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Effect of Leading Edge and Trailing Edge Radii on a 2-D Low Wave Drag Supersonic Busemann Biplane at Design and Off-Design Condition

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Abstract
This paper addresses the practical application of the Busemann biplane under design and off design conditions. The Computational Fluid Dynamics Techniques is used to optimize the practical application of Busemann biplane for different Leading edge and trailing edge radius (1mm, 2mm, 3mm and 5mm) for non-lifting conditions. We also discuss the practical application of the different stagger configuration for different leading edge and trailing edge radius. From our analysis we found that the total drag coefficient for both Busemann biplane and for the stagger configuration will decreases for a range of free increases $M_\infty < 1.0$, and the drag coefficient is remains constant for $1.0 \leq M\infty \leq 1.7$, but for the $M_\infty > 1.7$, the drag coefficient is increases due to the strong bow shock wave ahead of the elements. It is also observe that the drag will further increase with the leading and trailing edge radius due to the strengthen the bow shock wave.

Introduction
Supersonic aircraft are those, having speed of aircraft is more than the speed of sound. There are only few supersonic aircraft; the BELL XI is the first aircraft to fly at supersonic speed in 1947. Thereafter many military aircraft are designed to fly at supersonic speeds. But the passenger supersonic aircraft not so successful ever for Concorde (only for smaller route and very expensive). The fundamental problems with the supersonic commercial aircraft is the generation of strong sonic booms, sonic boom is the sound (sound like explosion) of shock waves produce by an aircraft moving at supersonic speed.
For supersonic aircraft, wing with low intensity sonic boom and low wave drag are preferred, because a strong bow shock wave is created in front of the wing and this will create a very high pressure in front of the wing, which produces a large amount of pressure drag, hence supersonic airfoil shape have a sharp Leading edge so that the bow shock wave is attached to the airfoil and high pressure zone ahead of the leading edge will be eliminated. Other side the airfoil with sharp leading edge has very high stalling speed and requires very large runways in subsonic flight. The concept of supersonic biplane is first introduced by Busemann in 1935[1], after the Concorde finished his last flight in 2003; it is our dream to develop the supersonic transport aircraft. The major problem with the supersonic aircraft is the strong sonic boom and the large value of the wave drag. Busemann proposed the biplane concept which significantly reduces the wave drag by wave cancellation effect.
Recently Kusunose proposed a new generation supersonic transport aircraft design. His research group carried out both computational fluid dynamics (CFD) and wind tunnel experiments [9, 10, 11, 13, 14, and 15]. They first analyzed the advantages biplane in supersonic flow and designed a two dimensional Busemann type biplane airfoils in zero-lift and non-zero lift conditions. Furthermore, they propose using inverse designed method, optimized Busemann type biplane. Then they studied the poor off-design aerodynamic performance parameters of the standard Busemann biplane and proposed a new design by introducing trailing edge and leading edge flaps.

A Biplane concept for Wave Elimination
A. Wave Reduction effect
Using thin airfoil theory for 2-D supersonic airfoil [11], the lift and wave drag coefficients of a flat plate at a small angle of attack α are given by:

$$C_L = \frac{4\alpha}{\sqrt{M_\infty^2 - 1}} \tag{1}$$

$$C_D = \frac{4\alpha^2}{\sqrt{M_\infty^2 - 1}} = \frac{\sqrt{M_\infty^2 - 1}}{4} C_L^2 \tag{2}$$

Where C_L and C_D are defined by Lift and wave Drag coefficient, where M_∞, q and c are the free stream Mach number, dynamic pressure and chord of the airfoil respectively.
Consider an airfoil with n parallel plates at an angle of attack α, in order to produce an lift coefficient equal to produce by a single flat plate as shown in figure-1. Assuming the chord of the flat plate is same in both the cases and the angle of attack α and αs are related as:

$$\alpha = \frac{\alpha_s}{n} \tag{3}$$

The wave drag for n plate system is equal to the sum of the wave drag produce by the each of the n individual elements.

