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Engineering Notes

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Computation of Limit-Cycle Oscillations of a Delta Wing

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Introduction

THE new unmanned combat air vehicle configurations being developed will need to be highly maneuverable while allowing for increased flexibility in the wing structure. As an archetypical nonlinear aeroelastic problem for this type of aircraft, the simulation of a limit-cycle oscillation (LCO) of a flat plate delta wing is considered. Tang et al.[1] have simulated LCOs of moderate sweep delta wings in low subsonic flows, which arise from geometric nonlinearities in the structure. Their aeroelastic model consisted of vortex-lattice aerodynamics coupled with a von Kármán plate model for the delta wing. In a subsequent work, Tang and Dowell[2] demonstrated the influence of small changes in angle of attack (<2 deg) on the limit-cycle response.

Gordnier and Melville[3] investigated the LCO of a cropped delta wing using an aeroelastic solver, which couples a Navier–Stokes code with a linear modal structural scheme. The computational fluid dynamics (CFD) scheme employed is based on the three-dimensional, Beam–Warming algorithm.[4] This flow solver has been applied to the simulation of a wide range of unsteady flow phenomena.[5–8] Implicit coupling between the CFD and structural solvers is accomplished using subiterations, avoiding the need to develop a completely new tightly coupled solver. Amplitudes of the computed LCOs using this aeroelastic code were significantly higher than the corresponding experimentally measured values.[9]

In Ref. 3, the proposed reason for the discrepancy in the amplitude of the response is the lack of nonlinear structural effects. The present Note investigates the effect of modeling geometric structural nonlinearities for this delta wing problem. The linear, modal structural solver in the earlier mentioned aeroelastic code is replaced with a nonlinear finite element solver for the von Kármán plate equations. This new aeroelastic solver, which has been described and applied to panel flutter problems in Ref. 10, will be employed for the delta wing LCO problem. The computed frequencies and amplitudes of the LCO response will be compared with the previous linear results and the experimental measurements.

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Delta Wing LCO Results

The delta wing investigated in this study (Fig. 1) is based on the experimental configuration of Schairer and Hand.[9] This model consists of a semispan cropped delta wing cut from 0.035 in thick cold-rolled steel plate. The delta wing has a 47.8-deg leading-edge sweep and a -8.7-deg trailing-edge sweep. The leading edge, trailing edge, and wingtip are all blunt. The mechanical properties for cold-rolled steel are specified as mass density $\rho_s = 0.283$ lbm/in.^{3}, Young's modulus $E_s = 30 \times 10^6$ lbf/in.^{2}, and Poisson's ratio $\nu = 0.25$. For the initial computations, the model is assumed to be rigidly clamped (no in-plane motion allowed) along the root chord of the wing. Computational results for the delta wing are compared with the experimental measurements of Schairer and Hand.[9] In this experiment, the amplitude and frequency of the deflection of the wingtip trailing edge were measured using stereo photogrammetry. LCOs of the delta wing were observed for an initially nonlifting wing ($\alpha = 0$ deg) in transonic flow at a series of freestream dynamic pressures. The flow conditions and structural parameters explored in the experiments of Schairer and Hand and the present computations are given in Table 1.

In the previous computations reported in Ref. 3, LCOs of the cropped delta wing are computed using an aeroelastic solver that couples a Navier–Stokes code with a linear, modal structural model. In these computations, the growth of the oscillatory response of the

Table 1 Flow conditions and structural parameters

Dynamic q, pressure, psi	λ	μ_s	Mach number	Reynolds number, $\times 10^6$
2.29	65.0	0.0188	0.878	
2.58	73.21	0.0216	0.879	2.700765
2.78	78.89	0.0235	0.878	2.931988
2.98	84.56	0.0253	0.874	3.154261
3.16	89.67	0.0270	0.872	3.361148
3.33	94.50	0.0286	0.869	3.558253
3.45	97.90	0.0304	0.860	3.722870
3.88	110.0	0.0338	0.860	
4.41	125.0	0.0388	0.860	
4.93	140.0	0.0438	0.860	2.931988
5.46	155.0	0.0488	0.860	

