Time Domain Aeroelastic Solutions – A Critical Need for Future Analytical Methods' Developments

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ABSTRACT

The design, development, and testing of today's aircraft involves the close integration of many diverse technologies and systems which is expected to become more and more complex in the future. This is particularly true for fighters which must be light weight, highly maneuverable and multi- roll in order to meet customer requirements. In addition, these aircraft routinely operate under very transient conditions at the edges of their envelopes – which are highly non-linear and unsteady states – that can only be effectively modeled in the time domain. Thus, the purpose of this paper is to present a strong case for increasing the efforts to more fully develop the capabilities to perform time-domain aeroelastic simulations that can bring together the many diverse technologies which are employed in today's and future aircraft. To support this case, this paper will present four major sections covering historical perspective, problem areas, path dependency and impact on aircraft design and development. It will be concluded that following the evolvement of aeroelastic capabilities from high aspect ratio wings (1920's – 1940's), to finite wings (1950's – 1970's), and to inclusion of controls (ASE, 1970's – 1990's), the technology is at the next major stage of development built on progress achieved since the 1980's – time domain methods.

1.0 INTRODUCTION

The design, development and testing of today's aircraft involves the close integration of many diverse technologies and systems which is expected to become more and more complex in the future. This projection is based on historical trends that have shown that the increase in integration complexity is driven by the growth of both aircraft on-board data processing power as well as computational modeling capabilities for engineering designs. However, a pitfall stemming from this growth is that small changes to one or more systems can have a significant domino effect on some or all of the other systems. Current analysis and design tools can account for some of this interaction, but the process is still a slow and cumbersome procedure that depends on iteration of systems capabilities.

Evaluations of static and dynamic aeroelastic characteristics as well as aeroservoelasticity (ASE) stability for an aircraft design currently depend on stable flight conditions for static effects and steady frequency domain solutions for dynamic effects. Handling qualities for aircraft designs are evaluated with time domain solutions using analytical models that do not incorporate real time aeroelastic effects. Evaluations of handling qualities for more mature configurations are accomplished with man-in-the-loop simulators that require significant capital and manpower investments to realize, but also do not use real time aeroelastic effects. Such tools can be adequate for the design and development of transport and commercial aircraft which tend to operate at more stable flight conditions.

Fighter aircraft which must be light weight, highly maneuverable and multi-roll in order to meet customer requirements, are not "steady state" platforms, and operate under the most taxing flight conditions. These configurations are subject to highly transient conditions at the edge of their design envelope and are at risk when the systems are developed based on stable flight conditions. Problem areas during high rate maneuvers include determination of peak loads and ASE stability where the flight control laws are rapidly changing. These problems are even more critical for unmanned aerial vehicles (UAV's) and micro vehicles where maneuver rates can be much higher due to the absence of human factor limitations. Aeroelastic and unsteady effects on flight paths during high rate maneuvers can, as a minimum, affect lift, drag and dynamics. Limit cycle oscillations (LCO), buffet, and control surface buzz are also non-linear, time domain phenomena that are not adequately treated in the frequency domain. Store ejection, both from external and internal carriage locations, is a highly transient and dynamic problem area that also involves rapid control law changes during the ejection event. An even more transient and potentially catastrophic flight event is that of wake vortex encounters such as those that occur during air-to-air combat "tail-chasing" as well as landing/take-off and approach/climb-out conditions.

The common features of the above phenomena are that each is manifested in the time domain which may or may not be path dependent, and that all involve some form of aeroelastic effects. Thus, the need for aeroelastic solutions in the time domain currently exists and will become more critical with future aircraft developments. Besides modeling the above phenomena, such time domain solutions are useful in diagnosing flight encountered anomalies as well as assisting in planning and optimizing flight test programs. Modeling for obtaining these solutions involves: linear aerodynamics (indicial, pade, etc.); non-linear aerodynamics (CFD, semi-empirical, etc.); linear structures (modes, stiffness matrix); non-linear structures (large deflections); propulsion; 6-DOF equations with trimming; and flight control laws. Time marching solution schemes can range from uncoupled (buffet), to partially coupled (dynamic and maneuver loads), and fully coupled (flutter, ASE, buzz and LCO).