$$D = n \frac{4}{\sqrt{M_\infty^2 - 1}} \left(\frac{\alpha_s}{n}\right)^2 qc = \frac{1}{n}\left(\frac{4}{\sqrt{M_\infty^2 - 1}} \alpha_s^2\right) qc \tag{4}$$

It is clear from the above expression, the wave drag produce by the n parallel plate is 1/n times then the wave drag produces by single flat plate under the same lift condition, whereas the skin friction drag is increased by n times then the single flat plate because of increment in the surface area of the airfoil.

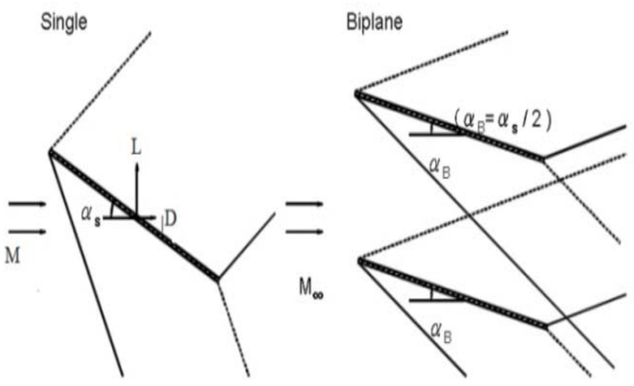


Figure-1. Wave reduction effect. [Ref. 11]

B. Wave Cancellation effect

The biplane configuration also reduces the wave drag produce by the airfoil thickness. The wave interaction between the two elements can be promoted by choosing their geometries and relative locations carefully Busemann shows that the wave drag is completely removed at zero lift condition at a free stream Mach number of 1.7, by simply divided the diamond airfoil into two half and placing them in such a way that the wave generated by the one element is cancelled by the other ($t/c = 0.1$, $z = 0.5c$ and wedge angle $\varepsilon = 5.71°$) as shown in figure-2.

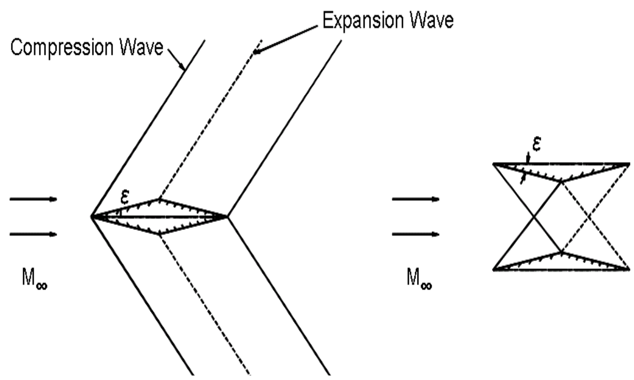
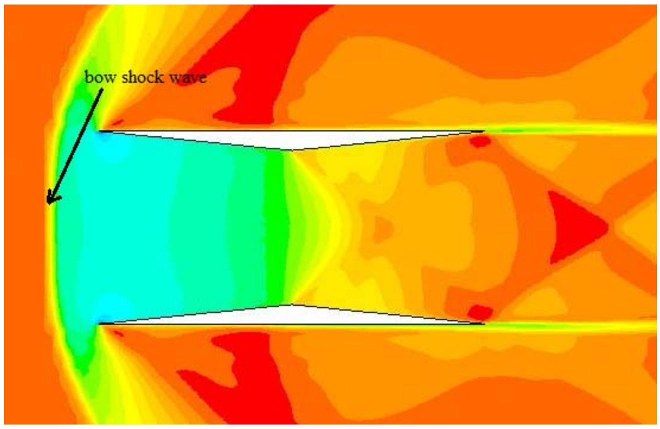


