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Preliminary Design of a 2D Supersonic Inlet to Maximize Total Pressure Recovery

Hongjun Ran* and Dimitri Mavris†
Aerospace Systems Design Laboratory, School of Aerospace Engineering
Georgia Institute of Technology, Atlanta, GA 30332

This paper provides a method of preliminary design for a two-dimensional, mixed compression, two-ramp supersonic inlet 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 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, conservative estimation of the total pressure recovery is given so that performance of the engine at the 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.

Nomenclature

α	=	Angle of attack
β_j	=	The installation angle of the jth ramp
γ	=	The ratio of specific heats
δ_j	=	The flow deflection angle of the jth shock (jth ramp half angle)
θ_d	=	The half expansion angle of the subsonic diffuser
θ_j	=	The shock wave angle of the jth shock
A^*	=	The cross section area of flow tube at throat where the flow is sonic
A_j	=	The cross section area of flow at jth station point
AR_{54}	=	The ratio of inlet cross section areas at station points 5 and 4
d_5	=	The engine diameter at station point 5 (engine face)
d_6	=	The engine diameter at station point 6 (fan face)
H	=	Flight altitude
h_c, h_0	=	The captured freestream flow tube height
h_i	=	The height of inlet at the entry, measured perpendicular to the flight direction
h_j	=	The height of jth station point, measured from the lower surface of the inlet and perpendicular to the flight direction
h_d_6	=	The ratio of h_j and d_6
h_t	=	Engine hub-tip ratio
K_d	=	The subsonic diffuser total pressure loss factor (for expansion loss)

* Graduate Research Assistant, Aerospace Systems Design Laboratory (ASDL), Student Member AIAA.
† Boeing Chair in Advanced Aerospace Systems Analysis, Director, ASDL, Associate Fellow AIAA.

K_{Mth}	=	Throat Mach number factor (for friction loss)
l	=	The distance between two station points, measured along flight direction
$l_j\_d_6$	=	The ratio of l_j and d_6
M_∞, M_0	=	Freestream Mach number
M_4_up	=	The upstream Mach number before the terminal normal shock
M_j	=	Mach number after the jth shock or station point
PR	=	The ratio of two total pressures
P_{tj}	=	The total pressure at j[th] station point
TPR	=	The total pressure recovery
w	=	The supersonic diffuser width
w_d_6	=	The ratio of supersonic diffuser width to fan face diameter
w_{a_eng}	=	The mass flow rate needed by the engine
w_{a_inlet}	=	The mass flow rate that the inlet will provide
W_{ff2}	=	The flow function value at engine face (in engine station nomenclature)
W_{ff4}	=	The flow function value at station point 4
W_{ff5}	=	The flow function value at station point 5
$\_design$	=	on design condition
$\_detach$	=	on condition of a detached shock
$\_sub$	=	subsonic diffuser

I. Introduction

The inlet is a duct before the engine. Its basic function is to capture a certain amount of air from the freestream and supply it to the engine. Most gas turbine engines require the Mach number at the 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 freestream to subsonic speed, and provide a matched air mass flow rate to the engine.

The gas turbine engine requires a supply of uniform high total pressure recovery air for good performance and operation, thus the quality of the 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. For 1% 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 freestream.

Because a supersonic inlet has two parts, the supersonic diffuser for supersonic compression 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 stages. 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 a supersonic inlet and the total pressure recovery in the early design stages. This paper provides such a method for a 2D supersonic inlet.

II. Background

As part of the Revolutionary Aeropropulsion Concepts or RAC project at NASA Glenn Research Center, a concept of “embedded wing propulsion” has been pursued. One of the ideas proposed for this concept is to embed the engines in the wing of a supersonic transport aircraft, or a supersonic business jet aircraft that flies at Mach 2.2 at 55,000 ft. Funded by URETI (University Research, Engineering and Technology Institutes), an investigation of this idea was undertaken at the Aerospace Systems Design Laboratory (ASDL) at Georgia Tech in early 2003.

