Aerodynamics of Blunt Nosed Bodies

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Abstract

Blunt front side bodies are unpropitious for the supersonic flow, but however, they cannot be avoided in some applications. In this case an extremely strong shock wave, as it is well known, appears in front of the body, causing an important increase of both pressure and temperature in the vicinity of the stagnation point. An intensive drag force of the body is direct result of this shock wave. Mounting a spike on the rounded nose of a blunt body moving supersonically can significantly reduce the drag force. In this way, instead of one strong shock in frontal zone, appears a system of conical waves, so that the driving force, and consequently the fuel consumption, as well as the aerodynamic heating.

Keywords: **Blunt Nosed Bodies, Aerodynamics, Fluid Dynamics.**

Forces acting on bodies

The forces acting on the projectile are axial force, Normal force and Side force these forces are defined in terms of dimensionless coefficients as follows:

$F_x = C_x\, q\, S$, (axial force),

$F_y = C_y\, q\, S$, (side force),

$F_z = C_z\, q\, S$ (normal force)

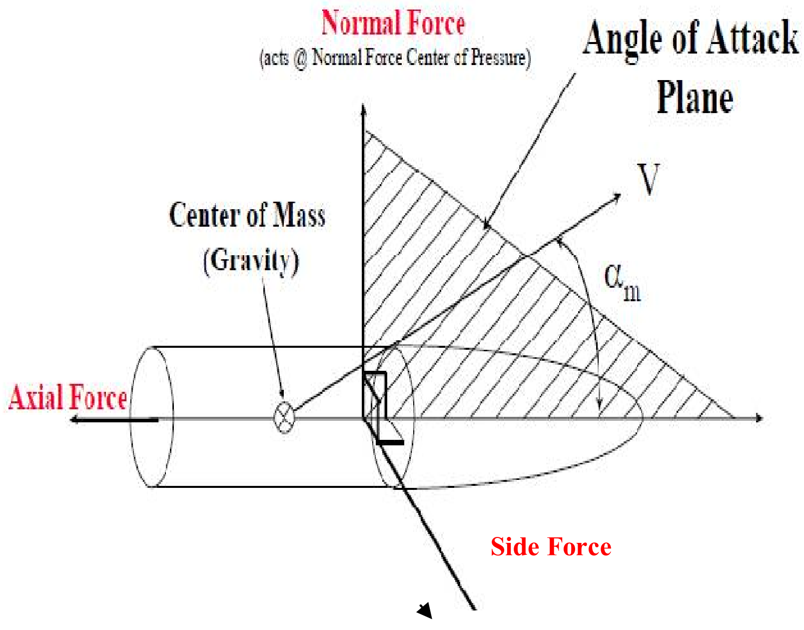

Figure 1.Forces acting on bodies

F_x, is axial force, F_y is Side force, F_z, Normal force, C_x, C_y, C_z are dimension less coefficients of axial, side, normal forces, q is dynamics pressure $0.5*\rho*V^2$, S is reference area ($\Pi*D^2/4$ D is Diameter of the blunt nosed body).

If we know the Normal force and axial force we calculate the Lift (L), and Drag (D) of the Missile using the following equations.

$$L = F_z \cos(\alpha) - F_x \sin(\alpha) \quad (1)$$

$$D = F_z \sin(\alpha) + F_x \cos(\alpha) \quad (2)$$

α is angle of attack

Aerodynamic moments acting on bodies

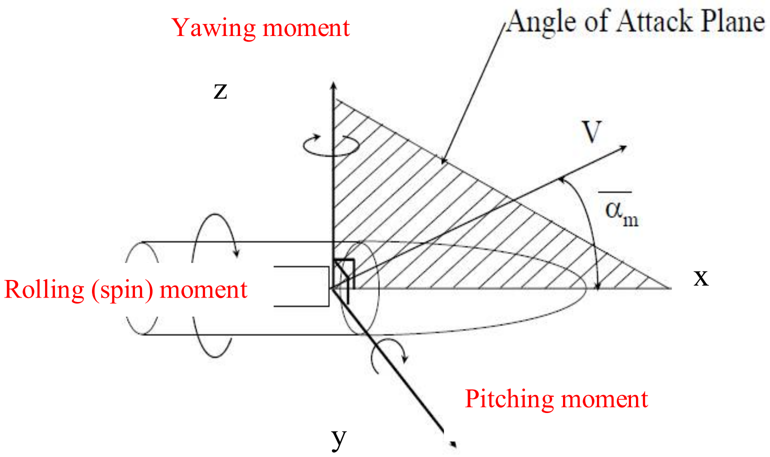

Figure 2.Moments acting on bodies

The components of aerodynamic moments are also expressed in terms of dimensionless coefficients

$L_{rm} = C_l\ q\ S\ D$ rolling moment (3)

$M_p = C_m\ q\ S\ D$ pitching moment (4)

$N = C_n\ q\ S\ D$ yawing moment (5)

L_{rm} is rolling moment, Mis pitching moment, N is yawing moment C_l, C_m, C_n are dimension less coefficients of rolling, pitching, yawing q is dynamics pressure $0.5*\rho*V^2$, S is reference area $(\Pi*D^2/4)$, D is reference length.

