Computational aerodynamic study of a supersonic reference projectile

S Zahir1† and Waseem Gul2
1,2Centres of Excellence in Science & Applied Technologies, Islamabad – 44000; PAKISTAN
†e-mail: cfdpak@apollo.net.pk

Abstract. In the present paper results of computational aerodynamic study of a reference projectile in supersonic flow is presented. Pressure distribution along the projectile's nose in longitudinal and circumferential direction was studied using numerical simulations. Static aerodynamic coefficients in the freestream Mach numbers of 3 and 4 at an angle of incidence of -4 to 20 degrees were computed. For validation, the numerical solution was initially compared for the conic part with the experimental results and found to be in good agreement. Nose related aerodynamic features were initially studied in the first part of the study. Further complete projectile’s aerodynamic analysis is carried out. The present paper covers the results obtained in the study conducted for the projectile’s geometry. Significant flow features were investigated, including correct capturing of bow shock’s location, low pressure density/pressure variations in the low pressure regions along with its influence on the stagnation zone. Further evaluation of peak pressure values associated with the stagnation region on the nose was made. Also, static aerodynamic coefficients were computed to depict projectile’s static stability.

1. Introduction
The flow around a projectile under supersonic conditions presents a flow situation comprised of bow-shock formation ahead of the nose, turbulent boundary layer growth on cylindrical mid-section with possible flow separation and formation of a large turbulent wake in the aft of the body [1-2]. In ballistic aerodynamics, prevention or control of the separation of the boundary layer is one of the most important aims commensurate with the design of an appropriate nose section of a projectile [3-4]. Adequate numerical techniques are available in computational fluid dynamics, CFD to simulate the flow situation for steady state and fully turbulent flows. These techniques were applied to determine the static aerodynamic coefficients for reference projectile. With an incoming free stream Mach numbers of 3 and 4 and a combination of angle of attacks, a representative complete flight envelope was investigated in this study. A salient feature of the flowfield that characterizes the supersonic flow interaction of reference projectile geometry is provided by mapping of the axial pressure distribution on the surface. Flowfield maps showing characteristic pressure and Mach number contours are also plotted. All computational fluid dynamics, CFD simulations are made using an in-house NS solver PAK-3D [5], with post-processing for static aerodynamic coefficients is performed using LOOK [6] software.

2. Computational methodology
2.1. Projectile model geometry
The geometry of a reference projectile is with a body-wing-tail configuration similar to standard body-tail configuration. It has same nose part geometry as of a reference blunted cone nose, as available in the literature [7]. The blunted nose part modeled initially is with a half-cone angle of 10°, a base diameter of 7.65 cm and having a length of 12.0 cm. Apart from blunted conic nose, aft consist of a cylindrical centre-body and a single fin set consisting of four panels. Fins are trapezoidal and with diamond-shaped airfoil cross-section and assumed to be with constant tapering thickness from root to tip. Each fin has a leading-edge sweep. Geometrical details of nose as well complete projectile are, as shown in Table 1.

| Parameter                  | Nose | Projectile |
|----------------------------|------|------------|
| Reference diameter, cm     | 7.65 | 7.65       |
| Reference length, cm       | 12.0 | 61.2       |
| No. of fins                | -    | 04         |
| Fin semi-span              | -    | 7.65       |
| Fin root chord             | -    | 10.2       |
| Fin tip chord              | -    | 5.1        |
| Fin Sweep back angle       | -    | 34°        |

Customary aerodynamic sign convention is used with an in-coming flow is in the positive x-direction and negative angle of attack is with the nose down condition. Axis orientation and geometrical detail (all dimensions are in mm) as a side-view, is also shown in Fig. 1.

Figure 1. Details of nose with complete geometry of the supersonic reference projectile.

