EFFECT OF ANGLE OF INCIDENCEON ROLL DAMPING DERIVATIVE OF A DELTA WING

Asha Crasta [1] and S. A. Khan [2]

1. Research Scholar, Department of Mathematics, Jain University, Bangalore, Ph: 9164495329
2. Principal, Bearys Institute of technology, Mangalore, Ph: 9845531478.

ABSTRACT: In the Present paper effect of angle of incidenceon roll damping derivative of a delta wing with curved leading edges with attached shock case has been studied. A Strip theory is used in which strips at different span wise location are independent of each other. This combines with similitude to give a piston theory. The Present theory is valid only when the shock wave is attached with the leading edge of the wing. Effects of secondary wave reflection and viscosity have not been taken into account. Effect of leading edge bluntness also has been neglected. Results have been obtained for hypersonic flow of perfect gases over a wide range of angle of attack and Mach number.

Keywords: Angle of incidence, Attached shock wave, Curved leading edges, delta wings, Hypersonic, Roll damping derivative, Sweep angle

Corresponding Author: Asha Crasta

INTRODUCTION

The analysis of hypersonic flow over flat deltas (with straight leading edge and curved leading edge) over a considerable incidence range is of current interest with the advent of space shuttle and high performance military aircrafts. The knowledge of aerodynamic load and stability for such types there is a need for simple but reasonably accurate methods for parametric calculations facilitating the design process. The dynamic stability computation for these shapes at high incidence (which is likely to occur during the course of reentry or maneuver) is of current interest. When descending shock waves which are usually strong and can be either detached or attached.

Pike [1] and Hui [2] have given theories for steady delta wings in supersonic/hypersonic flow with attached shocks. For 2-D flow exact solutions were given by Carrier [3] and Hui [4] for the case of an oscillating wedge and by Hui (1978) for an oscillating flat plate, which is valid uniformly for all supersonic Mach numbers and wedge angles or angles of attack with attached shock wave. Hui [5] calculated pressure on the compression side of a flat delta.

The role of dynamic stability at high incidence during re-entry or maneuver has been pointed out by Orlik-Ruckemann [6]. The shock attached relatively high aspect ratio delta is often preferred for its high lift to drag ratio.

Hui and Hemdan [7] have studied the unsteady shock detached case in the context of thin shock layer theory. Liu and Hui [8] have extended Hui's [5] theory to a shock attached delta wing in pitch. Light hill [9] has developed a "Piston Theory" for oscillating airfoils at high Mach numbers. A parameter δ is introduced, which is a measure of maximum inclination angle of Mach wave in the flow field. It is assumed that M_∞ δ is less than or equal to unity (i.e. $M_\infty \delta \le 1$) and is of the order of maximum deflection of a streamline. Light hill [9] likened the 2-D unsteady problem to that of a gas flow in a tube driven by a piston and termed it "Piston Analogy".

Ghosh [10] has developed a large incidence 2-D hypersonic similitude and piston theory. It includes Light hill's [9] and Mile's [11] piston theories. Ghosh and Mistry [12] have applied this theory of order of ¢^{2} where ¢ is the angle between the attached shock and the plane approximating the windward surface. For a plane surface, ¢ is the angle between the shock and the body. The only additional restriction compared to small disturbance theory is that the Mach number downstream of the bow shock is not less than 2.5.

Ghosh [13] has obtained a similitude and two similarity parameters for shock attached oscillating delta wings at large incidence. Crasta & Khan [14] have extended the Ghosh similitude to supersonic flows past a planar wedge. Crasta & Khan have obtained stability derivatives in pitch and roll of a delta wing with curved leading edges for supersonic flows [15] and Hypersonic flows [16]. Crasta & Khan have studied the effect of angle of incidence on pitching derivatives of a delta wing with curved leading edges of an attached shock case [17]. In the present analysis the effect of angle of incidence on the roll of damping derivative for hypersonic flows has been studied and results are obtained. The pressure on the lee surface is assumed zero.

Analysis:

To get a curved leading edge we superpose a full sine wave and or half sine wave on a straight leading edge. X-axis is taken along the chord of the wing and the Z-axis is perpendicular to the chord in the plane of the wing.

Equation of x-axis is z = o

Equation of full and half sine wave are $Z = -a_F \sin(\frac{2\pi x}{c})$

And $Z = -a_H \sin\left(\frac{\pi x}{c}\right)$ (1)

Equation of straight L.E $Z = x \cot \in$ (2)

Where a_F & a_H are the amplitudes of the full & half sine waves and c is chord length of the wing. Hence the equation of the curved leading edge is

$$Z = x \cot \in - a_F \sin\left(\frac{2\pi x}{c}\right) - a_H \sin\left(\frac{\pi x}{c}\right) \tag{3}$$

Area of the wing:

Area ABD = $\int_0^C Z dx$, Let $k = \frac{\pi}{c}$.

Hence, the wing Area $= C^2(\cot \in - \frac{4A_H}{\pi})$

Strip theory:

A thin strip of the wing, parallel to the centerline, can be considered independent of the z dimension when the velocity component along the z direction is small. This has been discussed by Ghosh's [13]. The strip theory combined with Ghosh's large incidence similitude leads to the "piston analogy" and pressure P on the surface can be directly related to equivalent piston Mach number Mp. In this case both Mp and flow deflections are permitted to be large. Hence light hill piston theory [9] or miles [11] strong shock piston theory cannot be used but Ghosh's piston theory will be applicable.

$$\frac{P}{P_\infty} = 1 + AM_P{}^2 + AM_P(B + M_P{}^2)^{\frac{1}{2}} \quad \text{Where } P_\infty \text{ is free stream pressure…} \tag{4}$$

Pitching moment derivatives:

Let the mean incidence be α_0 for the wing oscillating in pitch with small frequency and amplitude about an axis X_0. The piston velocity and hence pressure on the windward surface remains constant on a span wise strip of length 2z at x, the pressure on the lee surface is assumed to be zero. Therefore, the nose up moment is given by

$$m = -2\int_0^c p.z.(x - x_0)dx \tag{5}$$

Rolling Moment Derivative due to Rate of Roll:

Let the roll be p and rolling moment be L, defined according to the right hand system of reference

$$\therefore L = 2\int_0^c \left(\int_0^{Z=f(x)} pzdz \right) dx \tag{6}$$

The local piston Mach number normal to the wing surface is given by

$$(M_P) = (M_\infty)\sin\alpha_0 \frac{z}{a_\infty} \bar{p} \quad (7)$$

The roll-damping derivative is non-dimensionalised by dividing with the product of dynamic pressure, wing area, and span and characteristic time factor

$$\therefore \quad -C_{l_p} = \frac{1}{\int_\infty U_\infty C^3 b(\cot \in - \frac{4A_H}{\pi})} C^3 \tag{8}$$

$$-C_{l_p} = \frac{\sin\alpha_0 f(S_1)}{(Cot^2\varepsilon - \frac{4A_H}{\pi}\cot\varepsilon)}[\frac{\cot^3 \in}{12} + \cot^2 \in \frac{A_F}{2\pi} - \frac{A_H}{\pi^3}(\pi^2 - 4) +$$

$$\frac{1}{4}\cot \in (A_F{}^2 + A_H{}^2) - \frac{4}{9\pi}A_H{}^3 - \frac{16}{9}\frac{A_F A_H}{\pi^2} - \frac{16}{15\pi}A_F{}^2 A_H] \tag{9}$$

.

Where $S_1 = M_\infty \sin\alpha_0$

$$f(S_1) = \frac{(\gamma+1)}{2S_1}\left[\ 2S_1 + \frac{(B+2S_1^2)}{(B+S_1^2)^{\frac{1}{2}}}\ \right]$$

RESULTS AND DISCUSSIONS:

Primary objective of this investigation is to study the effect of angle of attack and Mach number on the roll damping derivative of delta wing with curved leading edge. Since, in case of fighter aircraft during the dog fight roll damping derivatives plays an important role.