Aerodynamic coefficients are C_x, C_y, C_z, C_l, C_m, and C_n are primarily function of *Mach number (Ma), Reynolds number (Re), Angle of attack (α) and Sideslip angle (β).* The dimensionless coordinate of the center of pressure X_{cp} is defined here as:

$$X_{cp} = (M_p / F_n - d) / L \quad (6)$$

M_p is pitching moment with respect to the top point of the spike. F_n is the normal force, d is the diameter of the hemispherical body, L is the length of the Hemispherical body.

Supersonic flow around blunt bodies

Mach number greater than one the flow is called supersonic flow, supersonic flow over blunt bodies produces the shock waves and expansion waves.

Shock waves: shock wave (also called shock front or simply "shock") is a type of propagating disturbance. Like an ordinary wave, it carries energy and can propagate through a medium (solid, liquid or gas) or in some cases in the absence of a material medium, through a field such as the electromagnetic field. Shock waves are characterized by an abrupt, nearly discontinuous change in the characteristics of the medium. Across a shock there is always an extremely rapid rise in pressure, temperature and density of the flow. In supersonic flows, a shock wave travels through most media at a higher speed than an ordinary wave.

Shock waves can be:

- Normal shock: at 90° (perpendicular) to the shock medium's flow directions.
- Oblique shock: at an angle to the direction of flow.
- Bow shock: Occurs upstream of the front (bow) of a blunt object when the upstream velocity exceeds Mach 1.

Normal Shock Relations for a Perfect Gas: the ratios of the thermodynamic variables depend only upon the Mach number of the flow upstream of the shock, M_1. As with the isentropic relations, these equations also require the knowledge of the ratio of specific heats, γ, for a calorically perfect gas.

Normal shocks

- The normal shock wave (and the shock waves in general) is a sudden discontinuity for flow properties.
- By definition, a normal shock wave is a shock wave that is perpendicular to the flow. The flow velocity decreases across the normal shock wave; the flow is supersonic ahead of the normal shock wave and subsonic after the shock wave.
- The static pressure, temperature and density all increase across the normal shock wave
- If the flow is supersonic (relative to a moving vehicle), then $V_{inf}>a_{inf}$ so the sound waves can no longer propagate upstream ahead of the vehicle. Instead, they coalesce ahead of the vehicle, forming a thin shock wave.
- The shock wave is usually a few mean free path thick, say, 10^{-5} cm for air at standard conditions.
- The flow is adiabatic across the shock waves.

Normal shock relations: For a calorically perfect as $a^{*2} = u_1 u_2$ or $M_1^* M_2^* = 1$

So the mach number behind the normal shock is always subsonic. For calorically perfect gas with a given γ, M_2, ρ_2/ρ_1, p_2/p_1, and T_2/T_1 are functions of M_1 only:

$$M_2^2 = \frac{1+[(\gamma-1)/2]M_1^2}{\gamma M_1^2 - (\gamma-1)/2} \quad (7)$$

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)M_1^2}{2+(\gamma-1)M_1^2} \quad (8)$$

$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma+1}(M_1^2 - 1) \quad (9)$$

$$\frac{T_2}{T_1} = \frac{h_2}{h_1} = \left[1+\frac{2\gamma}{\gamma+1}\left(M_1^2-1\right)\right]\left[\frac{2+(\gamma-1)M_1^2}{(\gamma+1)M_1^2}\right] \quad (10)$$

$$s_2 - s_1 = c_p \ln\left\{\left[1+\frac{2\gamma}{\gamma+1}\left(M_1^2-1\right)\right]\left[\frac{2+(\gamma-1)M_1^2}{(\gamma+1)M_1^2}\right]\right\} - R\ln\left[1+\frac{2\gamma}{\gamma+1}\left(M_1^2-1\right)\right] \quad (11)$$

From the 2nd law, $s_2>s_1$ only if $M_1>1$. So the only physically possible $M_1 \geq 1$

For $M_1 \geq 1$ $\quad M_2 \leq 1, \frac{\rho_2}{\rho_1}, \frac{p_2}{p_1}, \frac{T_2}{T_1} \geq 1$

For Note that for thermally perfect gas, the changes across the normal shock wave depend on both M_1 and T_1. For chemically reacting gases, they depend on M_1, T_1 and ρ_1.