Because the engines are embedded in the wing of the supersonic transport aircraft, preliminary design of the inlet, including estimations of the total pressure recovery and size, is necessary as the size of the inlet will affect the size of the wing and the total pressure recovery will affect the performance of the engine.

III. Proposed Method and Implementation

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 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, the maximum total pressure recovery of the on-design condition, and the estimations of total pressure recovery of off-design conditions.

A. Selection of Inlet Configuration

For a supersonic inlet, the freestream is decelerated to the subsonic engine face entry speed through a suitable shock system (including oblique shocks and a normal shock) 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.[1]

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

The cross section of the supersonic diffuser can be annular (axisymmetric) or rectangular (two-dimensional). In general, an annular supersonic diffuser will have higher total pressure recovery as long as the freestream flow is aligned with the center 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.[2] Also, a two-dimensional supersonic diffuser can provide a larger variation in inlet flow and is obviously much simpler in design.[3] 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 trade-offs 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 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. 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. 1 have the same pressure recovery, they are different in the following aspects:[1]

(1) Self-starting: inlet A is self-starting because of the external normal shock, neither B nor C are 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.

At Mach 2.2, the difference in external wave drag among the three inlets can be quite obvious.[4] Inlet B uses a mixed compression method, and was selected for this supersonic aircraft after considering trade-off among the previous four factors. In fact, this is the type of inlet used on the Concorde after much optimization and experimenation, and it was the inlet design proposed for both U.S. supersonic aircraft programs.

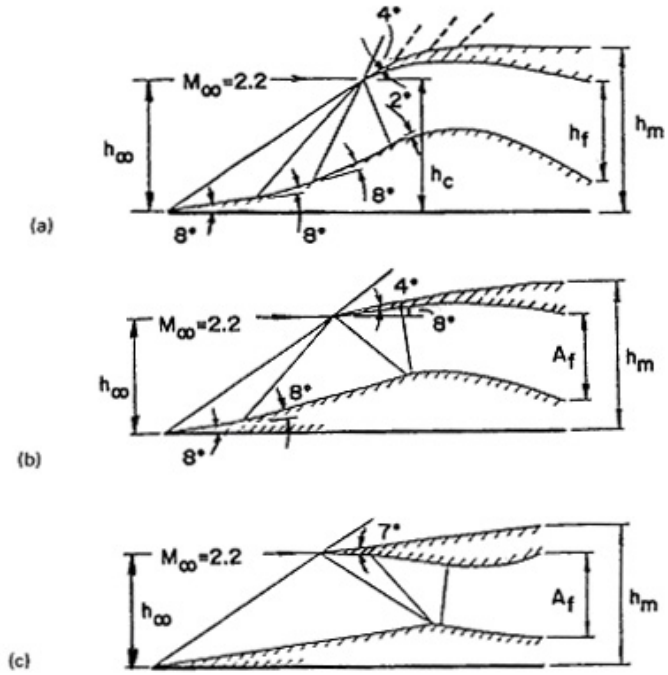

Figure 1. Inlets A, B, C depicting three compression methods[1]

Figure 2 shows how the shock wave pressure recovery varies with freestream Mach number and number of oblique shocks. The shock system in Fig. 2 includes a number of oblique shocks and one normal shock. With $n = 1$, there is a single normal shock and this curve represents pitot type inlet total pressure recovery. For a given Mach number, the more oblique shocks, the higher the percent of pressure recovery.

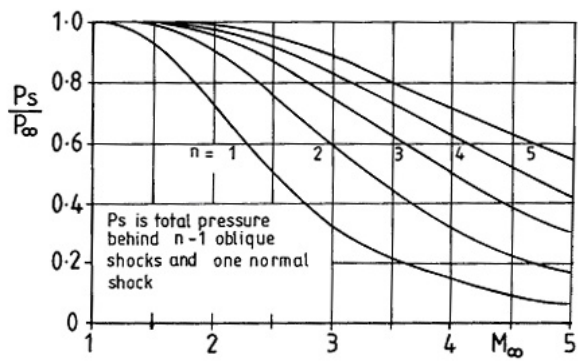

Figure 2. Shock pressure recovery for freestream Mach number and number of oblique shocks[1]

The number of oblique shocks is chosen by the designer, and usually increases as flight Mach number rises.[1] However, it has been found in practice that if the number of oblique shocks is increased, the flow complication will be greater and the flow quality will be less satisfactory because of the boundary layer and the interaction between the boundary layer and shocks.[1, 5] Also, the inlet length will increase with an increase in the number of oblique shocks and hence the inlet weight will increase.