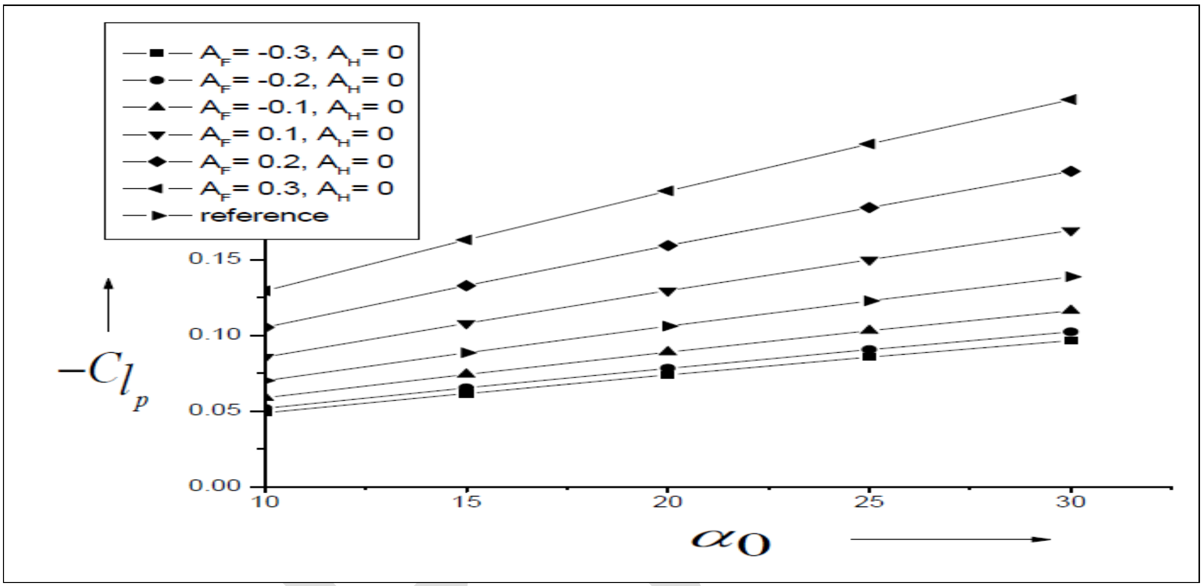

Fig 1:variation of rolling moment derivative with angle of incidence of a full sine wave(M=5)

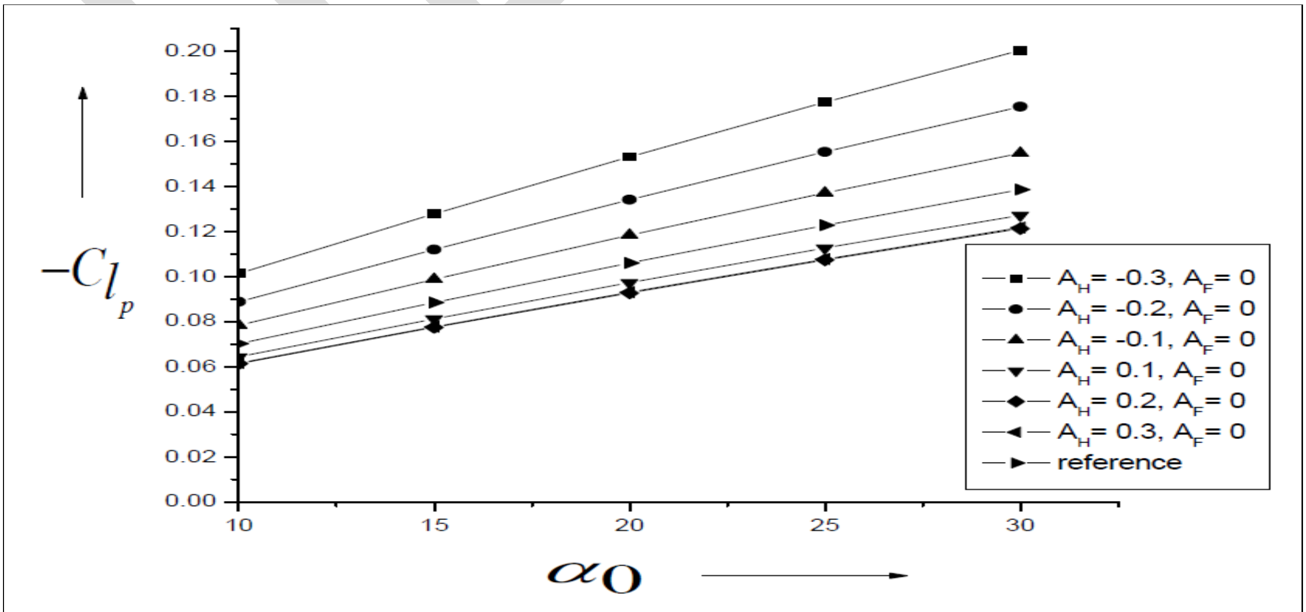

Fig. 2: Variation of roll damping derivative with angle of attack for a half sine wave(M=5)

From Fig. 1, it is seen that for a fixed value of Mach number M = 5, the roll damping derivative increases linearly with angle of attack for all the amplitudes of full sine wave of the present study. Further, it is found that in case of full sine wave when A_F = 0.1, 0.2, and 0.3, there is an increase in the area of the wing near the leading edge where as at the trailing edge the area will decrease but when A_F = -0.1,-0.2 , and - 0.3, the wing area at the leading will decrease and at the trailing edge it will increase due to shift in the area of the wing, which may be the reasons for this change in the trend of the roll damping derivative as well asthe change in the magnitude of the roll damping derivative for full sine wave when the amplitude of the full sine wave is positive and negative, respectively.

Fig. 2 shows the similar results forroll damping derivative as a function of angle of attack at Mach number M = 5. However, when these results are compared with that of Fig. 1 as shown above, it is seen that the magnitude is less compared to the full sine wave. The reasons for this trend may be due to the magnitude of area being shifted towards the leading edge and the trailing edge, respectively. In case of half sine wave there is overall reduction or increment in the wing area when its shape changes from concave to convex. It is also, seen that there is decrease in the roll damping derivative by 20 percent, for either of the case when the shape of the wing is changing from concavity to convexity which; implies that from the static and dynamic stability point of view the wing geometry having full sine wave superimposed on the leading edge of a delta wing will result in better performance in stability and control as compared to the wings having half sine wave at the leading edge.
Fig. 3 also presents results of roll damping Vs angle of attack for a fixed value A_F = 0.1 and Mach number = 5, here, the value of the half sine wave has been varied from ± 0.1 to 0.3 in steps of 0.1 to ascertain the combined effect of the variation of the amplitude of the half sine wave and a full sine wave with a fixed value of the amplitude. As we can see from the results that the trendis mixed one, however, there is a 20 percent increase in the roll damping derivative on the either side. This implies that from the wing dynamic stability performance point of view this combination will be better than what we have seen in figure 2.

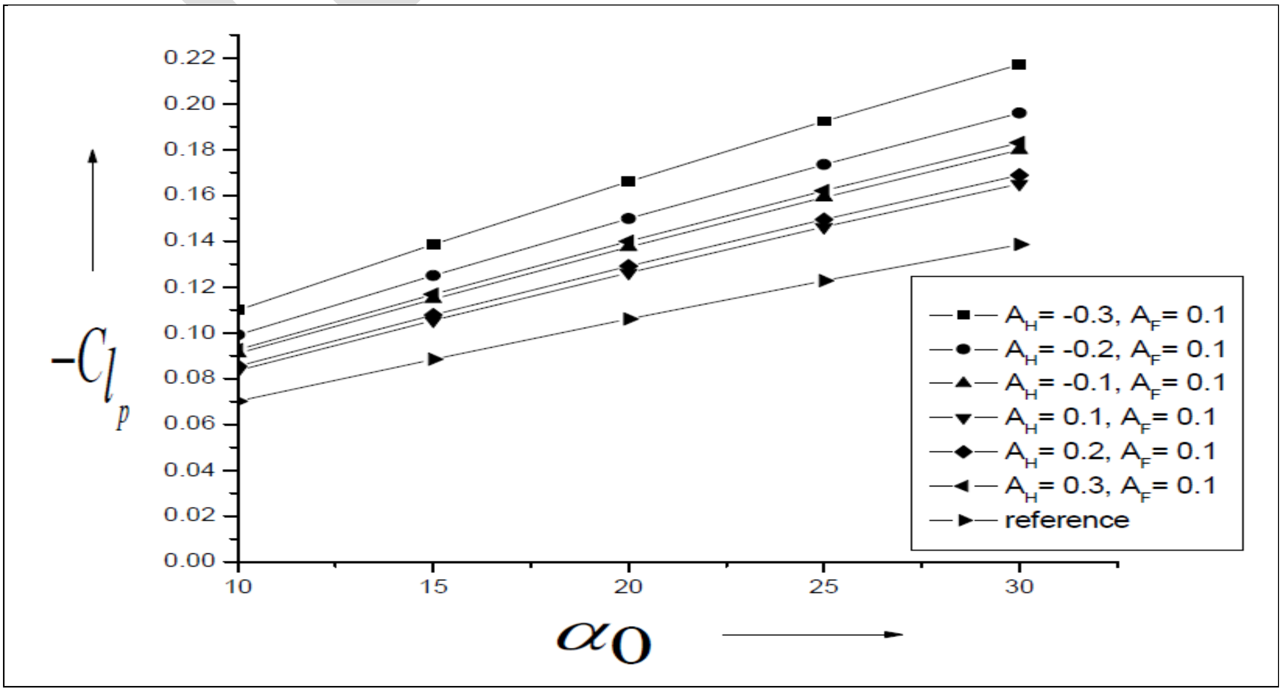

Fig.3:variation of roll damping derivative with angle of incidence of a half sine wave(M=5)