Figure-2. Wave cancellation effect of Busemann airfoil. [Ref. 11]

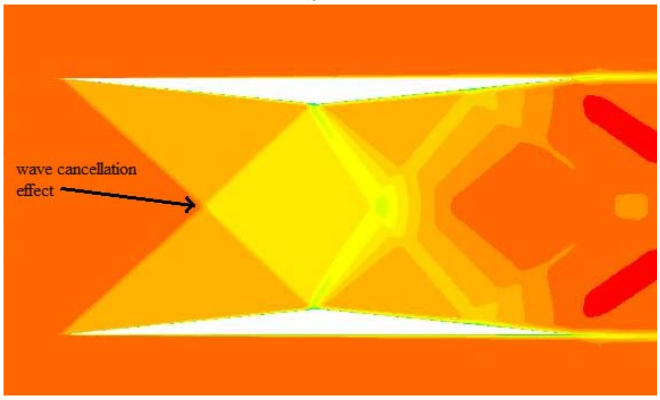
C. Off-Design condition of Busemann biplane

The wave cancellation effect for the Busemann biplane is possible to a specified flow conditions shown in Figure-3. Unfortunately at other flow conditions or Off-Design conditions ($M_\infty<1.7$), Busemann biplane generates a strong bow shock wave at upstream of the airfoil and produces a large amount of drag as shown in figure-4.

The detailed drag variation for the Diamond and the Busemann biplane with free stream Mach number ($0.5 \leq M_\infty \leq 3.0$) as shown in figure-4.



$M_\infty = 1.6$



$M_\infty = 1.7$

Figure-3. Busemann biplane and design and off-design condition

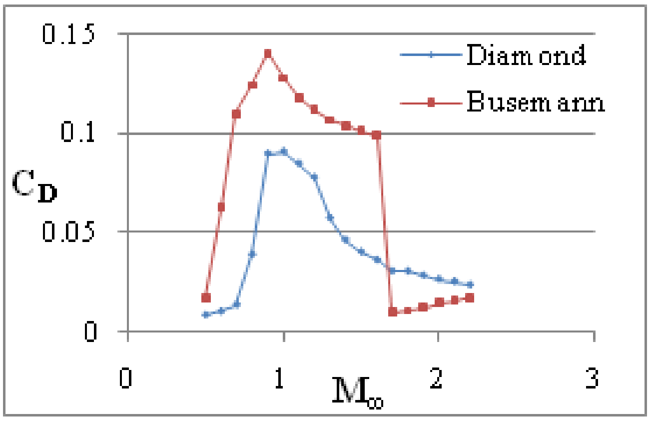


Figure 4. Drag Characteristics of Diamond and Busemann airfoils.

Results and Discussion

The Navier-Strokes equations were solved for all the configuration by the CFD tool (ANSYS-FLUENT) using structured grid, were utilized for viscid flow analysis, mainly focus on the shock wave properties arround the airfoil.

All geometries are created with the distance between the elements is 0.5c and the t/c is 0.05 for each element and the analysis is done for different vales Leading edge and trailing edge radius of 1mm, 2mm, 3mm and 5mm as shown in figure

5. The multi block unstructured grids are prepared as shown in Figure-6 with the help of ICEMCFD. The total number of elements is around 3.5×10^5, stretched perpendicular to airfoil surface. The boundary layer mesh has the first cell height of 8.11×10^{-6}m in order to resolve viscous stresses. A second order accurate, steady state results are obtained through time marching solution of coupled, 2-dimesional Navier-Stokes equations using FLUENT. A one-equation turbulence model by Spalart-Allmaras is used to consider the effect of turbulent boundary layer for viscous flow computations.