On the other hand, as shown in Fig. 2, for Mach 2.2 freestream speed, a shock system of two oblique and one normal shocks can get about 96% total pressure recovery, and with three oblique shocks, about 97% pressure recovery. Both results are quite satisfactory. However, this does not include the total pressure loss in the internal subsonic diffuser, which reduces the total pressure recovery by 1- 2%.

Therefore, after a trade-off among pressure recovery, flow complication, and inlet weight, a shock system of three oblique shocks and one normal shock was selected for this supersonic inlet in order to achieve a 95% total pressure recovery for the whole inlet.

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 (the Mach number after the normal

shock) 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 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. 3. The double lines originating from point 1, 2, 3 and 4 represent oblique or normal shocks. The symbols and station numbers will be referenced in the calculation code and a subsequent part of this paper.

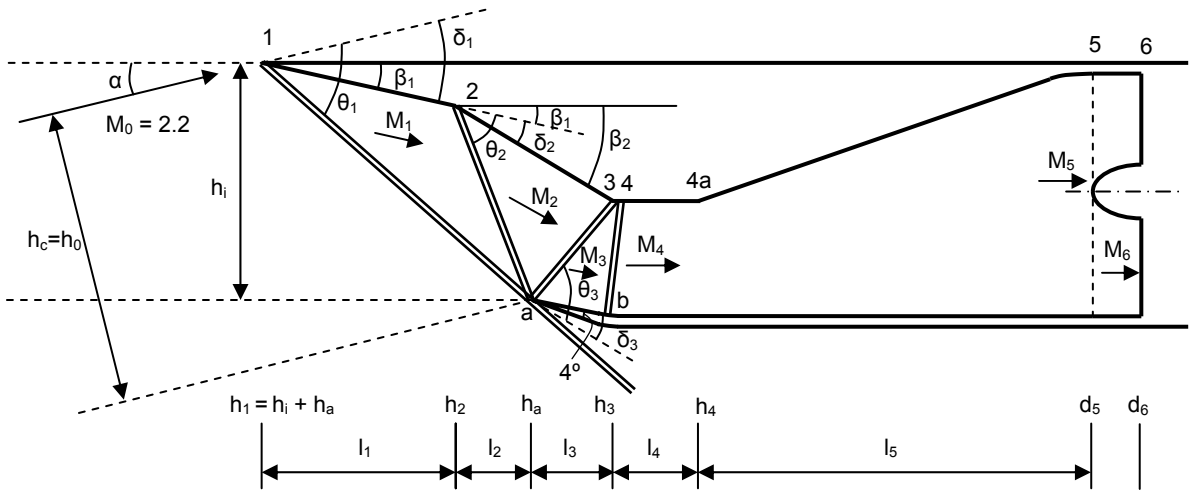

Figure 3. Sketch of the whole inlet system with on-design shock positions

B. The Optimization Criterion to Maximize 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 (1944) and is described as follows.[1]

In a system of (n-1) oblique shocks and one normal shock in two dimensions as shown in Fig. 4, 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.

$$M_1 \sin\theta_1 = M_2 \sin\theta_2 = \cdots = M_{n-1} \sin\theta_{n-1} \tag{1}$$