The impacts on aircraft design and development by the availability of time domain aeroelastic solutions is multi-faceted. These impacts include but are not limited to: early detection of potential ASE and transient problems; improved aircraft maneuver capability; better identification of maximum design loads; improved buffet and fatigue assessments; and flight test support. Currently, time domain aeroelastic solutions are applied to LCO and buffet as well as to transonic flutter for limited demonstration cases. Linear and semi-empirical aerodynamic based time domain solutions are more feasible in the near term. Non-linear CFD based solutions are not yet practical for design purposes due to the long turn-around times; however, with the rapid development of computing power, such methods will become more attractive and practical within the next 5 to 10 years.

In summary, the critical need for time domain aeroelastic solutions arises from several factors: (1) the design, development, and testing of today's aircraft involves the close integration of many diverse technologies and systems which is expected to become more and more complex in the future; (2) evaluations of static and dynamic aeroelastic characteristics as well as ASE stability for an aircraft design currently depend on stable flight conditions for static effects and steady frequency domain solutions for dynamic effects which are not adequate for the highly transient conditions that fighter aircraft must operate under; (3) path dependency for transient conditions and nonlinearities can only be accounted for in the time domain; and (4) impacts on aircraft design and development by the availability of time domain aeroelastic solutions are multi-faceted.

These factors will be discussed in the following sections covering historical developments, problem areas, path dependency, and impacts on aircraft design and development. Because of the association of the first author with the long history of the F-16, starting with the original proposal for the YF-16, many of the examples cited in this paper will be based on his experiences with that aircraft.

2.0 HISTORICAL DEVELOPMENTS IN AEROELASTICITY

The projected increased complexity in the close integration of diverse technologies and systems is based on historical trends that have shown that integration complexity is driven by the growth of demands for higher aircraft performance as well as both greater on-board data processing power and computational modeling capabilities for engineering designs. These trends extend all the way back in time to the beginning of aeroelasticity.

2.1 Advent of Mono-Wing Aircraft in World War I, 1917

A brief history of the early evolvement of the science of aeroelasticity is presented in Ref. 1. The development of the first high performance mono-wing fighter, the Fokker D-8, during World War I, provided the German air force with a formidable weapon that was given to its top pilots. However, it had a problem that resulted in catastrophic wing failures during high speed dives, the source of which was not understood. After considerable testing and analysis, Anthony Fokker correctly surmised that these failures were due to what is now known as static aeroelastic torsional divergence. The D-8 also experienced wing bending-aileron flutter which was initially solved by aileron mass balancing but later understood as a true flutter mechanism. With further developments of cantilever mono-wing aircraft during the 1920's and 1930's, research into aeroelastic phenomena progressed until by the mid 1930's, both static and dynamic aeroelasticity were sufficiently understood well enough to be used in the design of aircraft structures.

2.2 First Supersonic Flight, 14 October 1947

Figure 1 B-58 Aircraft

The next phase of aeroelastic development began on 14 October 1947 with the first supersonic flight by Chuck Yeager in the Bell X-1. However, practical supersonic flight required that wings and tails be swept back and that aspect ratios must be much smaller. Thus, the existing aeroelastic methods developed for high aspect ratio wings were not adequate for such aircraft as the B-58 (Figure 1) whose development began in 1951. Research at NACA by 1955 led to the oscillatory subsonic kernel function for finite wings (Ref. 2). A systematic method for application of this kernel function to calculate subsonic unsteady pressures for flutter analysis was available from NASA by 1959 (Ref. 3). A similar method was developed at MIT by 1957 (Ref. 4). Also at MIT, the supersonic Mach box method was developed by 1955 (Ref. 5).

The 1960's and 1970's produced a wide range of oscillatory aerodynamic methods for use in aeroelastic analyses. The subsonic doublet lattice method for single wings was published in 1969 (Ref. 6) and for multiple wings and bodies in 1971 (Ref. 7). Subsonic and supersonic kernel function collocation methods for multiple surfaces were published in 1971 (Ref. 8) and 1974 (Ref. 9). All of these methods have been used for the flutter analyses of the F-16 family of aircraft, especially Ref's 8 and 9.

Another development that would lead to a significant new form of aeroelasticity began after World War II. This new technology evolved into the stability augmentation systems concept (or SAS) that became widely used in all forms of aircraft. The function of SAS was to provide automated flight controls inputs that would improve stability margins and ride quality as well as reduce task loading on the pilot. For such aircraft as the B-58, this was a mandatory requirement due to its Mach 2.0 capability where pilot reaction times would be too long to respond to aircraft needs.

