Analysis of Damping Derivatives for Delta Wings in Hypersonic Flow for Curved Leading Edges with Full Sine Wave

Renita Sharon Monis, Asha Crasta, Mohammed Faheem, S. A. Khan

ABSTRACT--- In this study, an attempt is made to evaluate the effect of first arched ends on the damping derived due to the pitch rate aimed at the variable sine wave bounty, flow deflection angle δ, pivot position, and the Mach numbers. Results show that with the escalation in the bounty of the complete sine wave (i.e., positive amplitude) there is an enlightened escalation in the pitch damping derivatives from h = 0, later in the downstream in the route of the sprawling verge it decreases till the location of the center of pressure and vice versa. At the location where the reasonable force acts, when we consider the stability derivatives in damping for the rate of pitch q, there is a rise in the numerical tenets of the spinoffs. This increase is non-linear in nature and not like for position near the leading edges. The level of the stifling derivatives owing to variations in Mach numbers, flow bend approach δ, and generosity of the sine wave remained in the same range.

KEYWORDS: Damping derivative, Delta wing, Hypersonic.

I. INTRODUCTION

This paper deals with the parametric calculations and analysis of high-speed flow for a wing with a leading arched edge. The wing having curved leading edges have got innumerable advantages over the wing having a straightforward foremost edge. In the case of a delta wing with a straight leading edge has a linear distribution of the wing surface area. However, the wings having arched leading edge, as in the present case, where we are replacing a straight leading verge by a full sine wave. When the heft of the sine wave is positive, this will lead to the shifting of a considerable area towards trailing edge, and this shift will depend on the bounty of the sine wave. The shift in the space towards the trailing edge will result in a considerable shift in the position of the standard force location, resulting in more massive moment arm or the higher restoring moment. Hence, this arrangement of the arched front verge of the wing enhances damping derivative magnitude in pitch, which marks remarkable improvements in the dynamic stability derivative. This increase in the damping derivative has its importance at the design stage of high-performance fighter planes at high supersonic Mach numbers. Hence, these days, all the fighter planes are using delta wing or cropped delta wing due to its superiority during the dynamics conditions. Hence in this study for the aircraft at supersonic speeds, wherein numerical computations, by geometrical variations, are explored and compared with the results of the delta wings having the straight leading edge. While a vehicle involvements a modification in both due to pitch rate and direction of attack concurrently, the twinkling derivative due to the rate of pitch proportion and incidence rate have to be estimated discretely to evaluate the whole permanency.

Ghosh's (X) unified supersonic/hypersonic similitude is being used to cultivate an unstable piston philosophy for a forecast of moment derivatives of an oscillating delta wing with a shock wave being attached at the front edge. Hui et al. (IX) have deliberate the problematic of permanency of a fluctuating plane plate annex of indiscriminate planform positioned at a specific callous incidence of attaching at high-speed flow by relating band principle. Ghosh (III) has established a two-dimensional huge refraction hypersonic similitude, which includes the piston model not limited to slim forms, the case of Lighthill's (I), and Miles (II) piston philosophies. Fluctuating delta wings at substantial prevalence were studied by Ghosh (V).

The 2-D piston philosophy of Ghosh was used with the insertion of wave echo consequence for blunt wedges/plane ogives for high-speed flow (VII). His research was reached out to swaying blunt wedges by Crasta and Khan to figure the efficient subsidizing tiny subsidiaries, for both low and high inertia levels (Ghosh, K., VI) and extreme inertia levels streams (Asha Crasta and S. A. Khan, XI-XX). The consequence of stiffness derived due to the pitch rate for a delta wing with the straight and arched foremost verge for extraordinary inertia level inFlow has been studied by Renita et al. (XXI-XXIII) for unsteady flow. In this paper, the effect of leading arched edges on the damping derived due to the pitch rate for the various bounty of the sine wave, flow deflection angle δ, pivot position, and the inertia level.

Pavitra et al. (XXVI) studied the influence of Mach M, δ (2-D wedge), and the angle of incidence on the stability derives at low supersonic Mach numbers as well as at extraordinary supersonic and hypersonic Mach numbers.

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They showed that the stability derivatives are strongly dependent on the strength of the slanted shock wave at the nose of the wedge.

Khan et al. (XXVII) computed the flow over a wedge at various angles of attack for a specific value, half-sine wave amplitude. The concave surface of the wing performed better than the convex surface.

Monis et al. (XXVIII) evaluated the stability derivatives in pitch at low as well as extraordinary Mach M. At great Mach M, the pressure intensity at the lee surface is minimal and can be neglected. However, at low supersonic Mach numbers, the pressure on the compression as well as the expansion side are comparable and hence can not be neglected. Hence they accounted for the effect of lee surface at low Mach numbers. Hence, their computed aerodynamic derivatives were very closed with actual measured results. They also accounted for the unsteady effect.

Hamizi et al., (XXIX) reported that the influence of a factor that is the quotient of the indolence dynamism to the viscid power-on performance for a delta wing. They showed how one could obtain a stable flight even at low Reynolds numbers. Bashir et al. (XXX) used CFD to simulate the flow field for a 2-D planar wedge. Their computed results were in good agreement with the analytical results as supersonic, as well as the hypersonic Mach numbers. Khan, S. A., and Mohammed Asadullah (XXXI) evaluated the pitching moment derivatives for a wedge at high Mach numbers. They found that after the particular value of the Mach number, the stiffness and damping spinoffs become sovereign of the inertia level. Bashir et al. (XXXII) computed a 6-DOF trajectory for a projectile with the wraparound fin at supersonic Mach numbers. They deduced the aerodynamic derivatives from the flight data. Aerodynamic coefficients obtained from the fight data matched were well with the analytical as well as with the wind tunnel results. Shanbhag et al. (XXXIII) computed the surface pressure distribution over a delta wing. They compared their results with that of Lighthill's theory, second-order shock expansion theory, and by the Ghosh theory. Asha Crasra and S. A. Khan (XXXIV) computed the damping derives in pitching and rolling motion for a wing at low and high inertia levels.