Fig. 4 presents the results for roll damping derivatives as a function of angle of attack for fixed amplitude of half sine $A_H = 0.1$ and Mach number M = 5, and the amplitude of full sine wave has been varied from ± (0.1, 0.2, and 0.3). From the results it is seen that there is an increase as well as decrease of 20 percent in the roll damping derivative from its previous value when A_H was zero only the amplitude of full sine wave was varied. The reasons for this variation in the magnitude of the roll damping derivative is due to the change in the effective plan form area of the wing.

Roll damping derivatives results for Mach number M = 10, with zero amplitude of half sine wave and with varying amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3) are shown in Fig. 5. It is seen that roll damping derivative is varying linearly with angle of attack as expected, however, if we compare these results with Fig.1 then we notice that all the parameters are same except there is a change in the Mach number, and it is well known that with the increase in the Mach number the roll damping derivative will decrease, hence, the results are on the expected line. It is also, seen that there is decrease in the magnitude of roll damping derivatives by 15 percent and 10 percent for negative and positive amplitude of the full sine wave. Similar results for roll damping derivative as a function of angle of attack are shown in Fig. 6, for variable amplitude of half sine wave in the range ± (0.1, 0.2, and 0.3) for Mach number M = 10. It is seen that the magnitude has come down drastically when compared with Fig. 2 keeping all the parameters same, only the Mach number has increased from M = 5 to 10.

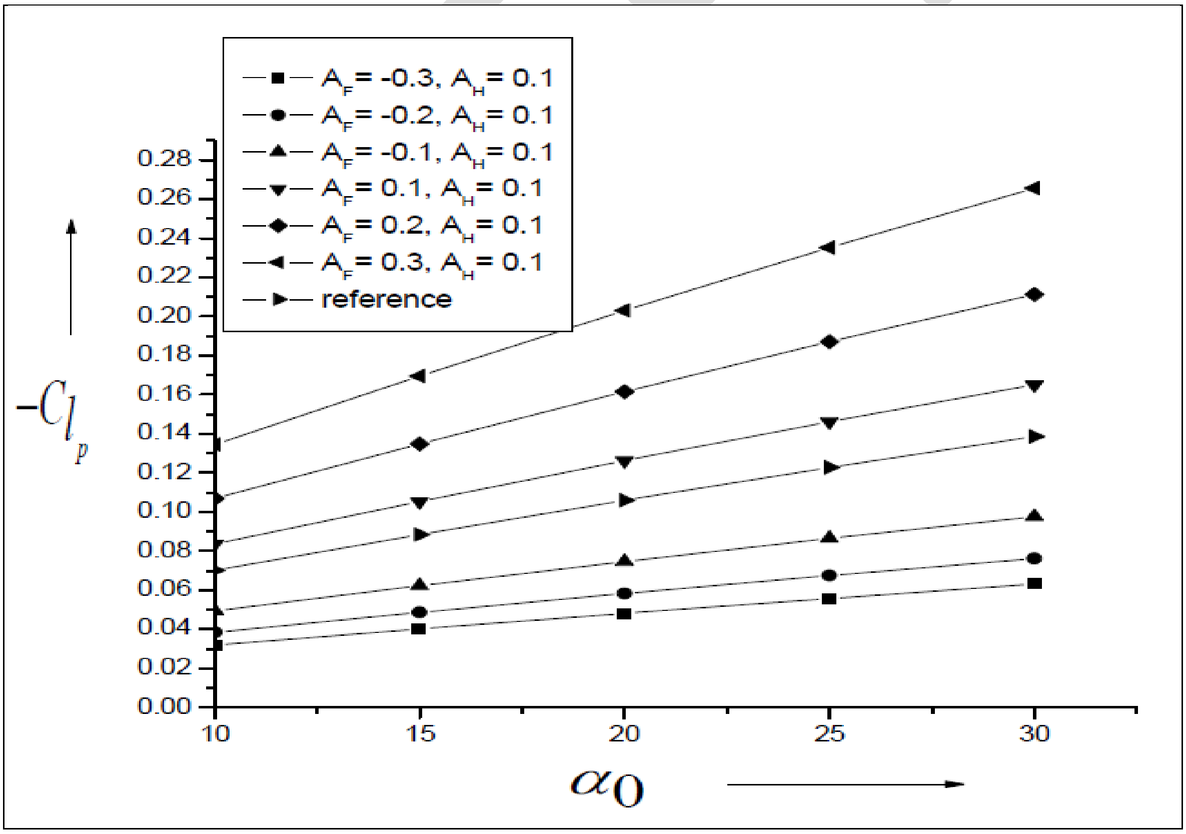

Fig. 4: variation of roll damping derivative with angle of attack for a half sine wave (M=5)

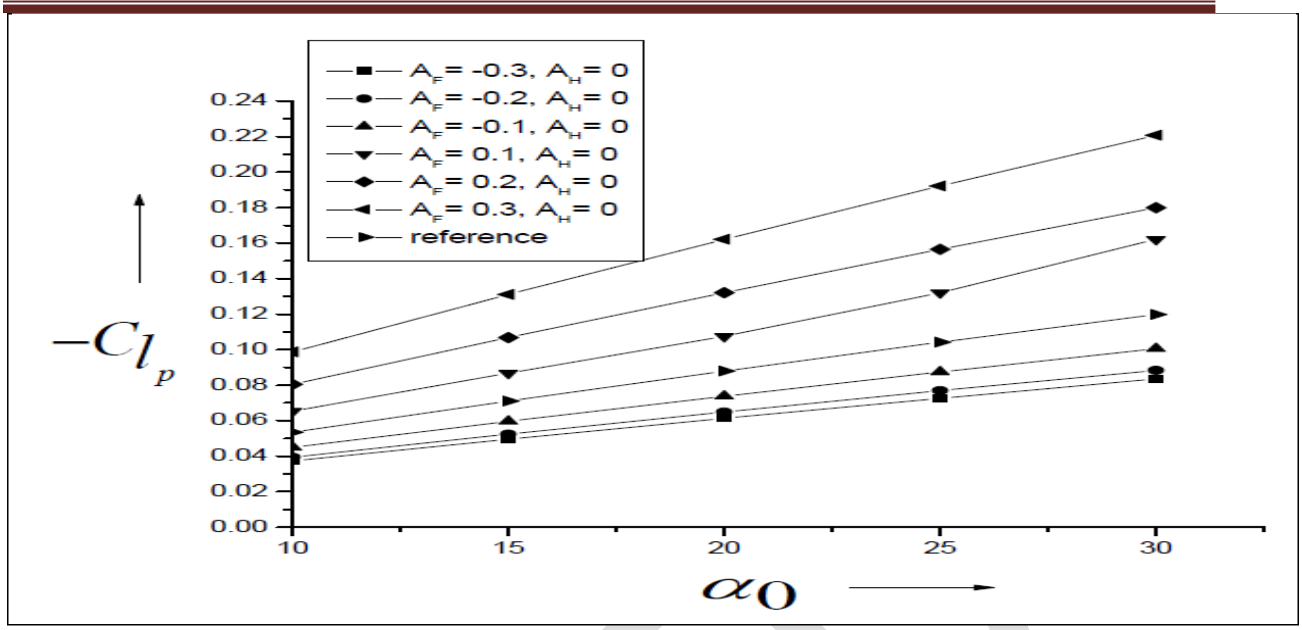

Fig.5:variation of roll damping derivative with angle of attack for a full sine wave (M=10)

