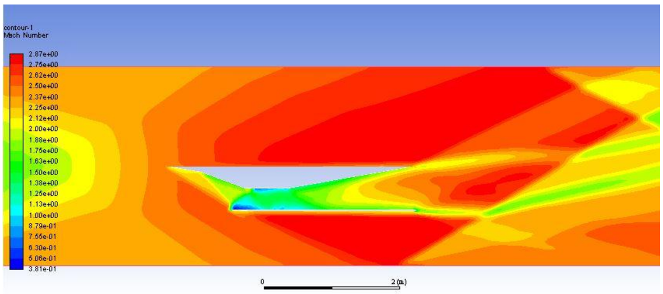

Fig 4. Mach number contour of 2.2 Viscous 3°degree ramp angle (k-ω SST)

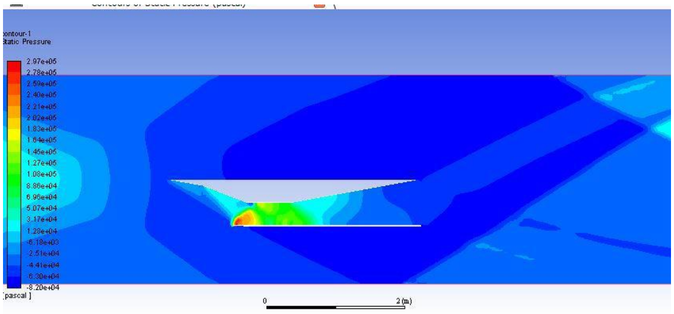

Fig 5. Contour of Co-efficient of Static Pressure for Mach 2.2 for 3°degree ramp angle Viscous (k-ω SST)

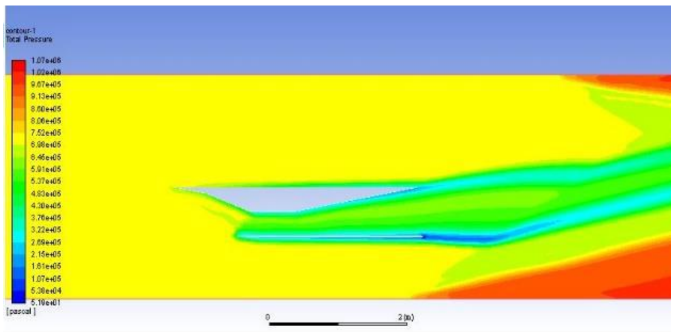

Fig 6. Contour of Co-efficient of Total Pressure for Mach 2.2.for 3°degree ramp angle Viscous (k-ω SST)

4. COMPARISSION of THEORETICAL & CFD ANALYSIS

The CFD Analysis values are plotted with respect to Mach No versus pressure recovery values. The values are tabulated for varying ramp angles and Mach no. such as ramp angles of -3°, 0° & 3°. The varying Mach no. of 1.7, 2.2 & 2.8.

CONDITION 1 (Ramp Angle 0°)

Sl No.	Mach No.	Pressure Recovery $\frac{P_{02}}{P0_1}$ Theoretical	Pressure Recovery $\frac{P_{02}}{P_{01}}$ Analysis
01.	1.7	0.9899	0.9858
02.	2.2	0.9653	0.9723
03.	2.8	0.9281	0.9439

CONDITION 2 (Ramp Angle 3°)

Sl No.	Mach No.	Pressure Recovery $\frac{P_{02}}{P_{01}}$ Theoretical	Pressure Recovery $\frac{P_{02}}{P0_1}$ Analysis
01.	1.7	0.9874	0.9754
02.	2.2	0.9636	0.9534
03.	2.8	0.9139	0.8839

CONDITION 3 (Ramp Angle -3°)

Sl No.	Mach No.	Pressure Recovery $\frac{P_{02}}{P_{01}}$ Theoretical	Pressure Recovery $\frac{P_{02}}{P_{01}}$ Experimental
01.	1.7	0.9913	0.9893
02.	2.2	0.9842	0.9799
03.	2.8	0.9751	0.9563

VALIDATION OF RESULTS

The obtained results are validated with the cfd simulation results. It is achieved with the help of the pressure ratio of Wall Static Pressure to the Free Stream Static Pressure along the upper surface of the ramp. It is observed that in their circulation zone, the pressure ratios from k-ω SST model seem to match more to the experimental results.

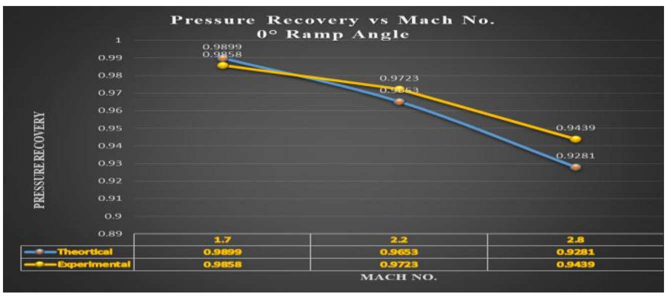

Fig 8. Theoretical and CFD Analysis vs Mach no. plot for 3° ramp angle

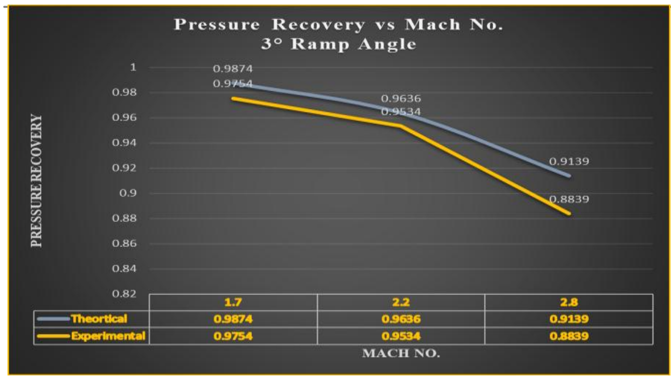

Fig 9 Theoretical and CFD Analysis vs Mach no. plot for 3° ramp angle

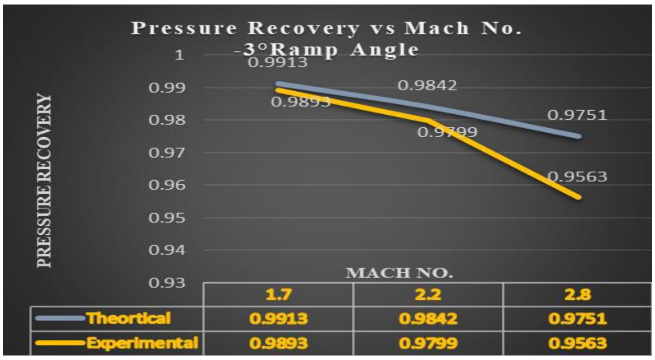

Fig 10 Theoretical and CFD Analysis vs Mach no. plot for -3° ramp angle

5. CONCLUSION

A method for preliminary design of a two-dimensional supersonic inlet has been proposed and implemented in this paper, with the objectives of maximizing the total pressure recovery and matching the engine mass flow demand. The result of total pressure recovery for on-design condition is considered acceptable according to the comparisons with theoretical data and CFD simulation. A method to estimate the total pressure recovery for conditions for different ramp angles such as -3°,0° & +3° for varying Mach no. 1.7, 2.2 & 2.8 are been verified and tabulated.

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