2.2. Computational domain and conditions
Quality of computational grid regarding its density and clustering required for capturing flow details in the entire supersonic range of Mach number 3 to 5 is maintained by placing the walls closest cells to lie below a $y^+$ of 2.0. Satisfactory convergence approaches have been described at more length in literature [8], [9] and [10] for the sake of brevity here it is enough to state that an extensive validation and verification was undertaken to arrive at the present approach. To help convergence rate, a systematic grid generation sequence was followed and it is from course to super-fine grids, ranging from 0.72 to about 1.91 million grid points, using grid generation software PAKGRID [12]. Convergence was deemed to be achieved when the force coefficients ceased to change by less than 0.01% over a subsequent 1000 iterations. Fully-structured multi-block meshes were constructed and tested for grid-independence of results with some local cell adaption for higher-resolution shock-capturing. A grid resolution in nose vicinity having local cell lengths of 0.030, 0.035 and 0.040 cm is used in this region to capture shock stand-off at Mach numbers of 5, 4 and 3 respectively. Grid clustering is also employed to capture bow shock at shock standoff distance. Some details of grids are as shown in Fig. 2. Freestream conditions at the inlet supersonic inflow are imposed near and as close
as possible to the foreshock location, an axisymmetric flowfield is studied with no-slip adiabatic conditions imposed on cone and pressure outflow condition was employed at the domain exit.

Figure 2. An 180° grid used to compute aerodynamic characteristics of complete projectile.

For the flight of Mach number 3 to 5 in air is considered with a specific heat ratio, $\gamma = 1.4$, Prandtl number, $\Pr = 0.9$, Gas constant, $R = 286.7$ J/kg K. Consequently, the freestream flow conditions are with; temperature $T_\infty = 300$ K, pressure $P_\infty = 1.0132 \times 10^5$ Pa and density $\rho_\infty = 1.1767 \times 10^{-4}$ kg/m$^3$. For Mach number 3 to 5, Reynolds’s number per unit reference length, is calculated to be of the order of $8.6 \times 10^6$ m$^{-1}$ to $14.3 \times 10^6$ m$^{-1}$.

2.3. Flow computations and solver
Computational fluid dynamics, CFD calculations are made using the Navier-Stokes solver, PAK-3D [5]. PAK-3D is an in-house developed solver and it has been validated against experimental data and other CFD methods [8], [10]. This method is for the block structured meshes and it treats fluid as an ideal gas using perfect gas law, Sutherland’s viscosity law and adiabatic heat transfer boundary conditions. Beam and Warming [13] approximation is used to discretize NS equations. Turbulent coefficient of viscosity was calculated by Baldwin-Lomax [14] algebraic turbulence model with modifications based on Reynold’s averaging assumptions. Numerical scheme follows approximate factorization formulation for the Navier-Stokes equations and correct differencing in transformation was retained by splitting of inviscid fluxes. Flux vectors got splitted according to the sign of the Eigen values. On attainment of convergence criteria, information of primitive variables at that iteration level got post processed for calculation of non-dimensional axial/circumferential pressure distributions as well as the static aerodynamic coefficients. All simulations were run as steady-state.

3. Results and discussions
3.1. Validation of numerical solutions
This section mainly presents the results based on the numerical simulation of flow investigation made for Mach 3 for the projectile’s blunted nose part at a fixed angle of attack. A general aim of comparison is to validate the numerical work for estimation of aerodynamic flow characteristics. One of the characteristic features of the aerodynamic flowfield pertinent to supersonic Mach number range is the computation of axial static pressure distribution on the nose surface. A comparison of computed results and experimental [14] axial static pressure distributions is made. The maximum variation of $p/p_\infty$, at an axial location, $x/L$ from 0.2 to 0.7, is determined to be less than 1%; as shown in Fig.3. In general, overall comparison of the computed results with the experimental data, showed similarity in terms of peak pressure distribution on the leeward side of the cone surface, upstream formation of bow shock and in an aft zone of low pressure.
Figure 3. Computed axial \( C_p \) distribution of projectile body at leeward side for Mach 4 and \( \alpha = 10^\circ \).