Asrar et al. (XXXV-XXXVI) simulated the flow field for a wraparound fin projectile motion. Their study relates how the fin canting angle can affect the performance of the projectiles. They suggested putting the projectile in chaotic motion in a significant part of the flight so that the aerodynamic vehicle is not traceable by the radar.

Asha Crasra and S. A. Khan (XXXVII) computed the aerodynamic variables for wing with variable-sweep angles. They have evaluated the effectiveness of the flow deflection angle.

Based on the above review, the present study presents the results of the damping results at low supersonic Mach numbers and also at hypersonic Mach numbers. This paper presents the consequence of the foremost arched edge over the conventional primary edge over an extensive range of angles of attack, pivot position, semi-vertex angle. The leading edge shape also discussed in detail.

II. ANALYSIS

The pitching moment per unit span about the pivot $x = x_0$ due to the only lower surface is

$$\overline{M} = \int_0^L (x - kL)\overline{P}_p \, dx$$

$$-\frac{\partial \overline{M}}{\partial \dot{\theta}} = \int_0^L (x - kL)\frac{\partial \overline{P}_p}{\partial \dot{\theta}} \, dx$$

$$-\frac{\partial \overline{M}}{\partial \dot{\theta}} = \int_0^L (x - kL)\rho_2 a_2 u_\infty \left(\frac{c_2 x}{u_\infty} + \frac{c_3 L}{u_\infty}\right) dx$$

For the windward side, the equation is

$$-\frac{\partial M}{\partial \dot{\theta}} = \rho_2 a_2 L^3 \left[c_2 \left(\frac{1}{3} - \frac{k}{2}\right) + c_3 \left(\frac{1}{2} - k\right) \right]$$

On the enlargement side of the plane plate, the flow turns through a Prandtl-Meyer expansion at the leading edge to become parallel to the upper surface.

First, Mach number M_θ downstream of the enlargement is determined first.

At zero angle of attack, flat plate wavering in a torrent of inertia level of M_θ, it is anticipated that pressure fluctuation is identical.

Due to this methodology, the fluctuation of the enlargement flow owing to flat plate swaying is not accounted for.

It is worth to mention that the wing lee exterior is at much lesser stress than the upwind area. The impact on damping at substantial Mach numbers is insignificant as paralleled to the upwind surface.
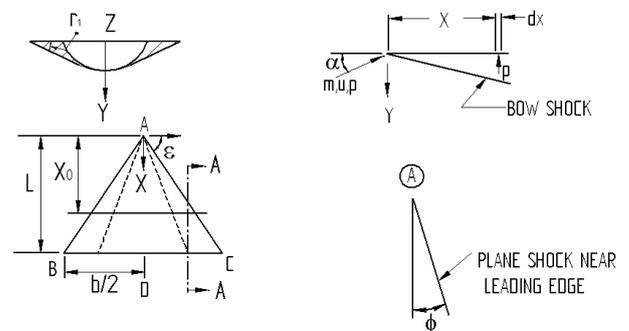

Fig. 1 Different views of the Delta Wing .

$$V_p = \frac{u_\theta}{\cos \mu_\theta} + \frac{(x - kL)\dot{\theta}}{\cos \mu_\theta}$$

Piston Mach number

$$M_p = \frac{v_p}{a_\theta} = \frac{M_\theta \theta}{\cos \mu_\theta} + \frac{(x - kL)\dot{\theta}}{a_\theta \cos \mu_\theta} \quad \ldots\ldots (1)$$

Where M_θ is the inertia levels behind the expansion area, a_θ is the speed at Mach unity & μ_θ is the Mach cone inclination behind the Meyer relief.

The ideally expanded relations for stress quotient is given by

$$\frac{p_p}{P_\theta} = (1 - \frac{\gamma - 1}{2} M_p)^{\frac{2\gamma}{\gamma - 1}}$$

As θ and $\dot{\theta}$ tend to zero, the equation for the acoustic expression

$$\frac{P_p}{P_\theta} \simeq (1 - \gamma M_P)$$

Pitching Moment,

$$\bar{M} = \int_0^L (x - kL)P_p\, dx = \int_0^L (x - kL)P_\theta(1 - \gamma M_P)\, dx$$

$$-\frac{\partial \bar{M}}{\partial \dot{\theta}} = \int_0^L (x - kL)P_\theta(-\gamma)\frac{\partial M_p}{\partial \dot{\theta}}\, dx$$

From equation (1)

$$-\frac{\partial \bar{M}}{\partial \dot{\theta}} = -\int_0^L (x - kL)P_\theta(-\gamma)\frac{(x - kL)}{a_\theta \cos \mu_\theta}\, dx$$

$$-C_{m_{\dot{\theta}}} = \frac{4}{\rho_\infty u_\infty bL^3}\left[-\frac{\partial \bar{M}}{\partial \dot{\theta}}\right]$$

$$C_{m_{\dot{\theta}}} = \frac{4\rho_2 a_2}{\rho_\infty u_\infty}\frac{1}{(\cot \varepsilon - \frac{4A_H}{\pi})}\left[c_2[(\frac{1}{8} - \frac{k}{6})\cot \varepsilon - \frac{A_F}{4\pi} - \frac{A_H}{2\pi} - \frac{2A_H}{\pi^3} + \frac{k(A_H + \frac{A_F}{2})}{\pi}] + C_4[(\frac{k}{2} - \frac{k^2}{2} - \frac{1}{8})\cot \varepsilon - \frac{3A_F}{4\pi} + \frac{A_H}{2\pi} - \frac{6A_H}{\pi^3} + \frac{k}{\pi}(\frac{3A_F}{2} - A_H) + 2k^2\frac{A_H}{\pi}]\right.$$

$$\left. + \frac{p_\theta}{p_\infty}\frac{a_\infty}{a_\theta}\frac{M_\theta}{M_\infty}\frac{1}{\sqrt{M^2_\theta - 1}}\left[(1 - \frac{8k}{3} + 2k^2)\cot \varepsilon + \frac{2A_F}{\pi} - \frac{4A_H}{\pi} + \frac{16A_H}{\pi^3} + \frac{4k}{\pi}(2A_H - A_F) - \frac{8A_H k^2}{\pi}\right]\right] \quad \ldots\ldots\ldots\ldots\ldots\ldots\ldots\ldots(2)$$