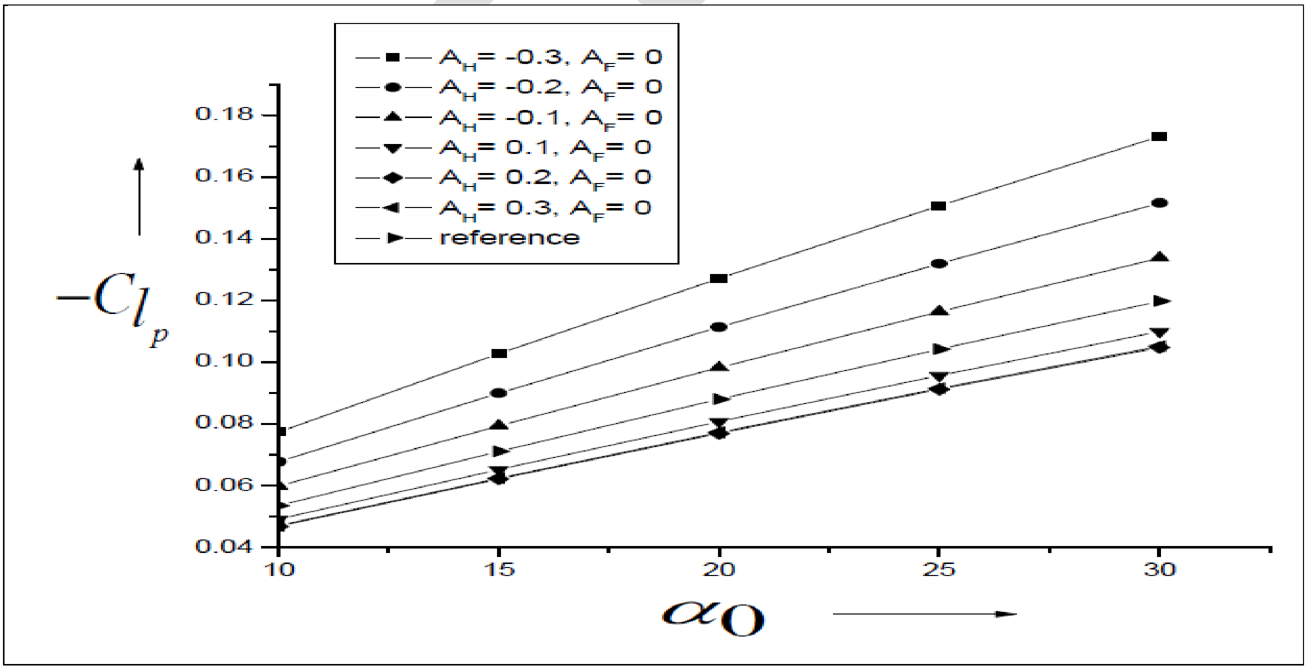

Fig. 6:variation of roll damping derivative with angle of attack for a half sine wave (M=10)

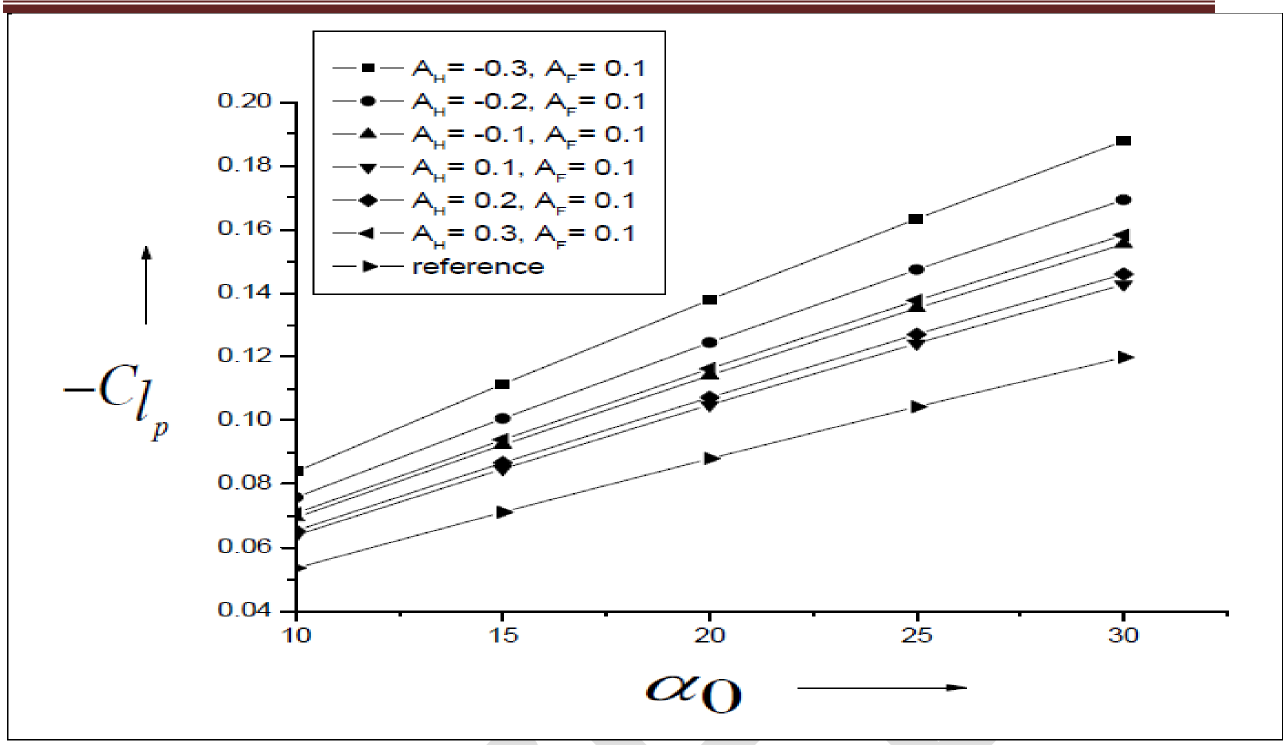

Fig. 7:variation of roll damping derivative with angle of attack for a half sine wave (M=10)

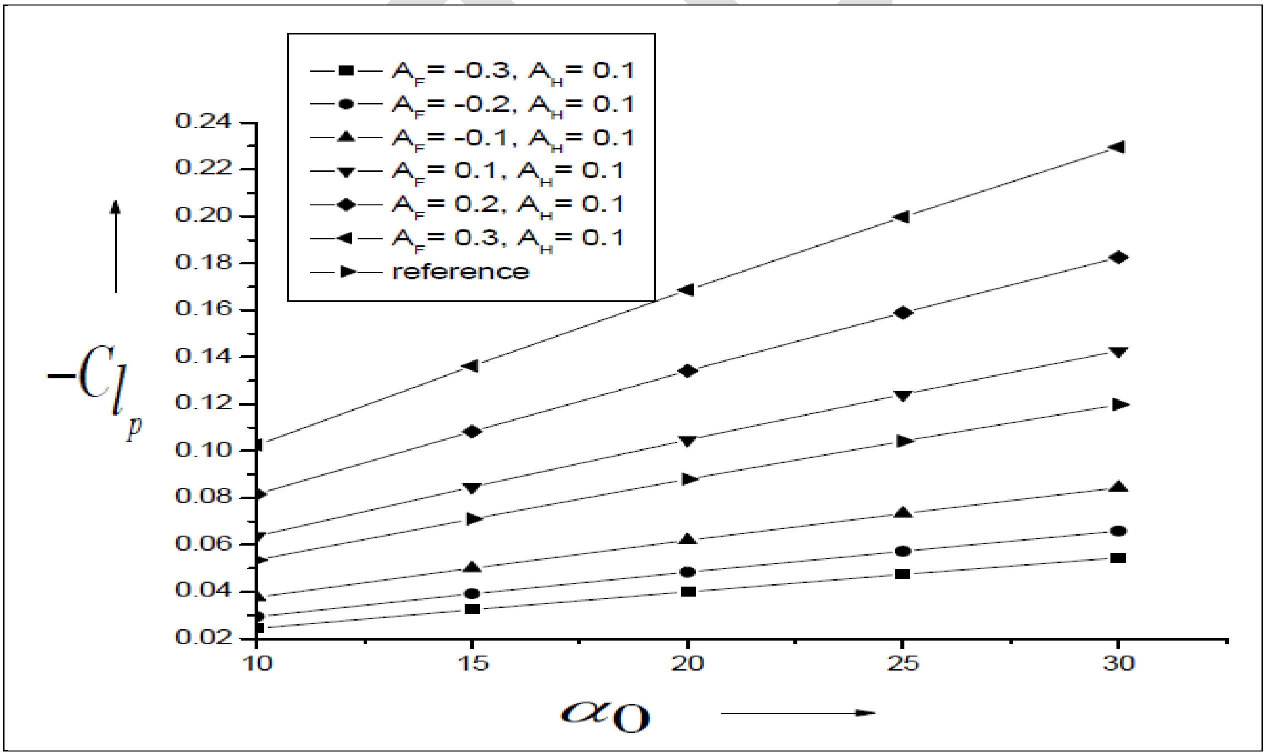

Fig.8:variation of roll damping derivative with angle of attack for a full sine wave (M=10)

Fig. 7 present the results for roll damping derivatives for fixed value of amplitude $A_F = 0.1$ of full sine wave with variable amplitude of half sine wave in the range ± (0.1, 0.2, and 0.3) at Mach number M = 10. From the figure it is seen that for fixed amplitude value of full sine wave when half sine wave is superimposed all the values of the roll damping derivatives are above the values which were obtained for straight leading edge. The physical reason for this behavior is that there is a variation in the amplitude of the half sine wave in the range ± (0.1, 0.2, and 0.3) which adds convexity or concavity, to the basic configuration of the wing geometry and due to this change in the geometry the change in the trend is observed.