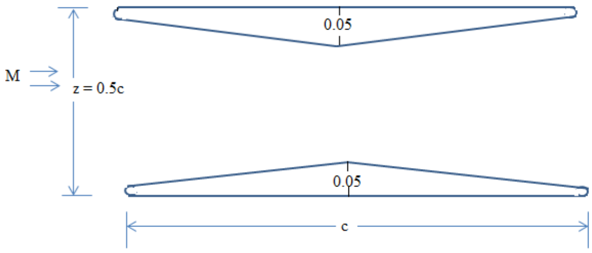


Figure 5. Geometry

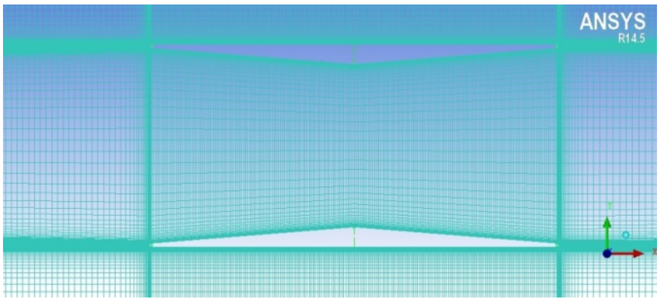


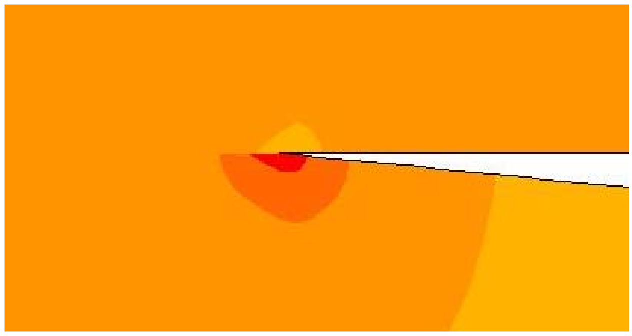
Figure 6. Mesh element around the airfoil

Here, to measure the accuracy of the solution using CFD tool, the standard Diamond and Busemann airfoils are used and the comparision of the solution with the standard thin airfoil as shown in Table 1.

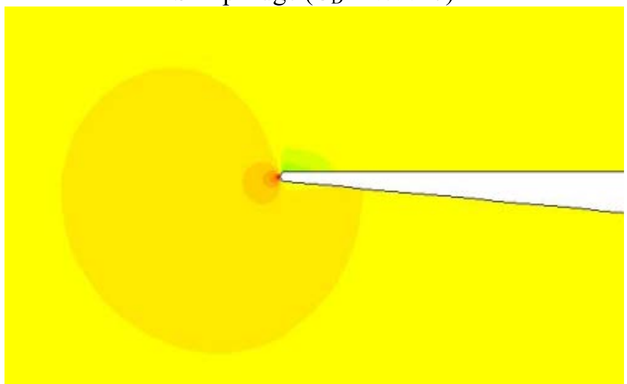
Table 1. Lift and Drag coeffient for Diamond and Busemann airfoil at zero angle of attack

	Thin airfoil theory		CFD	
	C_L	C_D	C_L	C_D
Diamond	0.0000	0.0316	0.0000	0.03086
Busemann	0.0000	0.0000	0.0000	0.00989