When

$A_H = 0$

$$-C_{m_{\dot{\theta}}} = \frac{4\rho_2 a_2}{\rho_\infty u_\infty}\frac{1}{(\cot \varepsilon)}\left[c_2[(\frac{1}{8} - \frac{k}{6})\cot \varepsilon - \frac{A_F}{4\pi} + \frac{k(\frac{A_F}{2})}{\pi}] + C_4[(\frac{k}{2} - \frac{k^2}{2} - \frac{1}{8})\cot \varepsilon - \frac{3A_F}{4\pi} + \frac{k}{\pi}(\frac{3A_F}{2})]\right.$$

$$\left. + \frac{p_\theta}{p_\infty}\frac{a_\infty}{a_\theta}\frac{M_\theta}{M_\infty}\frac{1}{\sqrt{M^2_\theta - 1}}\left[(1 - \frac{8k}{3} + 2k^2)\cot \varepsilon + \frac{2A_F}{\pi} - \frac{4k}{\pi}(A_F)\right]\right]$$

Using the above analytical expression for the damping derivatives results were computed for a comprehensive range of inertia levels in the various low to high levels of inertia (i.e., M = 4 to 20) for flow deflection angles from δ = 5 to 20 degrees. Results so obtained are deliberated in the next section.
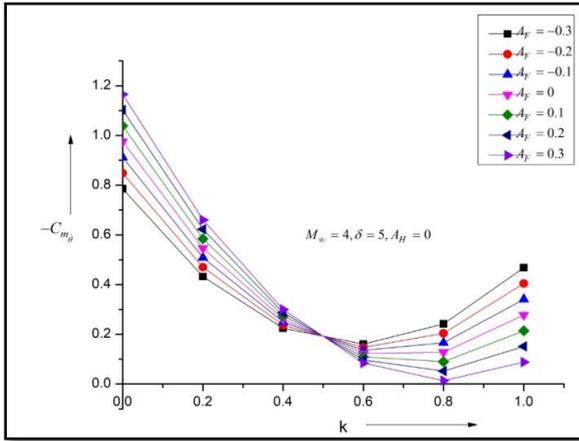
III. RESULTS & DISCUSSIONS

Fig. 2. Damping derivative Vs. k, $M_\infty = 4$, $\delta = 5$

Before analyzing the results, let us understand the reasons why we have considered the curved leading edge. It is well known that if the wing is a rectangular wing, then the position of the center of pressure will be at ¼ c_{mac} (quarter chord of the mean aerodynamic chord from the leading edge). When we replace a quadrangular wing with a tapering wing, due to the sweep angle to the rounded edge of the wing, the location of the point of application of normal force will move in the direction of the trailing edge depending upon the sweep angle. The location of the aerodynamic center will further get shifted towards the sprawling edge owing to the variation in the chord length at the root (C_r) and the tip (C_t). In this case $C_{mac} = (C_r + C_t)/2$. In the case of the wing with forty-five degrees sweep angle, the tip chord is zero. There will be a maximum shift in the position of the center of pressure for this case. Now while a sine wave is overlaid over the traditional foremost verge of the wing, the position of the normal force will additionally get progressed to the trailing verge due to the shift in the considerable area. Hence we will analyze the results in view of the above changes in the geometry of the wing.

Fig. 2 shows the variant of the damping derivatives with the pitching point (h), for high supersonic Mach number M = 4, and angle $\delta = 5$ degrees. When leading edge is straight $A_F = 0$, Fig. 2 reflects the outcomes of the stability derivatives for $A_F = 0$, ± = 0.1, 0.2, and 0.3. The results show that when $A_F = 0$ and the amplitude is increased from 0 to 0.1, the %increase in the damping derivative is in the range from 5 to 7% for the pivot positions before the center of pressure, later in the downstream towards the trailing edge it remains in the range from 10 to 30%. For the huge bounty of the full sine wave, namely at $A_F = 0.2$ and 0.3 for the forward position of the pivot, namely up to 0.4, the percentage increase in the damping derivatives remained in the same as was seen when amplitude was $A_F = 0.1$. However, in the downstream in the direction of the trailing edge on the negative side, they vary from 12 to 42 % for $A_F = 0.2$, and for $A_F = 0.3$, there is an increase in the %change, which is in the range 13 to 73%. This increase in the damping derivative in pitch is attributed in view of the shift of the wing part to the rambling edge. When we consider the negative amplitude of the full sine wave (i.e., $A_F = - 0.1, 0.2,$

and 0.3) for the forward locations of the wing up to h = 0.4, the %change is on the negative side in the range from 5 to 9%, in downstream the %change turn to be positive, but the numerical values are on the lower side in the range 8 to 23%. This change in the damping derivative can be linked through the swing in the planform area after the trailing verge to the leading edge.
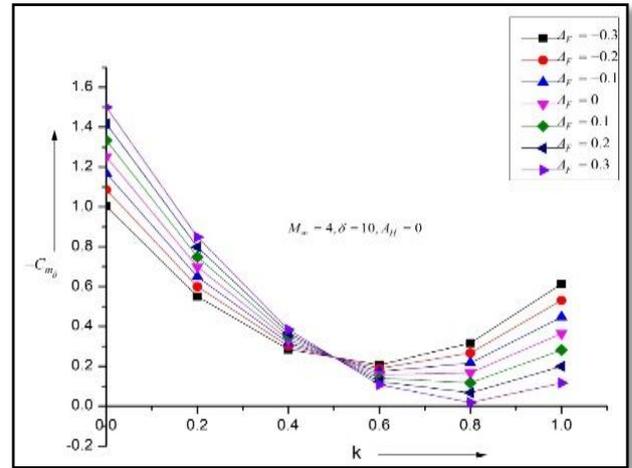

Fig. 3. Damping derivative Vs. k, $M\infty = 4$, $\delta = 10$ degrees.