Fig. 8 shows the results roll damping derivatives for Mach number M = 10, fixed value of amplitude of half sine wave for various amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3) for angle of attack in the range from 10 degrees to 30 degrees. From the figure it is found that the roll damping derivative attains maximum value at amplitude of 0.3 then there is linear decrease in the magnitude and this continues for negative amplitude as well due to the decrease in the plan form area of the wing.

Fig. 9 presents the roll damping derivatives for Mach number M = 15, for various amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3), having zero amplitude of half sine wave. From the results it is seen that there is drastic reduction in the roll damping derivatives and this amounts to be around 50 percent reduction in the values as compared to that for Mach number M = 10, keeping all other parameters same. These results clearly demonstrate the dependency of the roll damping derivative on Mach number.

Similar results for roll damping derivatives are seen in Fig. 10, for Mach number M = 15, keeping amplitude of full sine wave as zero and varying the amplitude of half sine wave in the range. Fig. 9 presents the roll damping derivatives for Mach number M = 15, for various amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3), having zero amplitude of half sine wave. Further, it is found that this configuration results in reduction of the roll damping derivative by 50 percent, and very clearly reflect the dominance of the Mach number.

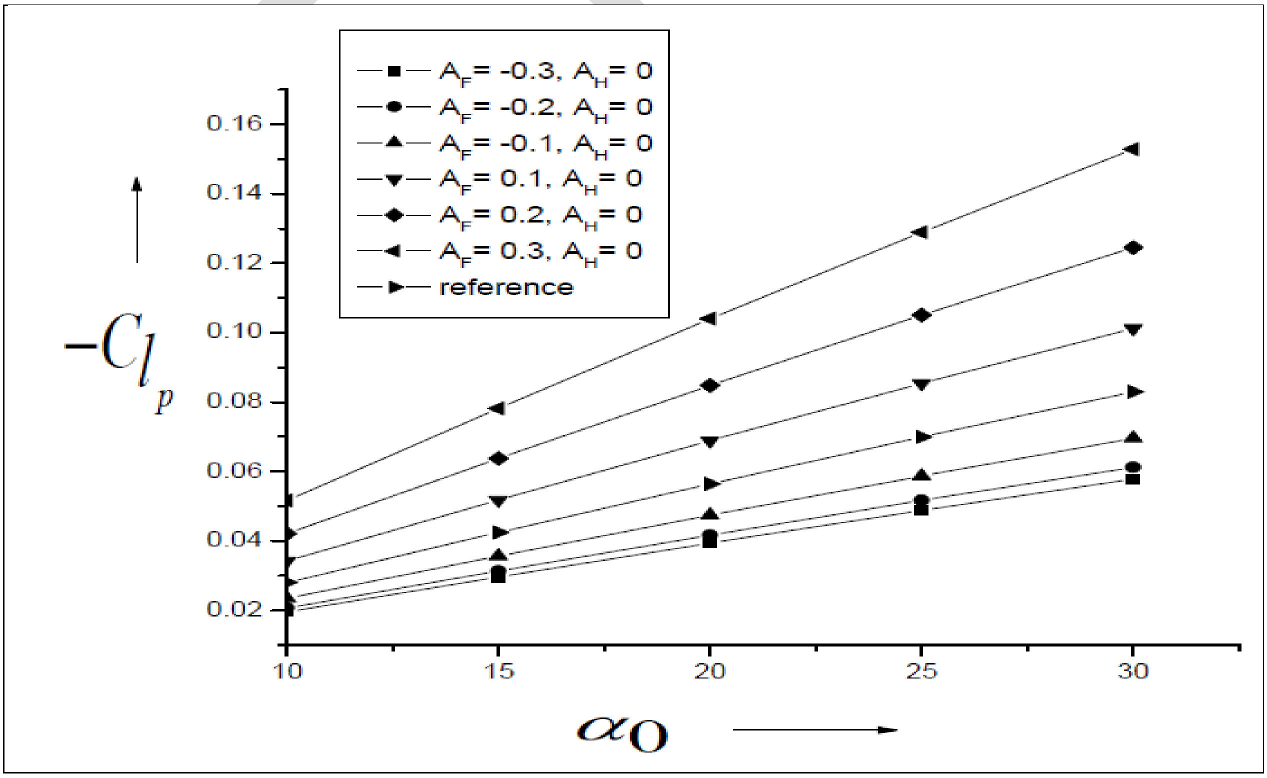

Fig. 9: variation of roll damping derivative with angle of attack for a full sine wave (M=15)

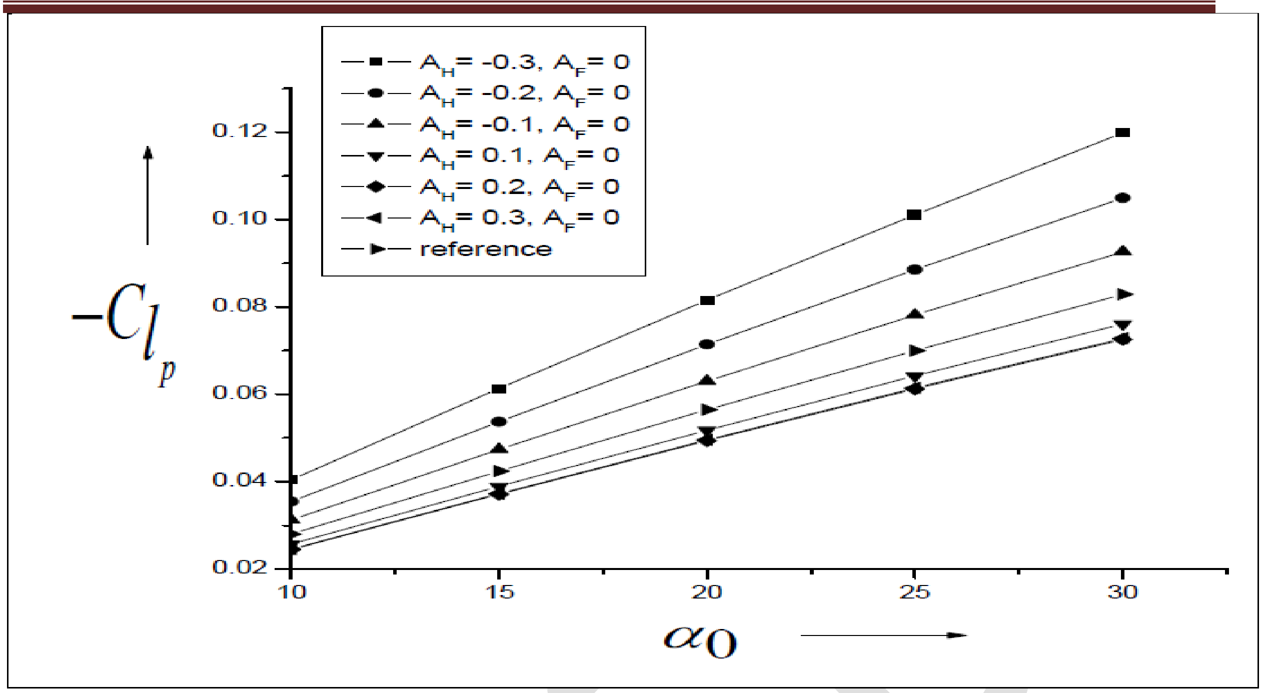

Fig. 10:variation of roll damping derivative with angle of attack for a half sine wave (M=15)

