ABSTRACT

For the development of a supersonic transport aircraft, there are two major challenges viz. fuel efficiency and the sonic boom. Due to the high intensity of the shock waves generated in front of the supersonic aircraft, the drag value is significantly high with a reduced lift. The high value of the drag requires more thrust or power from the engine to fly at supersonic speed, hence the efficiency is decreased. When an aircraft moving with the supersonic speeds in air, generates a large value of pressure ahead of the body and form high pressure bubble near the body, these high pressure bubbles will separate from the surface and generates noise, called the sonic boom. The intensity of the sonic boom is very large and it can harm the people on the ground. This restricts the application of supersonic transport aircraft fly over the surface of the ground. These effects can be minimized by reducing the strength of shock waves formed in supersonic flight. The Busemann biplane concept is very famous due to its shock cancellation and shock-interaction effect and nearly shock free supersonic flight is possible at its design Mach number. At off-design Mach numbers however, the performance of the Busemann biplane is poor due to flow choking and incomplete wave cancellation.

Staggered configurations have been proposed for the performance enhancement of these biplanes at off-design conditions. The staggered biplanes are designed in such a way that the throat area of the Busemann biplane increases and reduces the flow chocking ch lift and drag coefficients of various staggered configurations wherein the lower element of the Busemann biplane is shifted backwards by a distance of 0.1c, 0.2c, 0.3c, 0.4c and 0.5c from its original position is studied. In supersonic flow, the shock waves are generated at the sharp

edge, hence the pressure at the sharp edge becomes very large and generate large amount of force at the sharp edge. The edge of the element must be strong enough to resist the pressure increment by the shock waves; therefore it is necessary to use the rounded leading edge. Keeping this aspect in view, staggered Busemann biplanes with different leading edge and trailing edge radii has also been studied using Computational Fluid Dynamics.

All geometries and the subsequent multi-block structured grids are generated using state of the art grid generator ICEMCFD®. The multi-block structured grids are saved in an unstructured format with around 0.35 million quadrilateral elements for the non-staggered configuration. The number of elements is decided based on an extensive mesh independence study at various Mach numbers and angles of attack. The grid is stretched exponentially perpendicular to airfoil surface, in both the longitudinal and lateral directions and has a first cell height of 5×10^{-6} m inside the boundary layer in order to resolve viscous stresses correctly using a turbulence model. The distance of the first cell from the wall boundary is chosen such that the non-dimensional cell-wall-distance, $y+ \approx 1$. The system of equations is closed by perfect gas assumption of the continuum working fluid air. The variations in molecular viscosity, μ and the thermal conductivity k of air are computed using Sutherland formula and kinetic theory of gases respectively. The dependence of specific heat of air, C_p on temperature is computed using piecewise polynomial approximation. The system of nonlinear equations resulting from discretization of two dimensional compressible Navier-Stokes Equations over individual control volumes are solved simultaneously in a coupled manner through a point implicit Gauss-Seidel iterative algorithm in conjunction with the Algebraic Multigrid Method.

The pseudo steady state solutions of the governing equations have been obtained through a density based time marching algorithm using local time stepping for faster convergence. The spatial discretization of fluxes over the control volume is done using second order upwind scheme. The flux vector splitting scheme used to compute the convective flux at the cell face is *AUSM+*, which a modified version of the Advection Upstream Splitting Method (AUSM). In this scheme, the inviscid fluxes are split into convective and pressure terms and discretized separately based on the advection Mach number at the cell interface. As the simulations are performed at Mach numbers between 0.5 and 2.5 with chord based Reynolds number varying from 1.14×10^7 to 5.78×10^7 the Spalart–Allmaras (SA) turbulence model which incorporates both the strain and vorticity based production of the modified turbulent viscosity, has been used to compute eddy viscosity. The SA model is designed specifically for aerospace applications involving wall bounded flows and has been validated for a wide range of hypersonic flows. Boundaries in all the directions away from the biplane are assumed to be characteristic based pressure farfield wherein the freestream pressure, Mach number and static temperature are specified. The wall of the biplane are modeled as no slip viscous surface with $u = v = 0$, with additional isothermal boundary condition implemented on it.

It has been observed that the chocking phenomena seen in Busemann biplane for a Mach number of 1.6, is alleviated by a stagger of 0.2c or more. At lower supersonic Mach numbers, although the choking is not completely eliminated by staggering, the shock wave attached to the leading element is weaker than a bow shock and thus a reduced drag is observed at all Mach numbers. The advantage of using stagger is highly justifiable in the subsonic range where not only the drag is reduced significantly, but a substantial increase in lift is observed. The staggered configuration have shown a consistent list due to lower element suction even at 0^o angle of attack giving L/D ratios of up to 12. The reduction in drag increases with increase in stagger, however the increase in lift diminishes with increasing stagger. The L/D ratio or the aerodynamic efficiency however, in subsonic and transonic range is significantly higher than those for Busemann biplane at all angles of attack.

At Mach 1.7 and higher, the Busemann outperforms all staggered configurations because of small wave drag of the Busemann biplane and negative lift shown by staggered configurations, especially at $\alpha = 0^{o}$. It is found that the drag coefficient is decreased with the increasing leading edge radius for the subsonic range, and remains constant for $1.0 \leq M_{\infty} \leq 1.6$, as flow remains subsonic in front of the elements due to bow shock waves in front of the elements. But in case of supersonic condition the wave drag is increased as the strong bow shock is induced in front of the elements.

For the rounded leading edge biplanes, the drag coefficient is decreased with the increasing leading edge radius for the subsonic range, and remains constant for $1.0 \leq M_{\infty} \leq 1.6$, as flow remains subsonic in front of the elements due to bow shock waves in front of the elements. But in case of supersonic condition the wave drag is increased as the strong bow shock is induced in front of the elements. The increase in drag is however, very small and the staggered biplane configurations with rounded leading and trailing edges can be termed as a suitable candidates for supersonic transport aircrafts to alleviate the sub-design Mach number performance as the airplane has to spend a lot of its mission time at subsonic Mach numbers.