Similar results are seen in Fig. 3, where $\delta = 10$, which is increased from five to ten degrees. This increase in the flow angle δ is able to stimulus alteration in the damping results even at lower Mach number being is four, which is limiting Mach number for the flow being supersonic. But, the domination of the numerical values of the bounty of the sine wave is clearly seen. It is also perceived that for a known Mach number, and δ, the effectiveness of the bounty of sine wave remained the same intended for all the combinations of the δ.
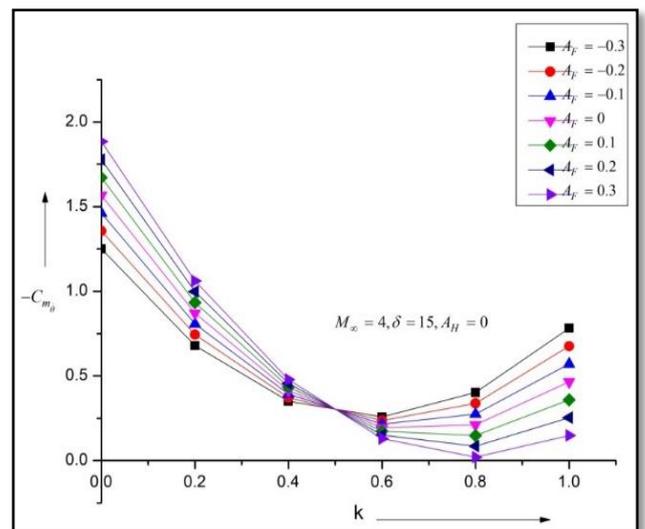

Fig. 4. Damping derivative Vs. k, $M\infty = 4$, $\delta = 15$ degrees

Fig. 4 shows the outcomes for Mach M = 4 and δ = 15 degrees for various amplitudes of the full sine wave in the range

A_F = ± 0.1 to 0.3. As discussed earlier, the incremental change in the angle δ from five to fifteen degrees yields in substantial in an increase of the damping derivatives, and for the higher values of the δ say at fifteen and twenty degrees results in enormous values for the reasons stated as above.
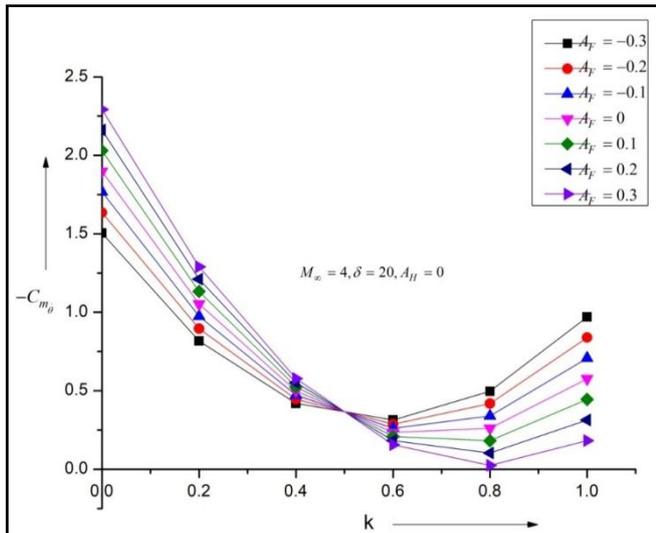

Fig. 5. Damping derivative Vs. k, M∞ = 4, δ = 20 degrees.

For the highest value of the δ = 20 degrees keeping all the parameters the same as in the case of Figs. 2 to 4, the results are shown in Fig. 5. Due to a substantial rise in the planform area, here is most significant in the value of the damping results of wing. However, the presence of the full sine wave still shows an active part in fixing the scale of the damping derivatives.

Fig. 6 depicts the damping derivatives dependence on the different locations of the pitching point for limited increased Mach number and hence the inertia ratio for the different amplitudes of the sine wave. This minimal rise in M, results in a marginal increase in numerical tenets of damping results, as this Mach number is the limiting highest value of the supersonic Mach number for higher values of δ. For lower values of δ, the flow may not remain supersonic and just sufficient for the flow to become hypersonic. The characteristics of hypersonic flows are totally different in comparison to the supersonic flow.

Figs. 7 to 9 contemporary the distribution of damping results with respect to the k position for Mach M = 5, for numerous flow angles (i.e., δ = 10, 15, 20 degrees). As we have seen while analyzing the results at Mach M = 4, the growth in the angle δ results in the similar results in substantial increase in the amount of the damping derives as was perceived for the lesser inertia and the same is right at Mach M = 5 also, as the flow characteristics almost are on the similar lines.
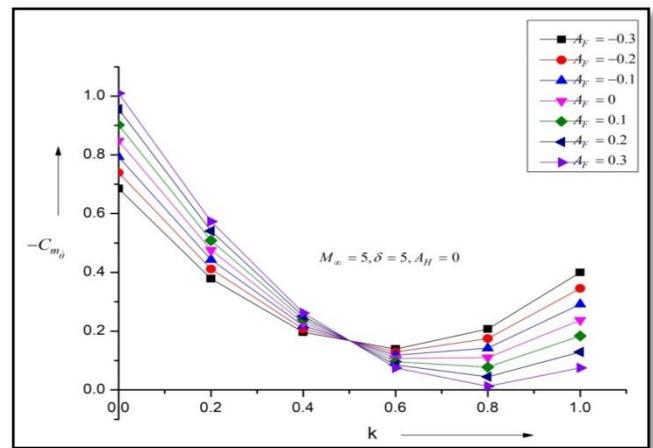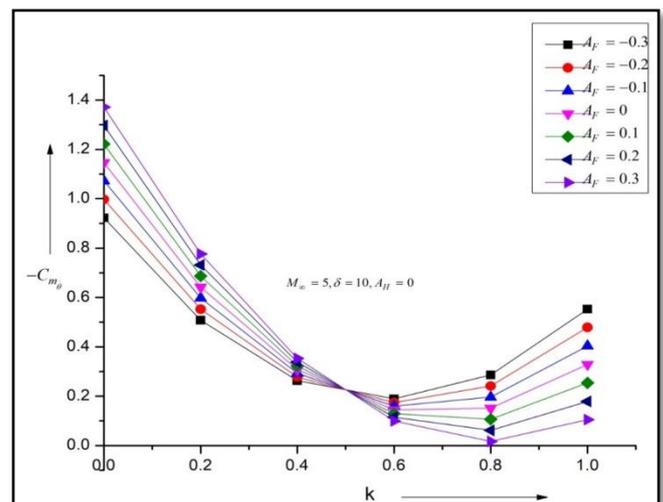