3.2. Computed shock-standoff distance

To understand the behavior of sonic line and bow shock, position of the bow shock at zero angle of attack for different Mach number is computed numerically. The separation distance between the nose and the shock mean thickness is referred as the shock standoff distance. The computed values are given in the Table 1. It is observed that with an increase in Mach number the tendency of the shock to come closer to the nose is evident. With an increase in positive angle of attack, it is noted that the bow shock moves closer to the body on the windward side, while the bow shock moves away from the conic body on the leeward side.

| Mach number | Shock-standoff distance [cm] |
|-------------|-----------------------------|
| 3.0         | 0.567                       |
| 4.0         | 0.461                       |

3.3. Axial centre of pressure, \( C_p \) distribution

\( C_p \) distribution was computed for Mach 3 and 4 along fin chord at mid-span locations. The values on leeward and windward sides of fin were calculated at 20 degrees angle of attack for Mach 4, as shown in Fig. 4. There is an increase in \( C_p \) value from leading edge to the center of the chord and then a decrease is observed till trailing edge of the fin on windward side. Negative values of \( C_p \) are observed at leeward side showing the suction condition, as shown in Fig.4.

Figure 4. \( C_p \) distribution along fin chord at mid-span location for \( M = 4 \) and \( \alpha = 20 \) degrees.
3.4. Circumferential $C_p$ distribution

Numerical simulations were also executed for an angle of attack of 8 degrees, to investigate the dispersion of circumferential $C_p$ distribution on the projectile. Axial locations of these planes are pre-selected at a distance of 7, 35 and 90 mm from the nose. Computed circumferential pressure distribution is shown in Fig. 5. As the bow shock moves away from the leeward side of the nose at an angle of attack of 8 degrees, the low pressure region grows as the axial distance on the nose surface is increased.

![Circumferential pressure distribution](image)

**Figure 5.** Circumferential pressure distribution at $M = 4$ and $\alpha = 10$ deg at axial stations of 0.7, 3.5, 12.0 and 5.61 cm from projectile’s nose.

3.5. Static aerodynamic coefficients of projectile

After satisfactory depiction of pressure distribution over the projectile at different Mach numbers and angle of attacks, post processing is done to compute the static aerodynamic coefficients. Normal force, pitching moment, Centre of pressure location and axial drag force coefficients are plotted for Mach 3 and 4. Variation of these coefficients with an angle of attack is shown in Fig. 6 to 8. The normal force coefficient increases with an increase in angle of attack, irrespective of the Mach number, as shown in Fig. 6. An increase in $C_{Na}$ evaluated at an angle of attack of 8 degrees, showed a rise of greater than 10% for both the Mach numbers. Forebody drag coefficient remains almost constant at a value of about 0.26 and 0.29, at angles of attack of 8 to 12 degrees for Mach 3 and 4 respectively. Slight change is observed with a further increase in angle of attack, as shown in Fig.6.

![Static aerodynamic coefficients](image)

**Figure 6.** Normal and Axial force coefficients at Mach 3.
A similar increasing effect in the pitching moment coefficient is also observed for both the Mach numbers, as shown in Fig. 7. Static stability remained about 0.409m to 0.429m for Mach 3 and 4 respectively, for an angle of attack of about 9 degrees. For a lower value of angle of attack, slightly less static stability was observed. Generally a blunted nose cone shape exhibit such a trend as its static stability is not much affected by a slight change in $\alpha$, at small angle of attacks; as shown in Fig.7.

3.6. Qualitative flowfield
Supersonic flowfield around a blunted nose cone consists of a stagnation point, a subsonic patch in the vicinity of the stagnation zone, which is followed by a supersonic conical region. Shock shape for the spherically blunted nose cone of the projectile is computed for Mach 3. Detailed features of the flowfield were obtained for the shock shape and the stagnation areas with a corresponding change in angles of attack. Representative pressure and Mach distribution for Mach 3 are as shown in Fig.10 and Fig.11, respectively.