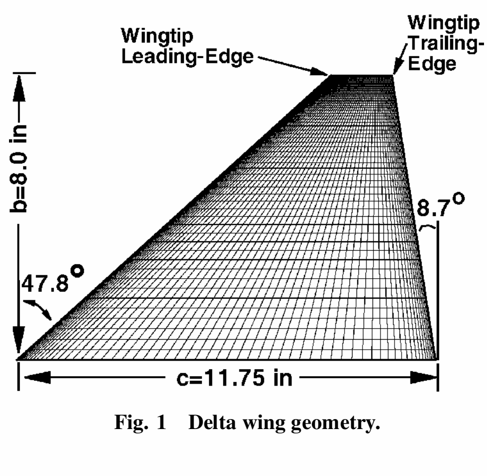
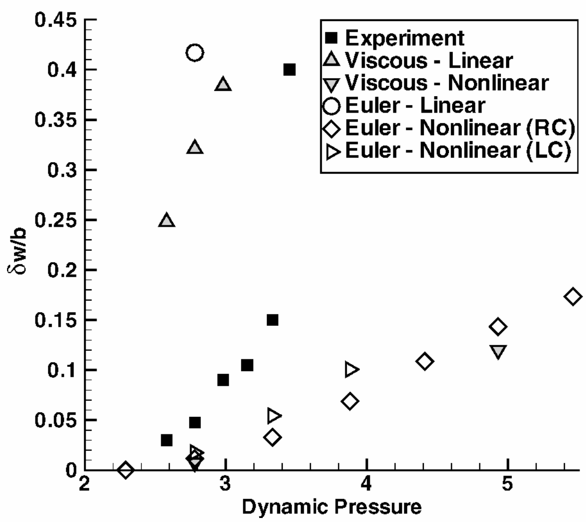


Fig. 1 Delta wing geometry.

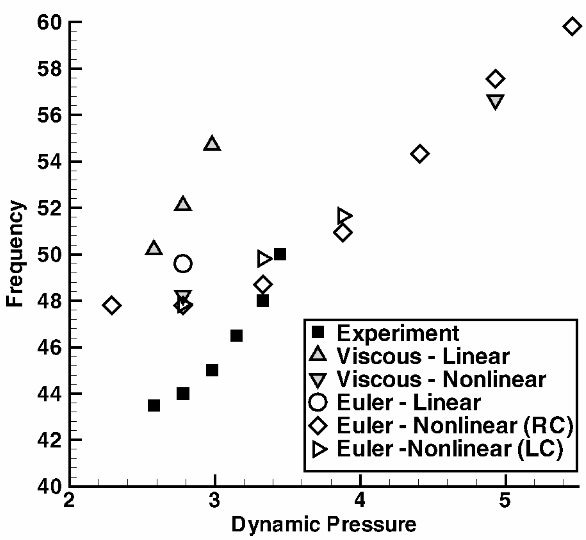
delta wing results from a lag between the first torsional mode and the first bending mode that produces a net energy input into the system. The nonlinear aerodynamic mechanism that limits the growth of the response and yields the limit-cycle motion is the development of a leading-edge vortex. This vortex acts like an aerodynamic spring, producing a normal force that is approximately 180-deg out of phase with the motion of the wing. This computational model, therefore, simulates a delta-wing LCO that results from nonlinear aerodynamic sources.

Comparison of the amplitudes and frequencies of the wingtip trailing-edge deflections for this computational model with the experimental measurements (Fig. 2) reveals substantial discrepancies. This is particularly true for the amplitude of the response, with the frequencies showing more reasonable agreement. The proposed reason for these differences given in Ref. 3 is the absence of nonlinear terms in the structural model. These terms should play an important role given the relatively large deflections of the delta wing.