Fig. 6. Damping results Vs. k, M∞ = 5, $\delta = 5$.

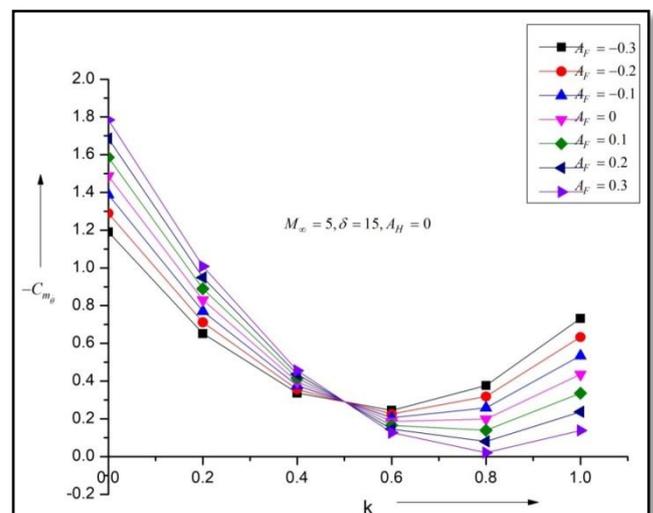
Fig. 7. Damping derivative Vs. k, M∞ = 5, $\delta = 10$.

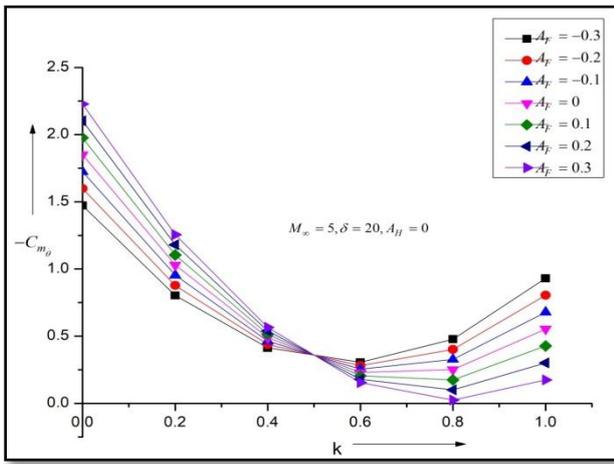
Fig. 8. Damping derivative Vs. k, M∞ = 5, $\delta = 15$.

Fig. 9. Damping derivative Vs. k, M∞ = 5, δ = 20.

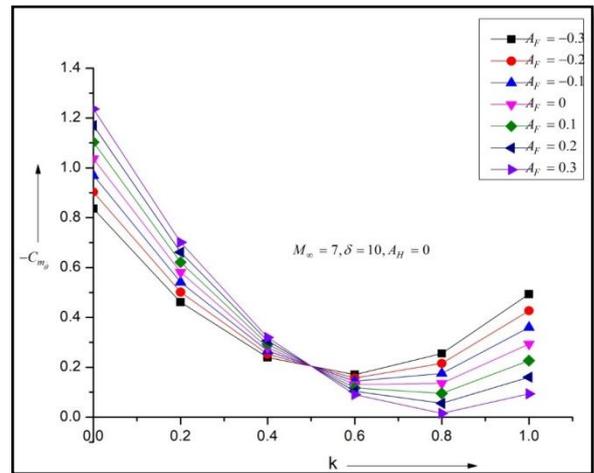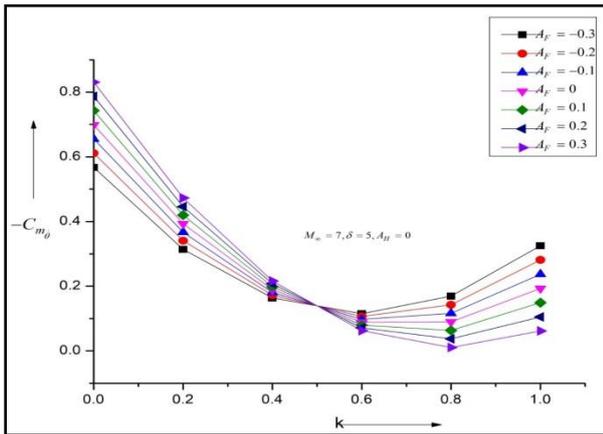**Fig. 11. Damping derivative Vs. k, M∞ = 7, δ = 10.**
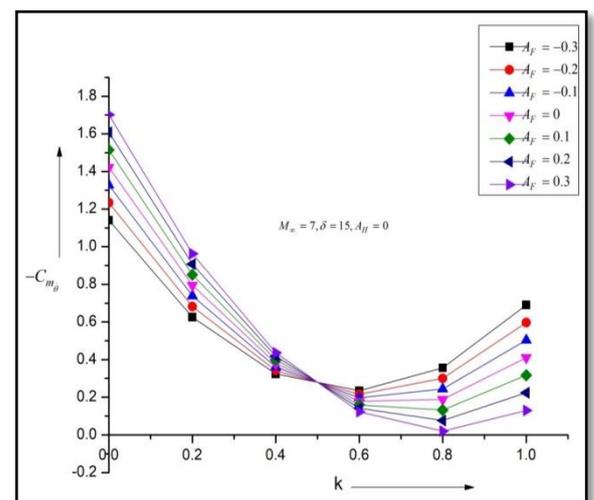

Fig. 10. Damping derivative Vs. k, M∞ = 7, δ = 5.