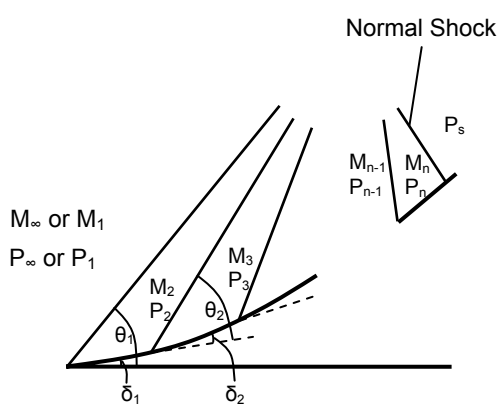

Figure 4. Multi shock compression for Oswatisch optimization

C. Design of the Inlet According to On-Design Conditions

Given the freestream Mach number M_0, angle of attack α, flight altitude H, the normal shock up-stream Mach number M_4_up, the engine face hub-tip ratio h_t, the fan face entry Mach number M_6 (in the annulus), 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 solve the shock system, otherwise the number of unknown variables is more than the number of the 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 the normal shock, 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 1.25 for freestream Mach number 2.2 in the example given later.

C1. Solution of the Mach Numbers 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 Fig. 3.

Given flight altitude H, the freestream air specific heats ratio γ can be determined. Given the freestream Mach number M_0, and specifying the normal shock up-stream Mach number M_4_up, there are 8 basic equations for the 3 oblique shocks shown in Fig. 3 (see Ref. 6 for more details).

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

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

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

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

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

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

Applying the optimum criteria stated in Eq. 1, there are two equations:

$$M_0\sin\theta_1 = M_1\sin\theta_2 \tag{8}$$

$$M_1\sin\theta_2 = M_2\sin\theta_3 \tag{9}$$

M_3 is assumed to be equal to M_4_up, and M_4_up will be a given input parameter, therefore M_3 is known.

$$M_3 = M_4\_up \tag{10}$$

There are 9 unknown variables in Eq. (2) – (10) and there are 9 equations, thus Eq. (2) – (10) can be solved numerically.

Since M_4_up is given, M_4, the Mach number just after the normal shock, is calculated by the normal shock equation:

$$M_4^2 = \frac{(\gamma-1)M_4^2\_up + 2}{2\gamma M_4^2\_up - (\gamma-1)} \tag{11}$$

In order to calculate M_5, assume that the M_6 and hub-tip ratio h_t are given from the engine data.

Assuming the duct diameter is constant from station 5 to 6, we have the following relation for the airflow areas:

$$\frac{A_5}{A_6} = \frac{1}{1 - h\_t^2} \tag{12}$$

$$\frac{A_5}{A_6} = \frac{A_5 / A^*}{A_6 / A^*} \tag{13}$$

According to the area-Mach number relation,[7] we have:

$$\left(\frac{A_6}{A^*}\right)^2 = \frac{1}{M_6^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M_6^2\right)\right]^{\frac{\gamma+1}{\gamma-1}} \tag{14}$$

$$\left(\frac{A_5}{A^*}\right)^2 = \frac{1}{M_5^2}\left[\frac{2}{\gamma+1}\left(1+\frac{\gamma-1}{2}M_5^2\right)\right]^{\frac{\gamma+1}{\gamma-1}} \tag{15}$$

Since M_6 and h_t are given, then $\frac{A_6}{A^*}$ and $\frac{A_5}{A^*}$ are calculated according to Eq. (12) – (14). Then numerically solve Eq. (15) for M_5, thus M_5 is calculated.

With M_4_up and M_6 given, the other Mach numbers at different positions are determined.

C2. Calculation of Total Pressure Recovery

For the oblique shocks shown in Fig. 3, the total pressure ratio across each oblique shock is calculated as following (see Ref. 6 for more details):

$$PR_i = \left[\frac{(\gamma+1)M_{i-1}^2\sin^2\theta_i}{(\gamma-1)M_{i-1}^2\sin^2\theta_i+2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{(\gamma+1)}{2\gamma M_{i-1}^2\sin^2\theta_i-(\gamma-1)}\right]^{\frac{1}{\gamma-1}},\ i=1-3, \tag{16}$$

The total pressure ratio across the normal shock is calculated as following:

$$PR_4 = \left[\frac{(\gamma+1)M_4^2\_up}{(\gamma-1)M_4^2\_up+2}\right]^{\frac{\gamma}{\gamma-1}}\left[\frac{(\gamma+1)}{2\gamma M_4^2\_up-(\gamma-1)}\right]^{\frac{1}{\gamma-1}} \tag{17}$$

For the subsonic diffuser, assume the total temperature is constant, then according to the equation of flow function, we have:

$$PR_{\_Sub} = \frac{P_{t5}}{P_{t4}} = \frac{1}{AR_{54}}\frac{W_{ff4}}{W_{ff5}} \tag{18}$$

The flow function values W_{ff4} and W_{ff5} are determined by static temperatures t_4 and t_5, and the Mach numbers M_4 and M_5.