The influence of geometric structural nonlinearities on the delta wing LCO response is now explored using the nonlinear aeroelastic solver developed in Ref. 10. As in Ref. 3, each simulation is initiated from a steady flow solution obtained at angle of attack $\alpha = 0$ deg. The delta wing is excited by providing an initial velocity to the first bending mode. Computations using the



a) Amplitude of LCO at the wingtip trailing edge



b) Frequency of LCO at the wingtip trailing edge

Fig. 2 Comparison of computed and experimental[9] amplitudes and frequencies, rigidly clamped (RC) and loosely clamped (LC).

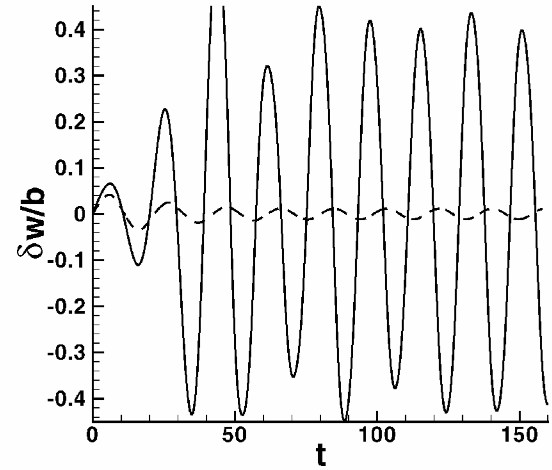


Fig. 3 Comparison of wingtip trailing-edge LCO using linear and nonlinear structural models for a freestream dynamic pressure, q = 2.78: ——, linear plate model and – – –, nonlinear plate model.

inviscid Euler equations for the aerodynamic model are considered first. Figure 3 shows a comparison of the dynamic response of the wingtip trailing edge for the linear and nonlinear structural models. The dramatic effect of the nonlinear structural terms on the computed LCO of the delta wing is clearly seen. The nonlinear case shows a large reduction in the amplitude of the LCO and a lower frequency.

Figure 2 shows the computed amplitudes and frequencies for the nonlinear structural case with a rigidly clamped edge for increasing freestream dynamic pressures. The first experimental point plotted is for a dynamic pressure at which sustained, low-amplitude vibrations were just beginning to appear.[9] This point is between the last point for which no computed LCO is obtained and the first point with a computed LCO. This result indicates that the experimental onset of the instability occurs in the same range of dynamic pressure as the experiment. Furthermore, the amplitudes of the computed response are now similar to the experimental measurements. The rate of growth of the LCO with increasing dynamic pressure is slower, however, with the computations requiring larger dynamic pressures to achieve the same amplitude of response. The frequencies for the computed LCO are slightly higher than the experiment at the onset of the instability but are closer to the experiment than the linear structural model values.

Several of the nonlinear structural cases are recomputed using the Navier–Stokes aerodynamic model to explore viscous effects. For the smaller amplitude deflections obtained with the nonlinear plate theory, the influence of viscosity on the amplitude and frequency of the LCOs is small (Fig. 2). This is in contrast to the case with the linear structural model, where viscosity plays a more important role. In this instance viscous effects resulted in much greater reductions in both amplitude and frequency.

Finally, the impact of the boundary condition specified along the root chord of the delta wing is investigated. Computations are recomputed using a loosely clamped boundary condition, which allows for motion in the plane of the plate. The computations with this boundary condition show a modest improvement in the rate of growth of the amplitude of the LCO response when compared with the experimental measurements. The amplitudes still fall below those observed in the experiment. The change in boundary condition has little effect on the frequency of the response.