Fig.12. Damping derivative Vs. k, M∞ = 7, δ = 15.

Figs. 10 to 13 show the outcomes for Mach M = 7, for different δ = 5, 10, 15, and 20 degrees, which results in a rise of plan form space progressively; when the full sine wave was placed at the leading edge of the wing. It is observed that with the growth in the inertia levels from supersonic to hypersonic, a drop in the numeric of the damping derives. However, the effect of an increase in δ on the numerical values of the damping derives; is on the comparable, as noticed at the lower inertia. The results further indicated that there is a continued reduction in the numerical values of the damping derives as the principle of independence of inertia is yet to hold, which may happen for higher Mach numbers around M = 10 and above. The pattern of the flow deflection angle remained similar, which just a geometrical parameter will influence the stress scattering on the plan form of a wing. It is realized that the magnitude of the damping derive nearly double when we compare the values at δ = 5 degrees.

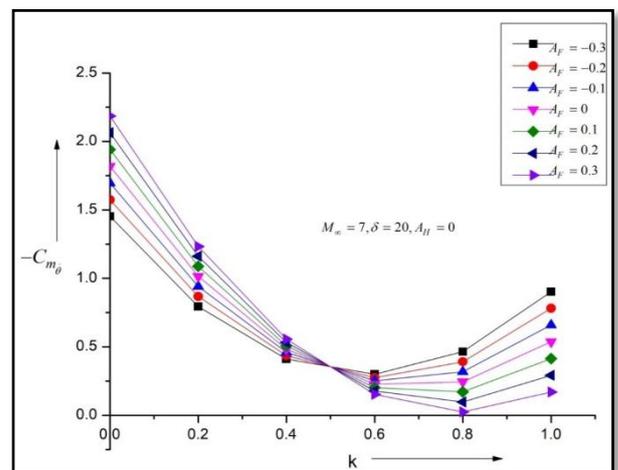**Fig.13. Damping derivative Vs. k, M∞ = 7, δ = 20.**

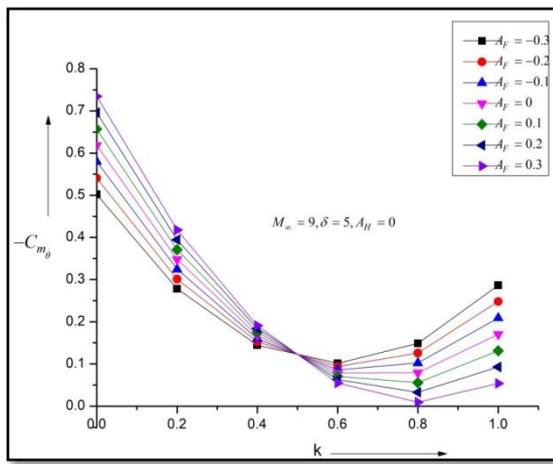**Fig.14. Damping derivative Vs. k, M∞ = 9, δ = 5.**

Figs. 14 to 17 present results for Mach number M = 9, at δ = 5 degrees, at various pivot position for different amplitudes in the range $A_F = \pm$ 0.1, 0.2, and 0.3. Results indicate that due to the increase in the inertia level from M = 4 to 9, here is an uninterrupted reduction in the values of the damping derives. The pattern of increase and decrease in the damping derivatives remained the same and also its range due to the modification in the largeness of a sine wave. Also, while the wing area was increasing, the δ = 5 to 20 degrees. The results show that there is a substantial growth in the numerical data of damping derives.
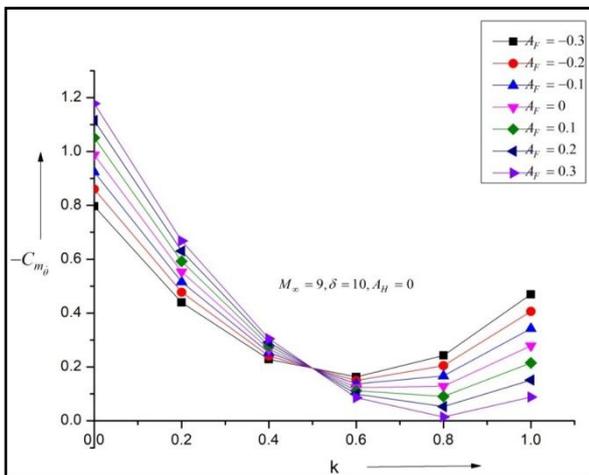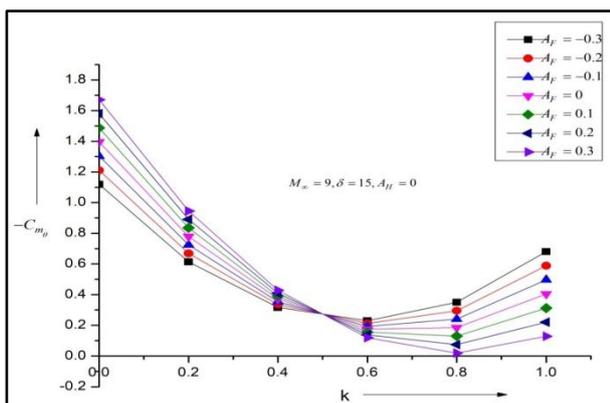