During the 1960's, auto-pilot technologies were developed to a high degree of sophistication with such aircraft as the F-106B and the F-111 (Figure 2). The F-106B was equipped with an on-board system that could fly an intercept mission from take-off to encounter by translating data from ground control intercept station into aircraft flight commands (Ref. 10). The F-111 terrain following capability (TFR) used auto-pilot or manual flight with signals to the pilot on the cockpit display, to fly at very low altitudes for radar avoidance and follow the local terrain at "tree-top" level.

Figure 2 F-111 Aircraft

2.3-YF-16 Fly-By-Wire, 1973

These technologies – SAS, auto-pilot and TFR, among others – were not directly connected with aeroelastic developments, however they did set the stage for what evolved into aeroservoelasticity during the 1970's. Based on the success and experience gained with these new systems, feasibility for the use of active controls with structural feedback to reduce aircraft dynamic response was investigated in the early 1960's and demonstrated on the XB-70 (Ref. 12) and B-52 (Ref. 13) aircraft. This feasibility was further enhanced by the miniaturization of electrical components in the space program that would be needed for lightweight on-board analog computers capable of providing the data processing power needed to drive such active systems. As a result, the B-52 control configured vehicles (CCV) program was initiated by the U.S. air force in 1971 under a contract to the Wichita division of the Boeing Company (Ref. 14). Under this program four CCV concepts were flight demonstrated: Ride Control (RC); Flutter Mode Control (FMC); Maneuver Load Control (MLC); and Augmented Stability (AS) (Ref. 15).

Design, development, and flight testing of the YF-16 (Figure 3) prototype aircraft put this new type of technology into a highly maneuverable, lightweight fighter that was the first to be designed as a fly-by-wire aircraft. Not only did it demonstrate the superior capabilities achievable with the technology, it provided a test bed for defining new areas of research needed to develop methodologies for analytically modeling the complex interactions of flight control systems and aeroelasticity. Finally, the success of the F-16 production program and incorporation of fly-by-wire technology into following aircraft such as the F-22 and F-35 further proves the practicality and advantages of the concepts.

Experience gained with the YF-16 flight testing revealed that the broadband capability of the flight control system designed for maneuver control could interact with the closely spaced structural modes of fighter aircraft in ways that differed from earlier experiences in the B-52 CCV program. The YF-16 flight control system consisted of three channels: pitch, roll, and yaw. All channels were stability augmenting which permitted operation with negative static margin. The pitch channel had AOA and *g* limiters independent of pilot inputs. As a result, a study contract was issued by the U.S. Air Force to the Fort Worth division of General Dynamics (now the Lockheed Martin Aeronautics Company) to investigate two observed instabilities and to evaluate methods for analytically modeling the phenomena (Ref. 16).

Figure 3 YF-16 Aircraft

In Reference 16, three structural representations for the YF-16 were evaluated using truncated GVT modes, truncated computed modes, and the residual flexibility method with computed modes. The latter provided more accurate treatment of static aeroelastic effects on the stability derivatives whereas the two other methods were more computationally efficient – this was an important factor in the 1970's. Three stability techniques were evaluated: Nyquist stability criteria, determinant plots, and root locus. The first two were frequency domain techniques and the latter was based on Laplace domain solutions.

In addition to the analysis technique evaluations, recommendations were made in Reference 16 for proper ground testing techniques as well as flight testing procedures. In writing the final report, it became apparent that a new name was needed for the interaction of flight control systems and aeroelasticity. One of the authors of that report, A. M. Cunningham, Jr., first suggested "servoaeroelasticity," but since he was more of an aerodynamicist, quickly changed it to "aeroservoelasticity" which greatly simplified the report writing. This term was quickly picked up by personnel at the Air Force Flight Dynamics Laboratory and has been used ever since.

This brief historical discussion is presented to make the point that major changes in aircraft performance demands have driven major changes in aeroelastic modeling capabilities as listed below:

1. high performance mono-wing aircraft – 2-D aeroelasticity
2. supersonic flight – 3-D aeroelasticity
3. active controls – aeroservoelasticity.

Since the early 1980's, the appearance of LCO and other non-linear aeroelastic phenomena have motivated significant research efforts into time domain modeling capabilities. These efforts have included both semi-empirical and CFD based methods and are pointing the way to future aeroelastic developments. But first, current problem areas will be discussed in the next section to highlight more specific needs for time domain aeroelastic solutions.