Based on Borda-Carnot loss equation, the following equation is derived with correction factors:

$$\frac{P_{t5}}{P_{t4}} = 1-K_{Mth}K_d\left(1-\frac{1}{AR_{54}}\right)^2\frac{\frac{\gamma}{2}M_4^2}{\left(1+\frac{\gamma-1}{2}M_4^2\right)^{\frac{\gamma}{\gamma-1}}} \tag{19}$$

The coefficient K_{Mth} accounts for friction loss and K_d accounts for expansion loss. Their values are determined according to the experimental data in Ref. 8. Then the values of PR_5 and AR_{54} are determined by solving Eq. (18) and (19) simultaneously.

The total pressure recovery is then calculated as following:

$$TPR = \prod_{i=1}^{4} PR_i \times PR_{\_sub} \tag{20}$$

C3. Calculation of the Dimensions of the Inlet

In order to calculate the dimensions of the inlet, the following selections of parameters and assumptions are made.

Define the part of the inlet from station point 1 to 4a as two dimensional with the width w equal to the engine face diameter d_6. Next, 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. According to Ref. 8, 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. According to Ref. 8, the expansion angle $2\theta_d$ should be 6 – 12 degrees in order to obtain a high total pressure ratio. Therefore, the expansion angle $2\theta_d$ is selected as 12 degrees to have short length.

The dimensions of the inlet are normalized with respect to the engine face diameter d_6. With the angles obtained from the solution of Eq. (2) – (10), the area ratio AR_{54}, and using the previously discussed parameter selections and assumptions, a series of geometric equations can be established. Solving these geometric equations yields the normalized dimensions of the inlet, i.e., $l_1\_d_6$ - $l_5\_d_6$, $h_1\_d_6$ - $h_5\_d_6$, and $h_a\_d_6$. Thus the dimensions of the inlet are calculated.

C4. Satisfaction of the Engine Mass Flow Demand

The continuity requirement is embedded in the process to calculate the dimensions of the inlet. For example, the flow function W_{ff} used to calculate the area ratio AR_{54} is actually derived from the continuity equation, and the continuity equation is also used to derive the shock equations. Because the dimensions of the inlet are normalized to the engine face diameter d_6, when d_6 is given, the dimensions of the inlet will be determined, and the inlet will capture the mass flow needed and supply it to the engine.

D. The Inlet Design Result and Comparison with Experimental Data

This section presents an inlet design completed using on-design conditions and compares the results with experimental data and a CFD simulation.

D1. The Inlet Designed According to On-Design Conditions

The on-design conditions are as follows:

Given:

$$M_0 = 2.2$$

$$H = 55{,}000\ ft$$

Assuming:

$$\alpha = 3°$$

$$M_6 = 0.395$$

$$h\_t = 0.3204$$

$$M_4\_up = 1.25$$

Results:

Total pressure recovery: $TPR = 0.942$.

Total length: 5.336 times engine face diameter.

The normalized dimensions of the inlet are shown in Table 1.

Table 1. Normalized Dimensions of the Inlet

$l_1\_d_6$	$l_2\_d_6$	$l_3\_d_6$	$l_4\_d_6$	$l_5\_d_6$
0.568	0.700	0.537	0.890	2.641
$h_1\_d_6$	$h_2\_d_6$	$h_3\_d_6$	$h_4\_d_6$	$h_a\_d_6$
0.801	0.751	0.445	0.445	0.043

A sketch of this normalized inlet is shown in Fig. 5.