Fig. 15. Damping derivative Vs. k, M∞ = 9, δ = 10.
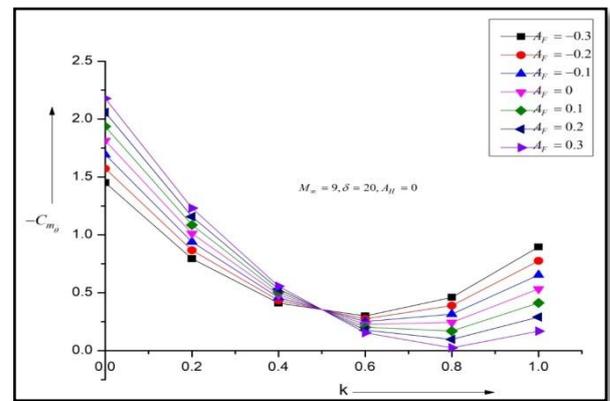

Fig. 16. Damping derivative Vs. k, M∞ = 9, δ = 15.
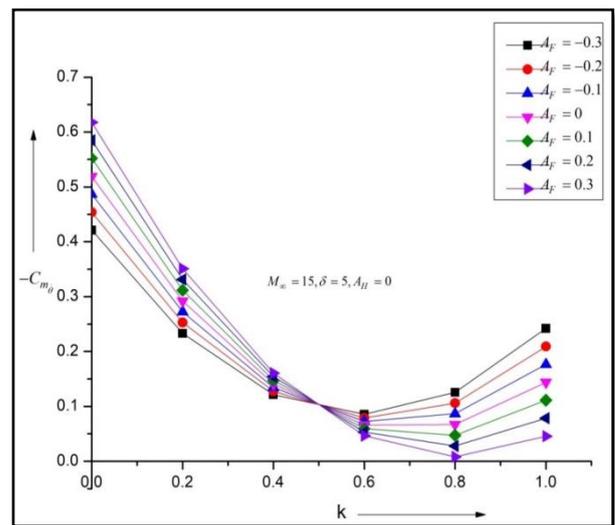

Fig. 17. Damping derivative Vs. k, M∞ = 9, δ = 20.
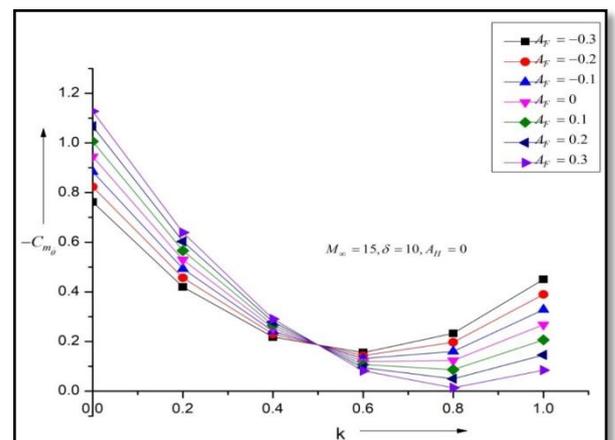

Fig. 18. Damping derivative Vs. k, M∞ = 15, δ = 5.

Fig. 19. Damping derivative Vs. k, M∞ = 15, δ = 10.

Figs. 18 to 21 for very high values of Mach (i.e., M = 15) for the same range of flow incidence angle δ, and also the bounty of the full sine wave. It is found that there is a considerable reduction in the values of the damping derives, however, when either the full sine wave is superposed at the first straight verge of a wing, or the flow incidence angle δ

is varied starting five degrees to twenty degrees effects in very high data of the damping results. But this increase in the damping derivatives is in such a way that when we evaluate the relative percentage increase, it remains in the same order of magnitude for rest of the remaining values of the Mach numbers, the incidence angle for the flow, and the increased area of the full sine wave in the range from ± 0.1 to 0.3.
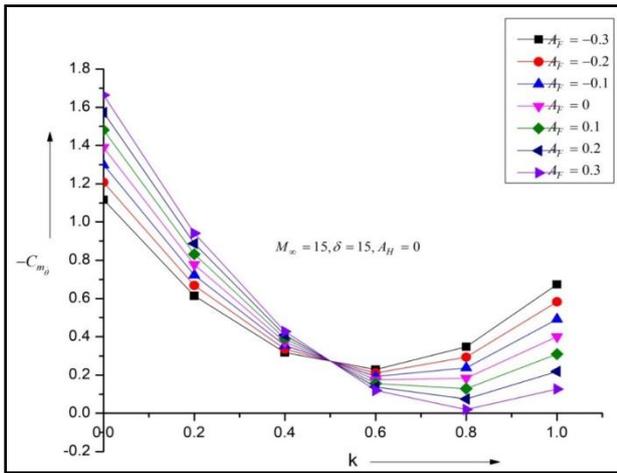

Fig. 20 Damping derivative Vs. k, M∞ = 15, δ = 15.

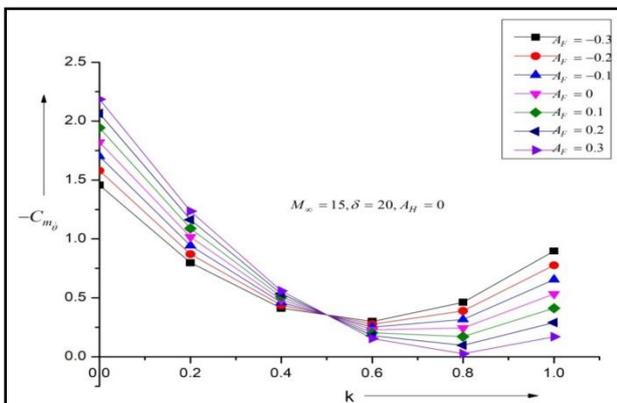**Fig. 21. Damping derivative Vs. k, M∞ = 15, δ = 20.**