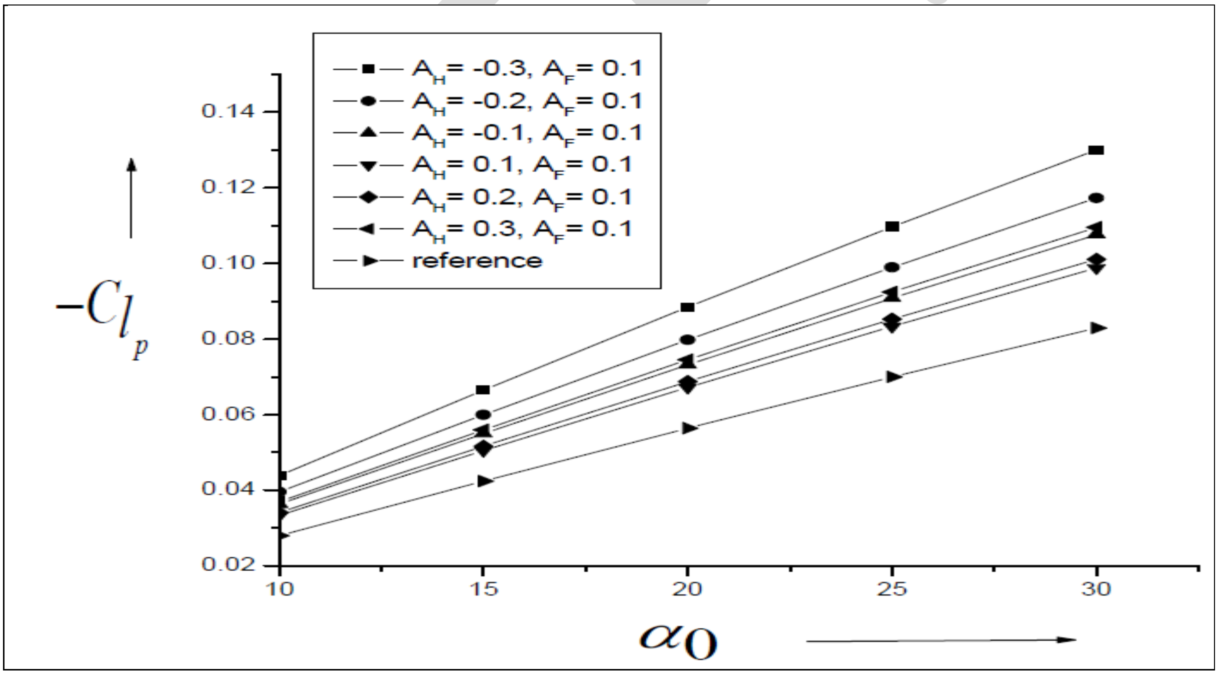

Fig. 11: variation of roll damping derivative with angle of attack for a half sine wave (M=15)

Fig. 11 presents the results of roll damping derivative at Mach number M = 15, for fixed value of amplitude of full sine wave as 0.1 and varying the amplitude of the half sine wave in the range ± (0.1, 0.2, and 0.3), whereas, Fig. 9 presents the roll damping derivatives for Mach number M = 15, for various amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3), having zero amplitude of half sine wave. In this case it is found that the roll damping derivative is being increased by another 40 percent, with respect to the previous case. This change in the damping derivative is because of increase in the plan form area. Similar results are seen in Fig. 12 for constant amplitude value of $A_H = 0.1$, and variable amplitude of full sine wave in the range ± (0.1, 0.2, and 0.3), for positive amplitude the maximum gain is around 40 percent and for the negative amplitude the reduction in the roll damping derivative is also around 40 percent.

Figs. 13 and 14 present the result of roll damping derivative at Mach number M = 20, when amplitude of half sine wave is zero and amplitude of full sine wave is varying in the range ± (0.1, 0.2, and 0.3) as shown in Fig. 13, for the case of full sine wave the amplitude is zero and half sine wave amplitude is varying in the range ± (0.1, 0.2, and 0.3) (Fig. 14), there is a significant increase in the values. If we compare the results of Figs. 15 and 16 with that of Fig. 13 & 14, it is found that in Figs. 13 & 14, there is a marginal change in the values.

Figs. 17 to 20 present the result of roll damping derivative at Mach number M = 25, when amplitude of half sine wave is zero and amplitude of full sine wave is varying in the range ± (0.1, 0.2, and 0.3) as shown in Fig. 17 and for the case of full sine wave the amplitude is zero and half sine wave amplitude is varying in the range ± (0.1, 0.2, and 0.3) (Fig. 18), there is a significant increase in the values. If we compare the results of Figs. 19 and 20 with that of Figs. 17 & 18, it is seen that in Figs. 17 & 18, there is a marginal change in the values. These changes are due to the change in the wing plan form area otherwise for these Mach number change in the values is not expected as Mach number independence principle will hold in such cases.

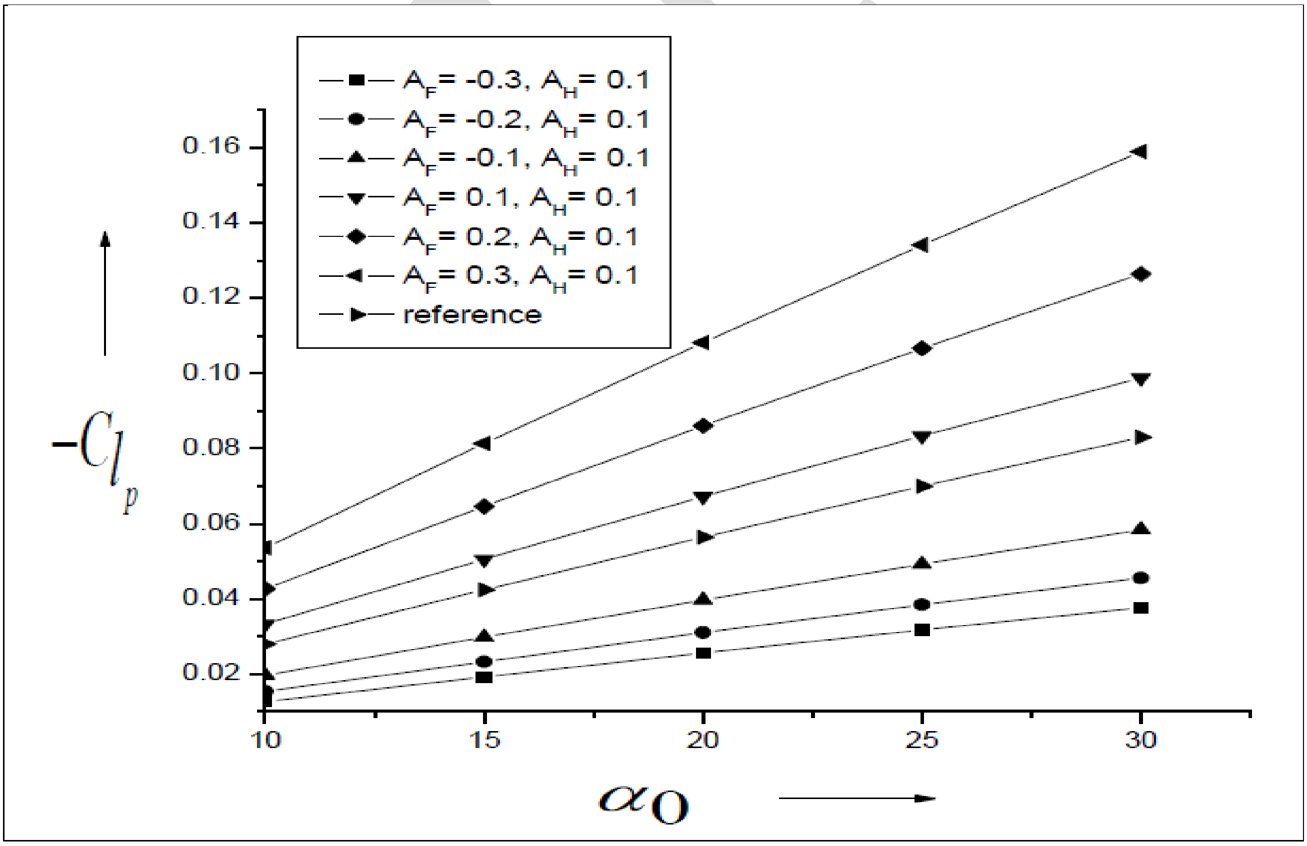

Fig. 12: variation of roll damping derivative with angle of attack for a full sine wave (M=15)