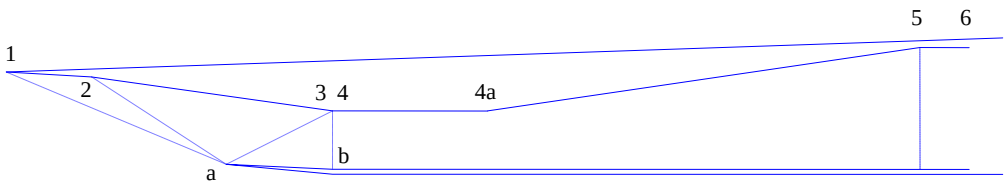

Figure 5. Sketch of the Normalized Inlet

D2. Comparison with Experimental Data

In Ref. 8, the maximum total pressure recovery of supersonic inlets with offset diffusers is given as between 0.94 and 0.97. As shown above, the total pressure recovery of this inlet is 0.942. Therefore, the calculated result for this inlet is acceptable.

D3. Comparison with CFD Simulation Result

A NASA aerodynamic code named CFL3D was used to simulate the optimum inlet obtained in subsection *D1*. CFL3D is a mature Navier-Stokes solver, and gives very accurate results if the model is properly established and the boundary conditions are properly set. Currently, only a two dimensional model has been established in CFL3D for the three dimensional inlet obtained in subsection *D1*.

The static pressure contours and Mach number contours obtained by CFL3D are shown in Fig. 6 and Fig. 7.

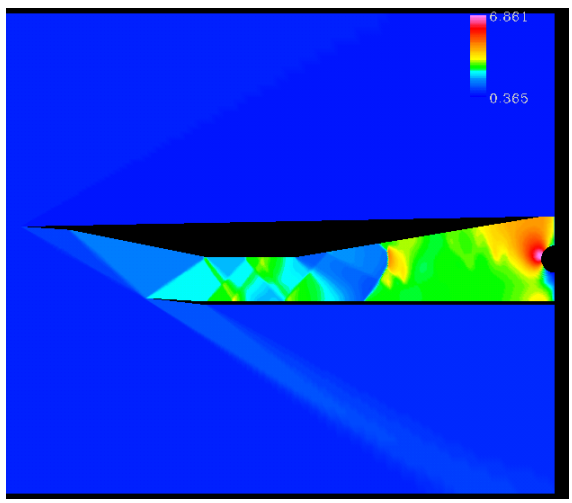

Figure 6. The Static Pressure Contours

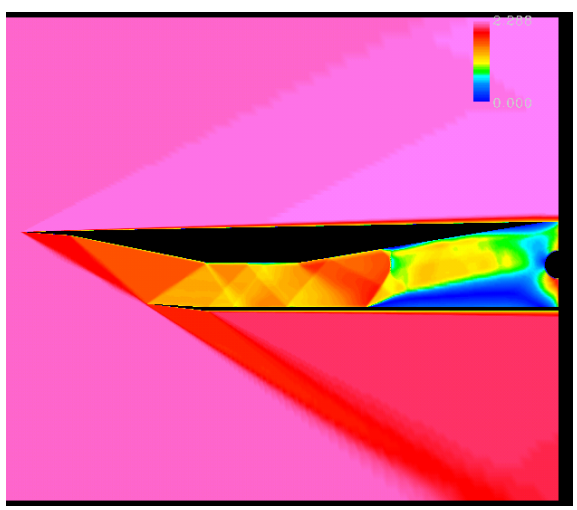

Figure 7. The Mach Number Contours

As seen in Fig. 6 and Fig. 7, the oblique shocks occur as expected, but the terminal normal occurs at a higher Mach number and far downstream of station point 3 where it was expected to occur. As a result of this unexpected terminal normal shock, the total pressure recovery obtained by this CFD simulation is about 0.88, much less than the calculated result of 0.942. Note that the inlet in the CFD simulation can be considered non self-started, thus leading to the conclusion that the CFD comparison shows that the calculation result is acceptable as well.