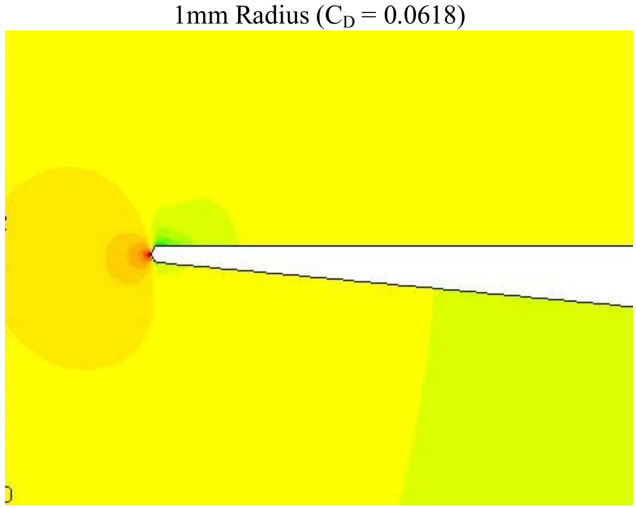
In order to examine the effect of leading edge and trailing edge radius of the Busemann biplane as shown in figure 5 (leading edge and trailing edge radius of 1mm, 2mm, 3mm and 5mm), the total drag was calculated using ANSYS FLUENT 14.5. This configuration decreases the drag at subsonic speeds ($M_\infty < 1.0$), as in subsonic flow the separation will be delayed as the flow is attached to the surfaces, and the drag is further reduced with the increment in leading and trailing edge radius. Figure-7(a) and 7(b) shows the pressure variation for the different leading edge radius at a free stream Mach number of 0.6.



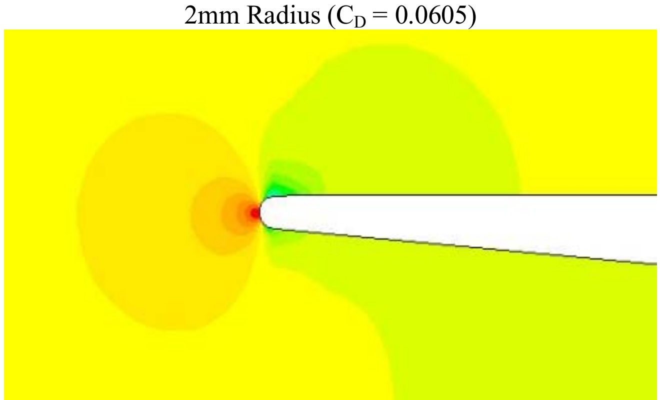
Sharp Edge (C_D = 0.0627)



1mm Radius (C_D = 0.0618)



2mm Radius (C_D = 0.0605)



3mm Radius (C_D = 0.0598)

Figure-7(a). C_P variation at different leading edge radius for M_∞ = 0.6.

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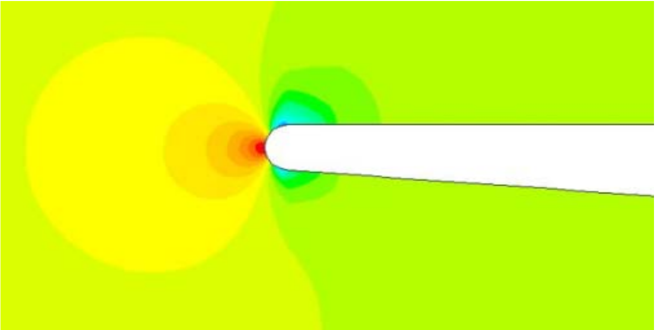
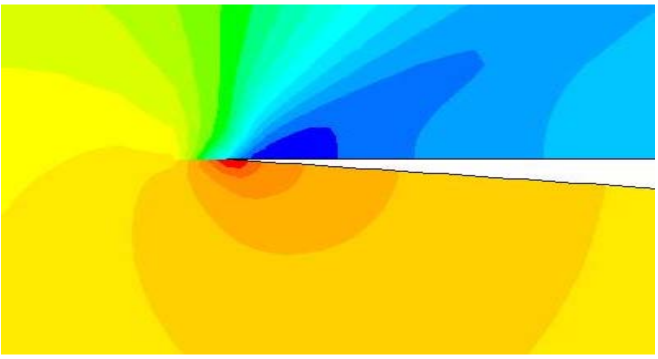


5mm radius (C_D = 0.0479)