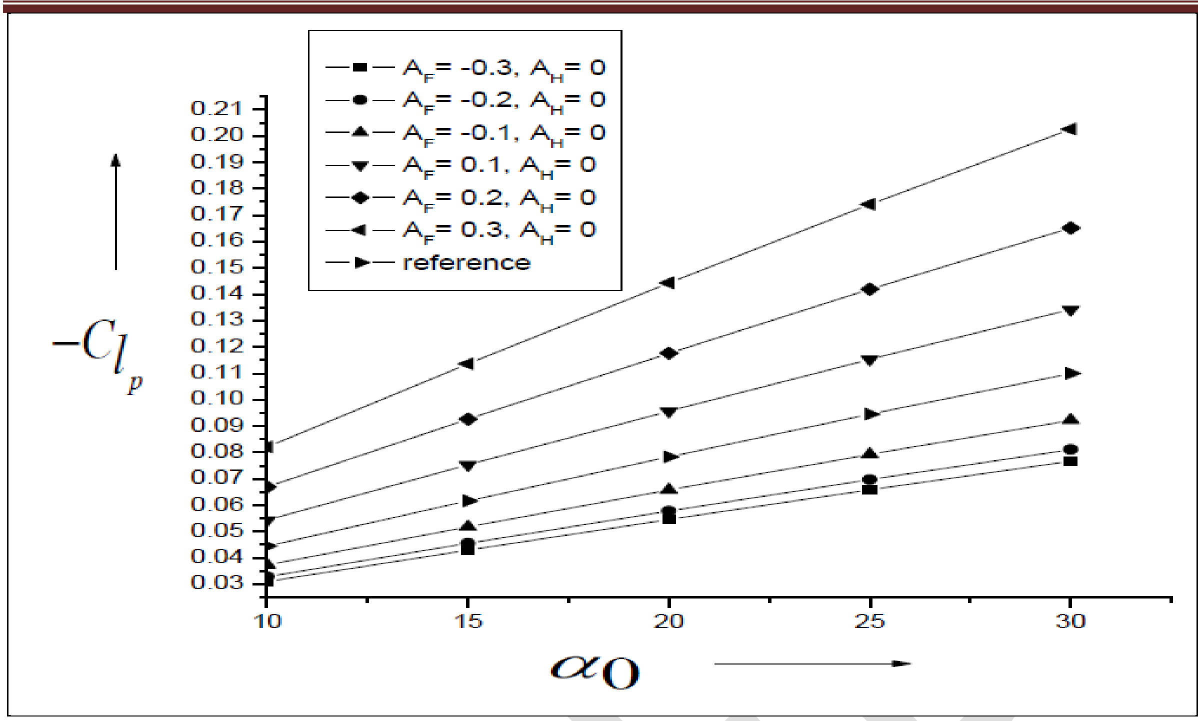

Fig.13: variation of rolling moment derivative with angle of attack for a full sine wave(M=20)

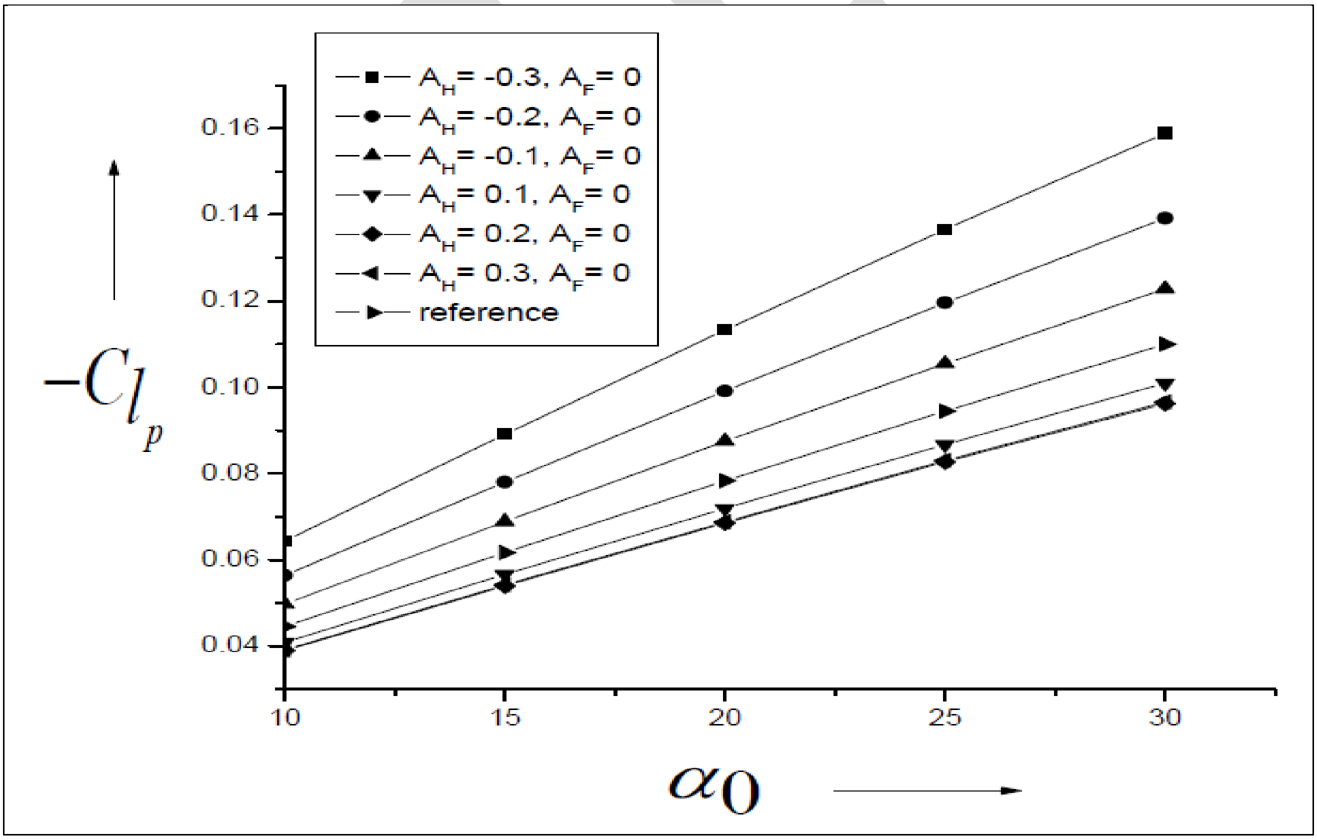

Fig. 14:variation of roll damping derivative with angle of attack for a half sine wave (M=20)