Fig. 22. Damping derivative Vs. k, M∞ = 20, δ = 5.
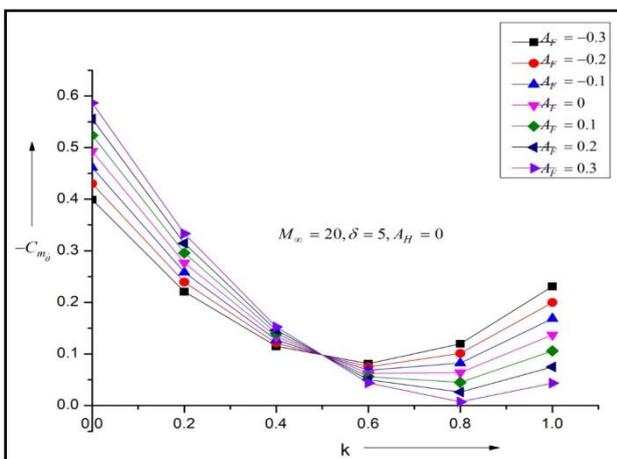
Figs. 22 to 25 display the results of the damping results due to the rate of pitch for the maximum Mach of the present learning (i.e., M = 20), for various wing planform area by varying the flow angles of incidence from 5 to 20 degrees and the bounty of the full sine wave as considered earlier.

As seen, when the flow incidence angle is 5 units, the values are minimal, but with an increase in the angle δ, after 5 to 10 degrees, here is a considerable growth in the damping derive, and this tendency stays till δ = 20 degrees. The values of the damping derivatives become almost double for every five degrees increase in the flow deflection angle. This trend is seen at higher Mach numbers; however, at lower inertia levels from M = 4 to 9, this increase was limited.

Fig.23. Damping derivative Vs. k, M∞ = 20, δ = 10.
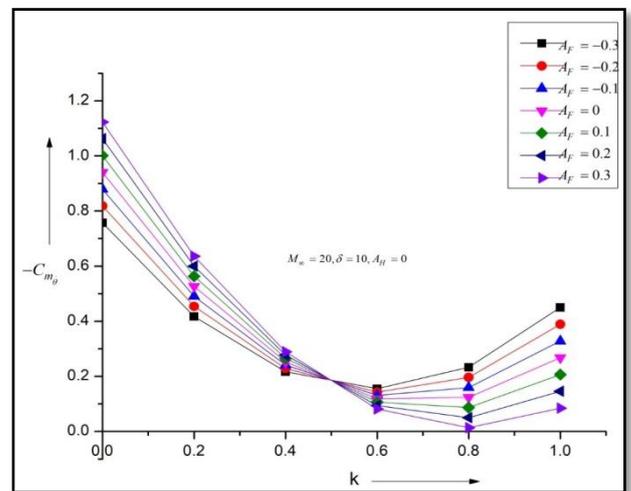

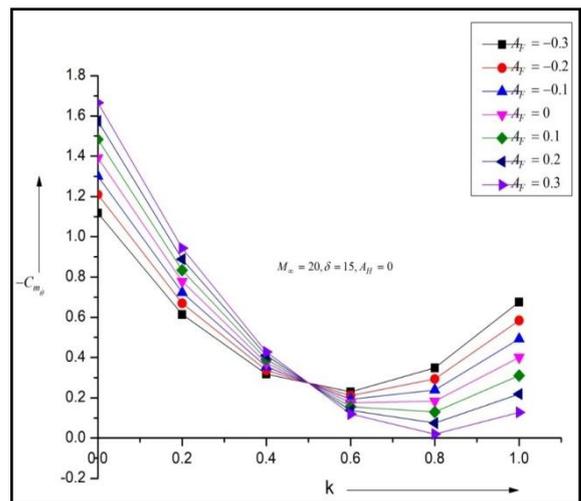**Fig. 24. Damping derivative Vs. k, M∞ = 20, δ = 10.**

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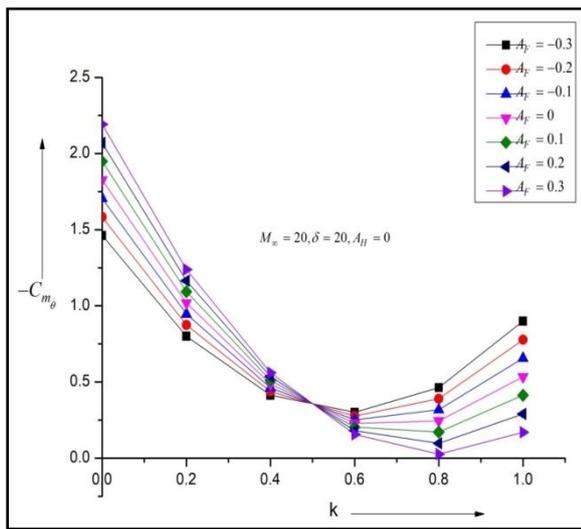*Blue Eyes Intelligence Engineering*
& Sciences Publication

Fig.25. Damping derivative Vs. k, M∞ = 20, δ = 20.

IV. CONCLUSIONS

Established in the directly above deliberations, we may conclude as the following:

- Results for a wing with conventional foremost edges and leading arched edges are computed and compared for a comprehensive assortment of the inertia levels, flow angle, and the bounty of a sine wave.

- Results indicate that by a rise in the Mach M, there is a continuous reduction in the values of the damping derives, and later for higher Mach M, it becomes independent of the Mach numbers.

- In view of the change in shape, due to the superposition of sine wave on the first straight verge of a wing, $A_F = \pm 0.1$, 0.2, and 0.3. It is seen that when the bounty of sine surge is positive, it takes an amount of wing area from the leading edge shifts the same amount of the wing area towards the trailing edge. This shift in the wing area results in considerable growth in the damping derives.

- It is found that the growth or drop in the damping derives remained constant for all the inertia levels and the flow deflection angles.

- It is also found that at higher Mach numbers, when the flow angles incidences are increased from 5 to 20 degrees at higher Mach numbers, the increase in the damping derivatives is massive, whereas, at the lower Mach numbers, this increase is marginal.

- From the results, it is observed that with the variation in the amplitudes of sine surge, the location of the center of pressure remained similar.

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