- For a stationary normal shock, the total enthalpy is constant across the shock wave.
- For calorically perfect gas, since $h = c_p T$, the total temperature is constant across the normal shock wave: $T_{o1} = T_{02}$
- For calorically perfect gas, the total pressure decreases across the normal shock:

$$\frac{p_{o_2}}{p_{o_1}} = e^{-(s_2 - s_1)/R} \quad (12)$$

If the shock wave is not stationary, neither the total enthalpy nor the total temperatures are constant across the shock wave.

Hugoniot Equation: The normal shock can be viewed as thermodynamic device that compresses gas following the Hugoniot equation:

$$e_2 - e_1 = \frac{p_1 + p_2}{2}(v_1 - v_2) \tag{13}$$

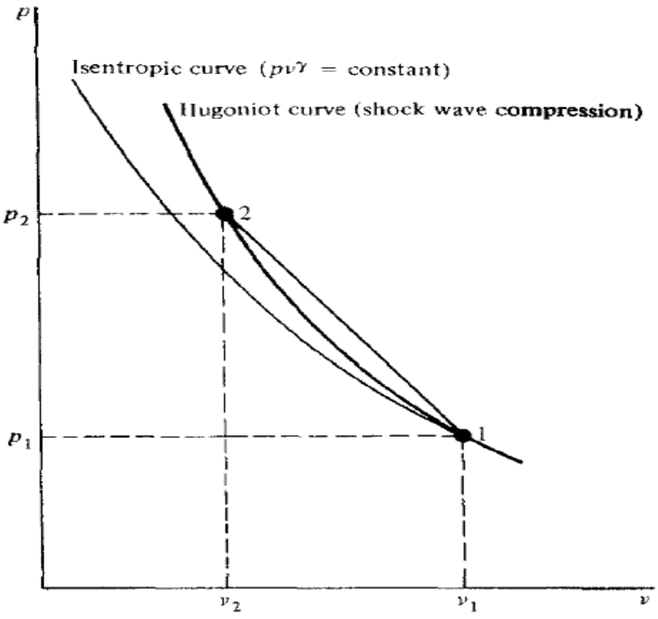

Figure 3.Hugoniot curve comparison with isentropic compression

Since e_1 is a function of p_1 and v_1 and e_2 is a function of p_2 and v_2, for given p_1 and v_1 upstream of a normal shock, each point on the Hugoniot curve (the p_2v_2 curve represented by the Hugoniot equation) represents a different shock with a different upstream M_1.

The basic oblique shock relations: The oblique shock waves typically occur when a supersonic flow is turned to itself by a wall or its equivalent boundary condition. In fig. 4 for instance, all the streamlines have the same deflection angle θ at the shock wave, parallel to the surface downstream the corner point A. Across the oblique shock, M decreases but p, T and r increase.

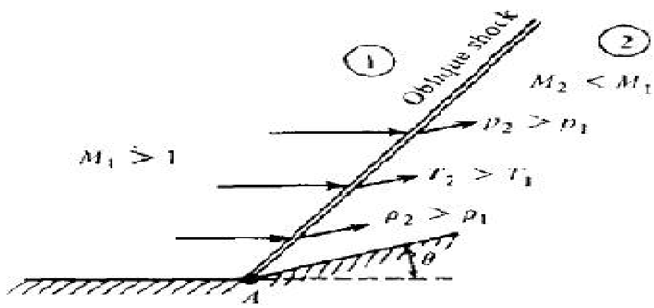

Figure 4.Oblique shock

Source of Oblique Waves: For a beeper moving at a supersonic speed, the beeper is always ahead of the sound wave fronts. This cause the sound wave fronts to coalesce into a line disturbance, called Mach wave, at the Mack angle m relative to the direction of the beeper. The physical mechanism to form the oblique shock wave is essentially the same as the Mach wave. The Mach wave is actually an infinitely weak shock wave.

$$\mu = \sin^{-1}\frac{1}{M}$$

Straight Oblique Shock Relations: The oblique shock tilts at a wave angle β with respect to V_1, the upstream velocity. Behind the shock, the flow is deflected toward the shock by the flow deflection angle θ. Let u and w denote the normal and parallel flow velocity components relative to the oblique shock and M_n and M_t the corresponding Mach numbers, we have for a steady adiabatic flow with no body forces the following relations:

Continuity:

$$\rho_1 u_1 = \rho_2 u_2 \qquad w_1 = w_2 \tag{14}$$

Momentum:

$$p_1 + \rho_1 u_1^2 = p_2 + \rho_2 u_2^2 \tag{15}$$

Energy:

$$h_1 + \frac{u_1^2}{2} = h_2 + \frac{u_2^2}{2} \qquad w_1 = w_2 \tag{16}$$

$$\frac{\rho_2}{\rho_1}, \frac{p_2}{p_1}, \frac{T_2}{T_1}$$

So and M_{n1} and M_{n2} all satisfy the corresponding normal shock relations, which are all functions of M_1 and β, because

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

θ -β-M Relation:

$$\tan\theta = 2\cot\beta\left[\frac{M_1^2\sin^2\beta - 1}{M_1^2(\gamma + \cos 2\beta) + 2}\right] \tag{17}$$

1. For any given free stream Mach number M_1, there is a maximum q beyond which the shock will be curved and detached.
2. For any given M_1 and $\theta<\theta_{max}$, there are two β's. *The larger β is called the strong shock solution*, where M_2 is subsonic. *The lower β is called the weak shock solution,* where M_2 is supersonic except for a small region near θ_{max}.
3. If θ =0, then β = Π/2 (normal shock) or β= μ (Mach wave)

For a calorically perfect gas the oblique shock relations are:

$$M_{n2}^2 = \frac{M_{n1}^2 + \left[2/(\gamma - 1)\right]}{\left[2\gamma/(\gamma - 1)\right]M_{n1}^2 - 1} \tag{18}$$

$$\frac{\rho_2}{\rho_1} = \frac{(\gamma+1)M_{n1}^2}{(\gamma-1)M_{n1}^2 + 2} \tag{19}$$