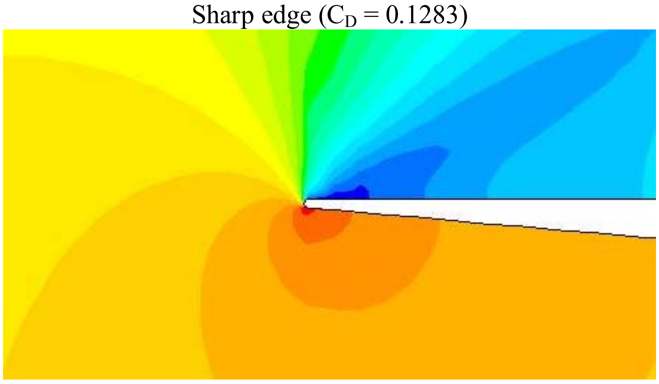
Figure-7(b). C_P variation at 5mm leading edge radius for M_∞ = 0.6.

The pressure near the leading edge of the element is decreased with increasing radius. For the subsonic flow the separation is take place at sharp edge and increases the pressure near the edge of the elements, but in case of leading edge radius, the flow is attached to the surface and delays the flow separation, hence decreases the pressure and drag produce by the elements.

With increasing Mach number the ($0.5 \leq M_\infty \leq 0.9$), the drag is further decreases, as pressure is decreased. At Sonic flow the effect of separation is neutralized due to the shock wave and the pressure is almost same at all values of leading edge radius, figure-8(a) and figure 8(b), shows the pressure variation at M_∞ = 1.0. Hence the drag coefficient is almost same for the $1.0 \leq M_\infty \leq 1.7$, because of the flow chocking; the Mach number ahead of the elements remains subsonic.

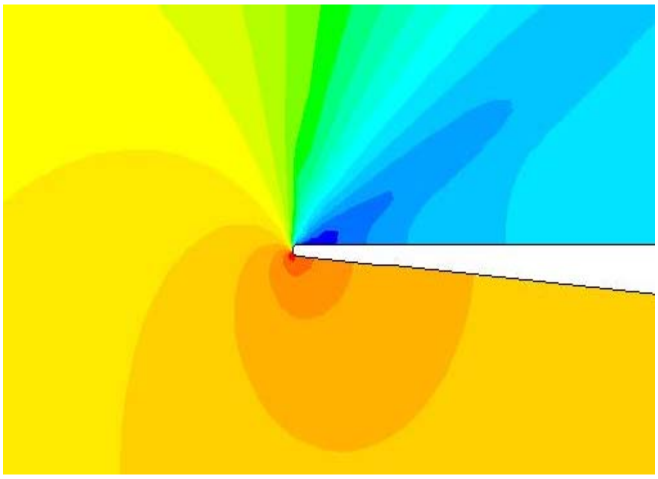


Sharp edge (C_D = 0.1283)

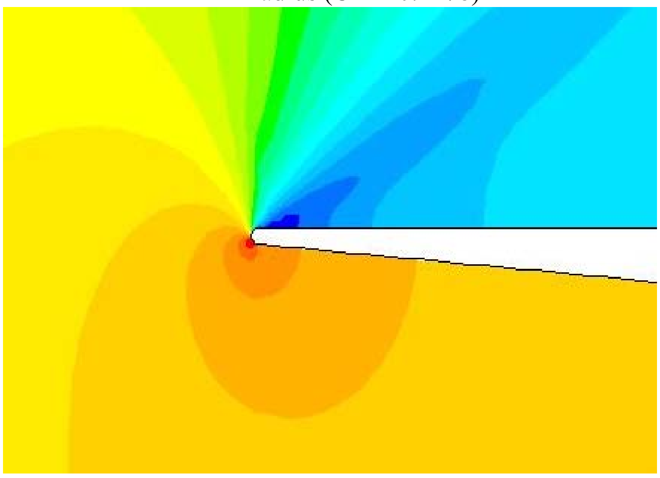


1mm Radius (C_D = 0.1277)