The preceding investigation indicates the significant impact the nonlinear structural terms have on the computed limit-cycle response for these delta-wing cases. Further examination of the aerodynamics of the delta-wing LCO with the nonlinear structural model reveals that a well-established leading-edge vortex with well-defined supersonic flow regions does not appear until dynamic pressures of $q = 4.41$ and above. Therefore, this nonlinear flow feature no longer provides the mechanism for the development of the limit-cycle response as described for the linear structural model.[3] Rather, the stiffening of the delta wing due to the development of the membrane stresses in the von Kármán plate model limits the growth of the delta-wing response. Tang et al.[1] have also shown that this type of geometric nonlinearity can produce LCOs of low-aspect ratio delta wings in low subsonic flows. The slower growth in amplitude of the LCO most likely results from excessive stiffness in the von Kármán plate model for the large plate deflections (2–40 plate thicknesses) that occur.

Conclusions

A new computational aeroelastic technique that incorporates a nonlinear von Kármán plate model has been applied to the simulation of LCOs of a cropped delta wing. In a previous work, computations for the this delta wing were accomplished using an aeroelastic solver that employed a linear modal structural representation for the wing structure. In these simulations the development of a leading-edge vortex provided the nonlinear mechanism leading to the LCOs. The amplitude of the resulting limit cycle was significantly higher than the experimental measurements, however. Computations for the same case with the present aeroelastic model that incorporates structural nonlinearities result in limit cycles with amplitudes similar to the experimental measurements, albeit with a slower rate of growth as dynamic pressure is increased. No leading-edge vortex or well-defined supersonic flow develops on the wing during these LCOs. In this case, the geometric structural nonlinearities provide the physical mechanism leading to the development of the LCO. Further modifications to the nonlinear structural model may be required to capture more accurately the growth of the amplitude of the LCO.

Acknowledgments

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Structural Sizing for Buckling Critical Body Structure of Advanced Aircraft

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Introduction

STRUCTURAL weight estimation for use at the pre-preliminary design level or in a vehicle synthesis computer code is an important and challenging part of any advanced aircraft design study. Two methods are commonly used to estimate structural weight, but both methods have serious shortcomings and limitations for this purpose.

The first method is the use of empirically derived weight-estimating relationships (WERs). These are derived by postulating which configurational variables influence the weight and then using regression analysis applied to existing vehicles to get the explicit form of the relationship. The advantage of this approach is that the resulting WERs are very simple and require minimal input; for example, typically nothing about the structural concept need be known.

There are two major problems with using WERs in a vehicle synthesis study. First, the method only can be used reliably to estimate the weight of vehicles similar to those in the database upon which the WERs are derived. For example, the most commonly available WERs are based on low-temperature, aluminum-alloy, skin-stringer-frame structure of vehicles with circular cross sections, and these WERs are appropriate for this class of vehicles. This is a serious limitation for advanced vehicle studies, in which many of the designs under consideration might not be of this type. The other disadvantage is that the standard WERs cannot be used to assess the effect on weight of key design options such as different structural concepts and materials, information of key importance to designers.

The second common method of structural weight estimation is finite element analysis. This approach has become the standard for obtaining an accurate structural weight estimate at the detailed design level and has been developed for use in preliminary design as well. This method, however, is rarely applicable for prepreliminary design or use in a synthesis code because all the structural geometry required for input is not generally available at the earliest stages of vehicle definition.

What is needed is a structural weight-estimating procedure that has the following features: 1) it is useful for a wide variety of vehicle sizes and shapes, including configurations with noncircular cross sections; 2) it depends directly on structural material properties; 3) it depends on the structural concept but does not require detailed structural dimensions; 4) it depends on overall vehicle dimensions and geometry; and 5) it depends on vehicle loads. Such a procedure would not only provide a weight estimate but would allow design engineers to evaluate key design tradeoffs and assess the impact of advanced concepts and technologies.

To meet this need, a procedure for estimating the body weight of launch vehicles has been developed over a period of many years. The procedure is described in several NASA reports and conference papers[1–8] but never has been published in an archival journal. The method is used in the NASA Ames Hypersonic Air Vehicle Optimization Code and in the AirCraft SYNThesis computer code.

In this Note, we focus on one aspect of the structural analysis—the sizing of body structure to preclude failure by buckling. This is the most important part of the weight-estimating procedure, first

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