$$\frac{p_2}{p_1} = 1 + \frac{2\gamma}{\gamma+1}(M_{n1}^2 - 1) \tag{20}$$

$$\frac{T_2}{T_1} = \frac{p_2}{p_1}\frac{\rho_1}{\rho_2} \tag{21}$$

Bow shock on a blunt nosed body

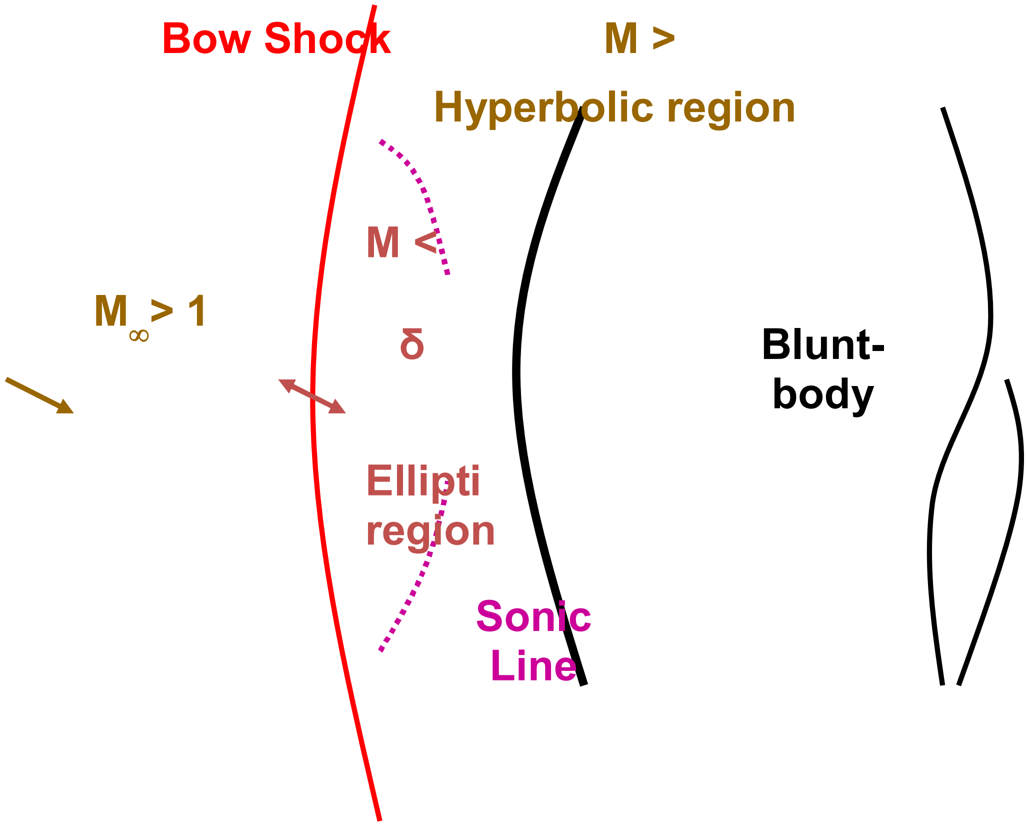

Figure 5.Bow shock on a blunt nosed body

- Blunt-nosed body designs are used for supersonic and hypersonic speeds (e.g. Apollo capsules and space shuttle) because they are less susceptible to aerodynamic heating than sharp nosed bodies.
- There is a strong, curved bow shock wave, detached from the nose by the shock detachment distance δ.
- Calculating this flow field was a major challenge during the 1950s and 1960s because of the difficulties involved in solving for a flow field that is elliptic in one region and hyperbolic in others.

Today's CFD solvers can routinely handle such problems, provided that the flow is calculated as being transient.

Expansion waves: Expansion fan is a centered expansion process, which turns a supersonic flow around a convex corner. The fan consists of an infinite number of Mach waves, diverging from a sharp corner. In case of a smooth corner, these waves can be extended backwards to meet at a point. Each wave in the expansion fan turns the flow gradually (in small steps). It is physically impossible to turn the flow away from itself through a single "shock" wave because it will violate the second law of thermodynamics. Across the expansion fan, the flow accelerates (velocity increases) and the Mach number increases, while the static pressure, temperature and density decrease. Since the process is isentropic, the stagnation properties remain constant across the fan.

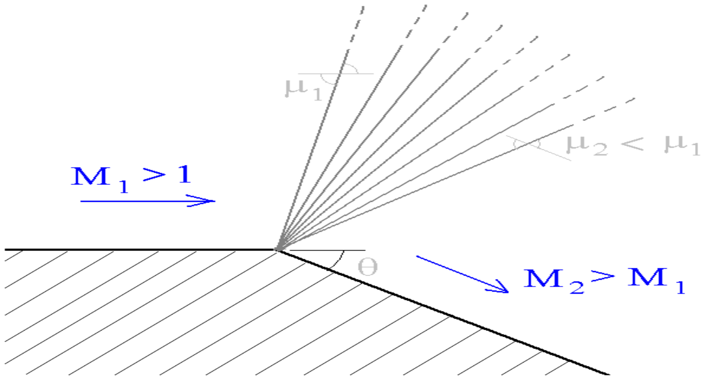

Figure 6.Expansion fan

Supersonic flow encounters a convex corner, it forms an expansion fan, which consist of an infinite number of expansion waves centered at the corner. The figure shows one such ideal expansion fan.

- Prandtl-Meyer Expansion Waves relations: $M_2 > M_1$. An expansion corner is a means to increase the flow mach number.
- $P_2/P_1<1$, $\rho_2/\rho_1<1$, $T_2/T_1< 1$. The pressure, density, and temperature decrease through an expansion wave.
- The expansion fan is a continuous expansion region, composed of an infinite number of Mach waves, bounded upstream by μ_1 and downstream by μ_2.
- A centered expansion fan, also called Prandtl-Meyer expansion wave. Where $\mu_1 = \sin^{-1}(1/M_1)$ and $\mu_2 = \sin^{-1}(1/M_2)$.
- Streamlines through an expansion wave are smooth curved lines.
- Since the expansion takes place through a continuous succession of Mach waves, and ds = 0 for each wave, the expansion is isentropic.

Contact and tangential discontinuities: Contact and tangential discontinuities are transition layers across which there is no particle transport. Thus, in the frame moving with the discontinuity, $v_{n1} = v_{n2} = 0$.

- *Contact discontinuities* are discontinuities for which the thermal pressure and the velocity are continuous. Only the mass density and temperature change.
- *Tangential discontinuities* are discontinuities for which the total pressure is conserved. The normal component of the magnetic field is identically zero. The density and thermal pressure can be discontinuous across the layer.

References

1. Milcev SS, Pavlovic MD. Influence of Spike Shape at Supersonic Flow Past Blunt-Nosed Bodies. University of Belgrade, 11000 Belgrade, Serbia, Yugoslavia publication, 29 January 2002. Copyright 2002 by the American Institute of Aeronautics and Astronautics, Inc .
2. Nagdewe SP, Vaghela MB, Shevare GR. Comparison of turbulence models for the Simulation of Internal and External flows. IIT Bombay, India. Proceedings of the XXVII Iberian Latin American Congress on Computational methods in Engineering, 2006.
3. Spalart P, Allarmas A. A one equation turbulent model for Aerodynamic flows AIAA paper 92-0439, 1992.
4. Blazek J. Computational Fluid Dynamics: Principles and Applications, 2001.