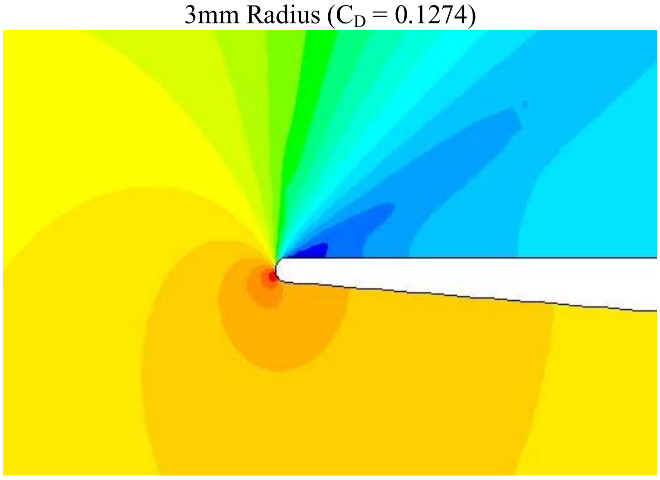
Figure-8(a). C_P variation at different leading edge radius for M_∞ = 1.0



2mm Radius (CD = 0.1278)



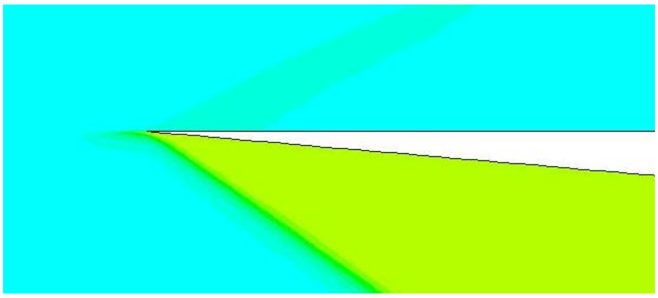
3mm Radius (C_D = 0.1274)



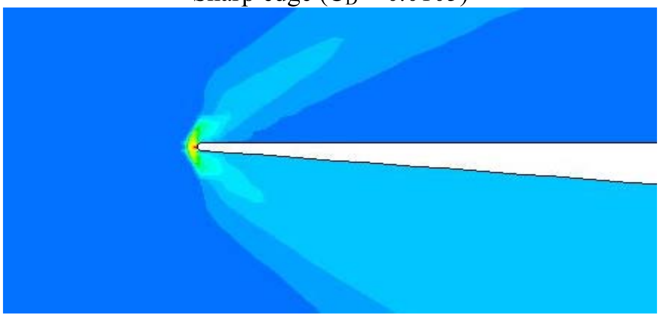
5mm Radius (C_D = 0.1281)

Figure-8(b). C_P variation at different leading edge radius for M_∞ = 1.0.

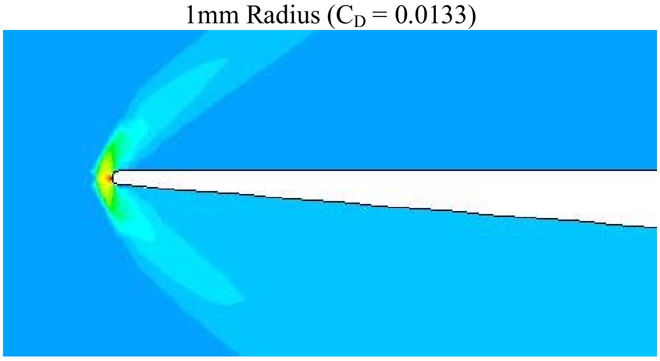
For the range of $M_\infty \geq 1.7$, the flow chocking is eliminated and the flow ahead of edge is supersonic, the supersonic flow induced attached oblique shock at the sharp corner, but at the blunt edge, there is strong oblique shock is induced and pressure near the blunt edge is more than the sharp edge, hence increases the drag coefficient. The C_P variation at M_∞=1.8 as shown in figure-9.