E. The inlet for Off-Design Conditions

It is very difficult to calculate the total pressure recovery for off-design conditions. However, with some assumptions and simplifications, a conservative total pressure recovery can be calculated. For off-design conditions, some procedures are required to match the captured mass flow with the mass flow demand of the engine, such as adjusting the second ramp angle or opening a bypass door. The following describes how to estimate the total pressure recovery and match the engine mass flow demand for off-design conditions.

E1. Estimation of Total Pressure Recovery for Off-Design Conditions

Assume the on-design cruise speed is the highest speed the aircraft will fly. Then at off-design points, in general, the total pressure recovery will be higher than those of the on-design points because the flow speed is lower and thus the shocks are weaker and losses are less. However, it is very hard to calculate the total pressure recovery at off-design points, because the flow field is much more complicated. For example, a detached shock may occur. In order to provide an estimation of inlet pressure recovery for engine performance evaluation, a conservative value of inlet pressure recovery can be given by using the following assumptions or simplifications:

1) Whenever an oblique shock becomes detached, the equations of normal shock are used to calculate the parameters of this shock, and the shock line is perpendicular to the upstream flow and the downstream flow is parallel to the ramp.

2) If the shock that originates at the lower cowl lip is an attached oblique shock, assume a normal shock will occur just after this oblique shock.

3) The total pressure ratio of the subsonic diffuser is $PR_{\_sub} = 0.98$, which is a conservative estimation according to calculations performed. However, let $PR_{\_sub} = 0.985$ if there is a detached shock because the equations of normal shock were used to calculate the parameters of this shock, resulting in a lower total pressure recovery. Therefore, $PR_{\_sub}$ was increased a small amount to compensate.

E2. Matching the Mass Flow Demands of the Engine

The mass flow demands of the engine for off-design conditions are provided by engine data and are known to the designer.

For off-design conditions, the second ramp angle is first adjusted to match the captured mass flow with the mass flow need of the engine. The second ramp angle is adjusted between 0 degree and a higher limit. The higher limit is the on-design angle, or the angle at which the shock will detach, whichever is smaller. Decreasing the angle will allow more air to enter and increasing the angle will allow less air to enter. At the beginning, the second ramp angle is set equal to the on-design angle and the captured air mass flow is calculated. If the captured mass flow is different from the engine need, the second ramp angle is adjusted until it reaches the lower limit of 0, or the higher limit. If the captured mass flow after adjustment of the second ramp angle still does not match the engine demand, a bypass door will open to dump extra air, or an auxiliary door will open to provide more air.

E3. Results for off-Design Conditions

The off-design conditions used in the calculation are shown in Table 2. The freestream dynamic pressure is kept constant for all off-design points.

Table 2. The off-Design Conditions

Off-Design Point	Freestream Dynamic Pressure (psi)	Mach No.	Ambient Static Pressure (psi)	Altitude (ft)
1	4.4835	1.8	1.9768	46,650
2	4.4835	1.6	2.5019	41,749
3	4.4835	1.4	3.2677	36,192

Note: the engine face diameter $d_6 = 2.1$ ft.

The results for the above off-design conditions are shown in Table 3. Comparing with the TPR of 0.942 at on-design condition, the TPR's at off-design conditions are greater and increase with decrease of Mach number as expected. Comparing with results calculated according to the military standard MIL-E-5007D, which is to estimate the inlet pressure recovery, the TPR's at off-design conditions are smaller. These results show that the previous assumptions and simplifications are acceptable.

Table 3. Results for the off-Design Conditions

Off-Design Point	Mass Flow Demand per Engine (lbm/sec)	δ_2	Bypass door	TPR
1	67.0	7.5º	open	0.956
2	69.7	7.5º	open	0.974
3	75.7	2.2º	open	0.978

Note: 1. the on-design second ramp angle is $\delta_{2\_design} = 7.5°$.

2. for off-design point 3, the second ramp angle is adjusted to the detach angle ($\delta_{2\_detach} = 2.2°$).

IV. 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 experimental data and CFD simulation. A method to estimate the total pressure recovery for off-design conditions was also proposed and implemented in this paper. The results for off-design conditions show this method is acceptable.

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