4. Conclusions
A numerical simulation for a reference projectile’s nose in supersonic flow was performed to characterize and map the flowfield features. The numerical computation made possible to depict the characteristic features of blunted cone flowfield interactions, typically the formation of bow shock with associated stagnation region on the nose and resultant shock-standoff distance. All computed solutions adequately captured the pressure/ density variations in low pressure region. Further, study was made to perform computational aerodynamic investigations for the nose to determine its static aerodynamic coefficients under supersonic freestream conditions of Mach number 3 to 5 with an increasing angle of incidence from -4 to 20 degrees. Thus at this stage of the study a complete static aerodynamic behavior of the projectile’s nose configuration was evaluated under supersonic flow conditions. Though this predicted aerodynamic data needs further validation through wind tunnel experimentation.
References
[1] Massey K C McMichael J Warnock T and Hay F 2008 Development of mechanical guidance actuators for a supersonic projectile (The Aeronautical Journal) Vol. 112 No. 1130, pp: 181-195.
[2] Marx W J Wessling F C III Peters B R Thies A Strickland B R and Lianos D 2002 Miniature smart munitions/ guided projectiles for the objective force (In 23rd Army Science Conference)
[3] Spaid F W and Cassel L A 1973 Aerodynamic interference induced by reaction controls (McDonnell Douglas Corporation, USA) AGARDograph No. 173.
[4] Massey C A Guthrie K B and Silton S I 2005 Optimized guidance of a supersonic projectile using pin based actuators AIAA 2005-4966, (23rd AIAA Applied Aerodynamics Conf) (Toronto, Canada).
[5] Aerodynamics Division 1998 A user’s guide to PAK-3D (Aerodynamics and structural analysis centre) (Islamabad, Pakistan).
[6] Aerodynamics Division 1997 User’s manual–LOOK (Aerodynamics and structural analysis centre) (Islamabad, Pakistan).
[7] Kurita M Okada T and Nakamura Y 2001 The effects of attack angle on aerodynamic interaction due to side jet from a blunted body in supersonic flow. AIAA Paper 01-16173.
[8] Samar R Zahir S, and Khan M A 2010 Flight control using pin-protuberances for blunted cones (The Aeronautical Journal) Vol. 114, No. 1154.
[9] Zahir S Shabbar and Asif M 2005 Computational Aerodynamic Behaviour of a Plate/ Jet–Interaction with a Hypersonic Flowfield for a Biconic Configuration (The 8th Inter Symp on Fluid Control, Measurements and Visualization, 8FLUCOM) (PRC).
[10] Zahir S and Ye Z 2006 Computational Aerodynamic Interaction of a Short Protuberance /Lateral Plate on Blunted Cone Configurations in Hypersonic Flow AIAA-2006-3172 (24th Applied Aerodynamics Conference) (CA, USA).
[11] Zahir S and Ye Z Hypersonic 2007 Flow Interaction of Pitched Plates on Blunted Cone at Incidence (Inter Journal of Mathematical Models and Methods in Applied Sciences) Vol.1, Issue3, pp: 133–136.
[12] Aerodynamics Division 1997 PAK-GRID User’s Manual (Aerodynamics and Structural Analysis Centre) (Islamabad, Pakistan).
[13] Beam R M and Warming R P 1978 An Implicit factored scheme for the compressible Navier-Stoke equations (AIAA Journal) Vol.16, pp: 393-402.
[14] Baldwin B and Lomax H 1978 Thin-layer Approximation and Algebraic model for separated Turbulent flows; (AIAA paper 78-257).

Acknowledgements
The authors are grateful for the assistance of the Aerodynamics and Structural Analysis Centre, ASAC, Islamabad, Pakistan; for providing computational support.