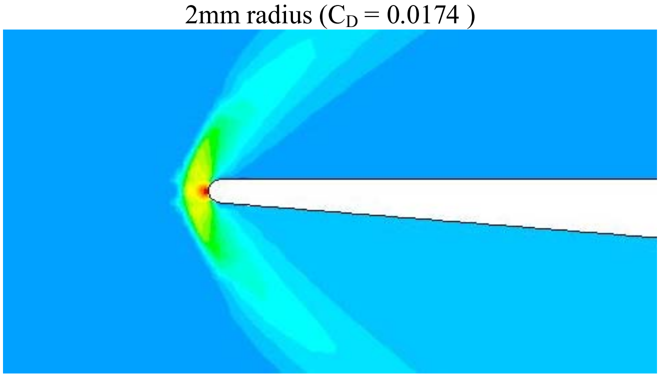
Sharp edge (C_D = 0.0105)



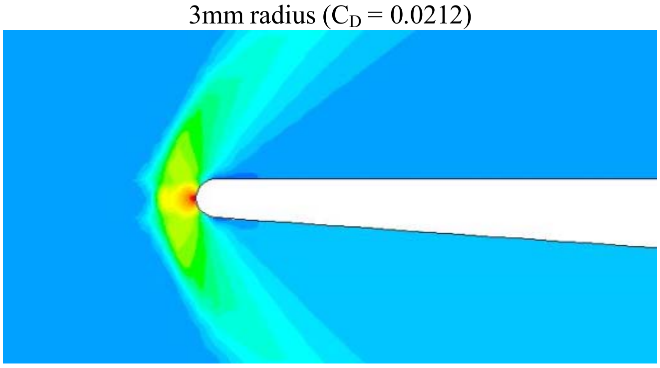
1mm Radius (C_D = 0.0133)



2mm radius (C_D = 0.0174)



3mm radius (C_D = 0.0212)



5mm Radius (C_D = 0.0311)

Figure-9. C_P variation at different leading edge radius for M_∞ = 1.8.

With further increment in the free stream Mach number the strength of the bow shock wave is increased and increases the wave drag. The variation of drag coefficient with free stream Mach number and for different leading edge and trailing edge radius as shown in Figure-10.

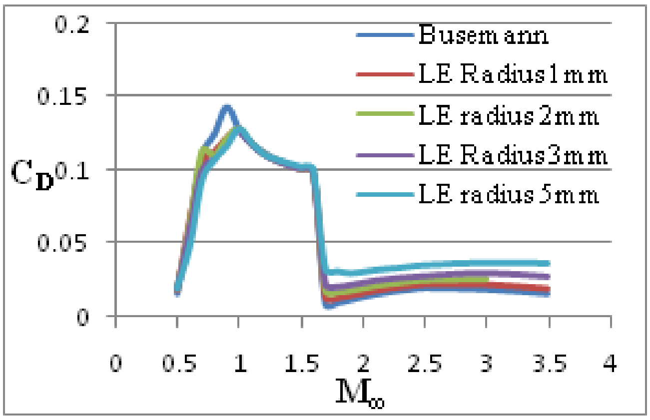


Figure 10. Drag Variation for different Leading Edge Radius with Free Stream Mach Number.

Conclusion

In this paper the practical application of the Busemann biplane at different leading edge and trailing edge radius is discussed and found that the drag coefficient is decreased with the increasing leading edge radius for the subsonic range, hence in subsonic case the flow will be attached at the rounded leading edge and reduces the drag. But for the range of Mach numbers $1.0 \leq M_\infty \leq 1.6$, the drag remains constant as flow remains subsonic in front of the elements the effect remains same for different leading edge radius. But in case of supersonic condition ($M_\infty \geq 1.7$) the flow chocking is eliminated and the wave drag is increased with the increment in the leading edge radius as the strong bow shock is induced in front of the elements.

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