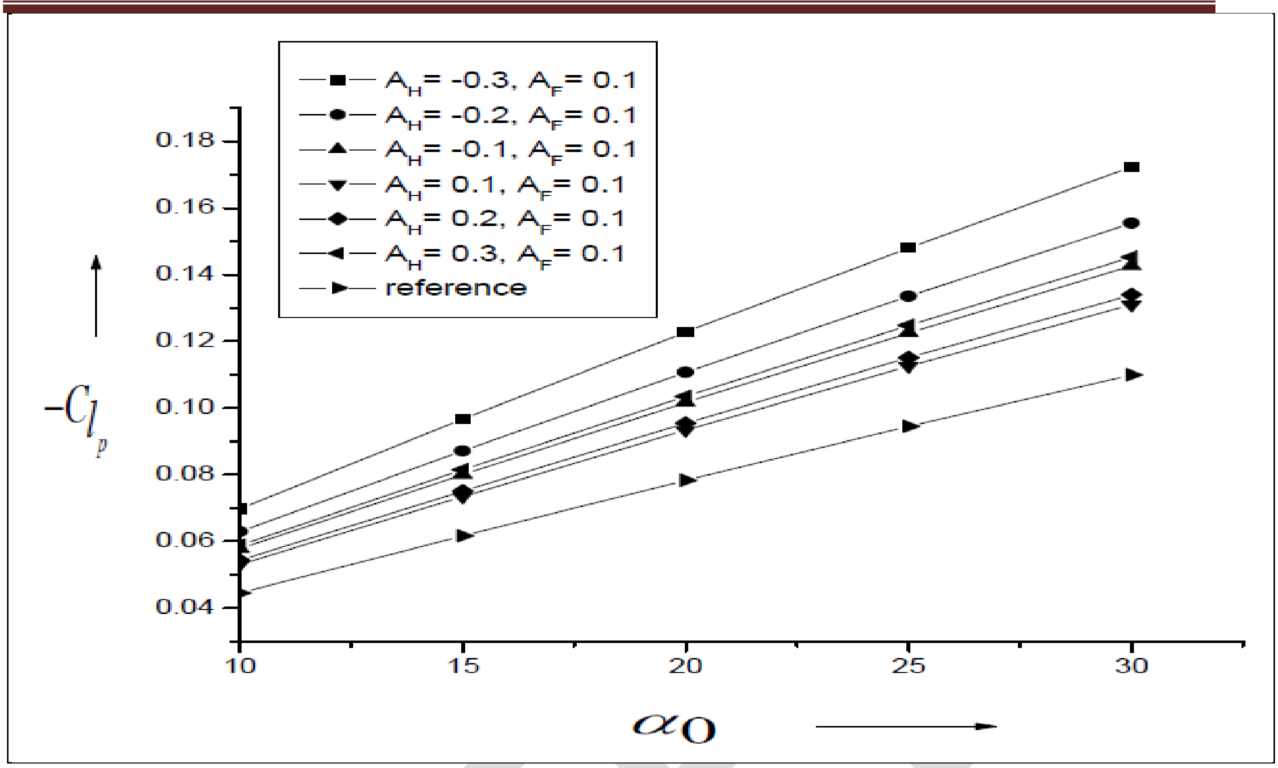

Fig. 15:variation of roll damping derivative with angle of attack for a half sine wave (M=20)

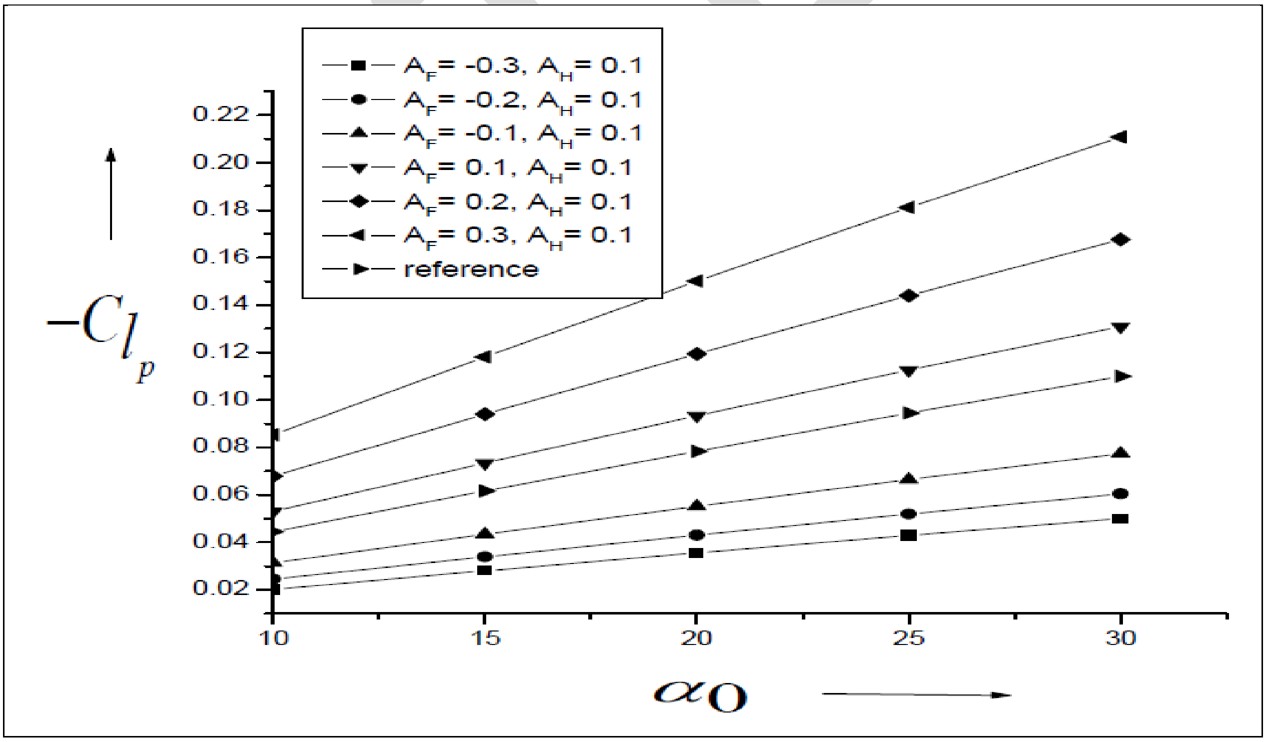

Fig. 16:variation of roll damping derivative with angle of attack for a full sine wave (M=20)

Conclusions:

In the present theory, the hypersonic similitude and the piston theory have been extended to the wing with curved leading edges. The linear dependence of the roll of damping derivativewith angle of attack is seen for all parameters of the present study. For Higher Mach Number the present theory exhibits the Mach Number Independence Principle, however, these variations the roll damping derivatives are because of change in the wing plan form area due to the variation in the amplitude of full and half sine waves. The present theory is valid for large angle of incidence and Mach number. The present theory is simpler than both Lui and Hui and Hui et al and brings out explicit dependence of the stability derivatives on the similarity parameter. The present theory is not valid for a detached shock case. Future research can be done by taking into account the effects of shock motion, viscosity, wave reflections, leading edge bluntness, and the real gas effects.

References:

[1] Pike. J, The pressure on flat and anhydral delta wings with attached shock waves, The Aeronautical Quarterly, November 1972, XXIII, Part 4, pp. 253-262.

[2] Hui, W.H., Stability of Oscillating Wedges and Caret Wings in Hypersonic and Supersonic Flows, AIAA Journal, Vol. 7, Aug. 1969, pp.1524-1530.

[3] Carrier, G.F. 1. 1949, The oscillating Wedge in Supersonic stream, Journal of Aeronautical Sciences, Vol. 16, No. 3, pp. 150-152, March.

[4] Hui, W. H., Supersonic/hypersonic flow past on oscillating flat plate at high angles of attack, ZAMP, Vol. 29, 1978, pp. 414-427.

[5] Hui, W. H., Supersonic and hypersonic flow with attached shock waves over delta wings, Proc of Royal Society, London, 1971, A. 325, pp. 251-268.

[6] Orlik-Ruckemann, K. J., Dynamic stability testing of aircraft needs versus capabilities, Progress in the Aerospace Sciences, Academic press, N.Y., 1975, 16, pp. 431-447.

[7] Hui, W. H. and Hemdan, H. T., Unsteady hypersonic flow over delta wings with detached shock waves, AIAA Journal, April 1976, 14, pp. 505-511.

[8] Lui, D. D. and Hui W. H., Oscillating delta wings with attached shock waves, AIAA Journal, June 1977,15, 6, pp. 804-812.

[9] Light Hill, M. J., Oscillating Aerofoil at High Mach Numbers, Journal of Aeronautical Sciences, Vol. 20, June 1953, pp. 402-406.

[10] Ghosh K, A new similitude for aerofoil in hypersonic flow, Proc of the 6th Canadian congress of applied mechanics, Vancouver, 29th May 3rd June, 1977, pp. 685-686.

[11] Miles, J. W., Unsteady flow at hypersonic speeds, Hypersonic flow, Butterworths Scientific Publications, London, 1960, pp. 185-197.

[12] Ghosh, K and Mistry B. K., Large incidence hypersonic similitude and oscillating non-planar wedges, AIAA Journal, August 1980, 18, 8, pp. 1004-1006.

[13] Ghosh K., Hypersonic large deflection similitude for oscillating delta wings, The Aeronautical journal, Oct.1984, pp. 357-361.

[14] Asha Crasta and Khan S. A., High Incidence Supersonic similitude for Planar wedge, International Journal of Engineering research and Applications, Vol. 2, Issue5, Sep-Oct 2012, pp. 468-471.

[15] Khan S. A. and Asha Crasta, Oscillating Supersonic delta wings with curved leading edges, Advanced Studies in Contemporary mathematics, Vol. 20(2010) , No.3, pp.359-372.

[16]Asha Crasta and S. A. Khan, Estimation of Stability derivatives of an Oscillating Hypersonic delta wings with curved leading edges, International Journal of Mechanical Engineering and Technology,Vol.3, Issue3,Dec 2012pp.483-492.,

[17]Asha Crasta and S.A. Khan, Effect of Angle of Incidence on Stability derivatives of a wing, in the proceedings of conference Challenges in research and Technology, in the coming decades, Sdmit Ujire, 27-28th September 2013, p. 47.