3.0 PROBLEM AREAS

Evaluations of static and dynamic aeroelastic characteristics as well as ASE stability for an aircraft design currently depend on stable flight conditions for static effects and steady frequency domain solutions for dynamic effects. However, these analyses are not adequate for the highly transient conditions that fighter aircraft must operate under. This section will present some examples of such problem areas that include: high rate transients; LCO, buffet and buzz; and wake vortex encounters.

3.1 High Rate Transients

Fighter aircraft are routinely subject to highly transient maneuvers at the edge of their design envelope and are at high risk when the systems are developed with tools based on stable flight conditions. Problem areas during these maneuvers include determination of peak loads and ASE stability where flight control laws are rapidly changing. These problems are even more critical for UAV's and micro vehicles where maneuver rates can be much higher due to the absence of human factor limitations.

Various factors can affect the performance of flight control laws during high rate transients. System lags introduced by filters, limiters, data processing and actuator response characteristics – among others – can diminish performance to the point that, in cases of extreme degradation, the system can become unstable. An example of a non-critical anomaly during a transonic high –g roll maneuver by an F-16 is shown in Figure 4 (Ref. 17). Normal load factor is plotted as a function of roll rate where the leading edge flap position is denoted at each data point as it schedules from 7 deg to 10 deg during the maneuver.

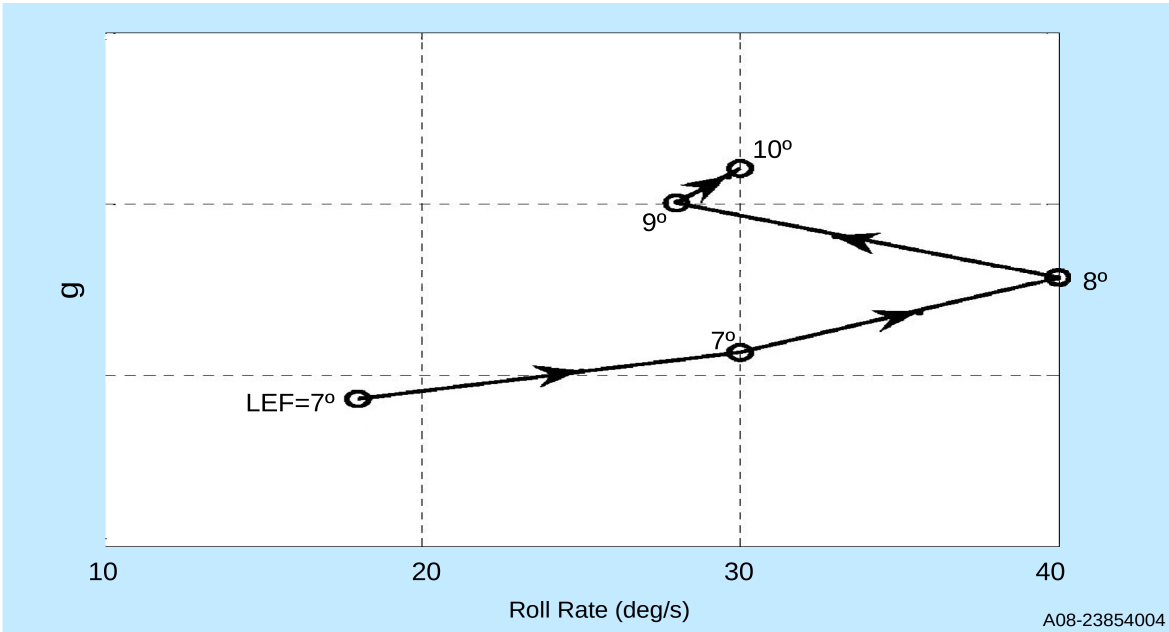

Figure 4 F-16 High-g/Roll Rate Anomaly With L.E. Flap Scheduling (Ref. 17)

The aircraft was rolling right-wing-down (RWD) while pulling high symmetric g's at increasing AOA. During this maneuver, the five data points shown in Figure 4 occurred over a 1 sec period. The AOA range plus the roll induced incidence corresponded to conditions where the wing tip flow is separated due to shock induced separation. At the maximum roll rate of 40 deg/sec., the R.H. wing would be at a slightly higher incidence of about 1.0 deg than the L.H. wing and would be more subject to wing tip flow separation. If the flow over the R.H. wing was separated for the leading edge flap at 7 deg (1^{st} two data points) then scheduling the flap from 7 deg to 10 deg would cause the flow to re-attach. This re-attachment of the R.H. wing would produce a rapid increase in g level since its downward force would be reduced. This would also result in a significant reduction of roll rate such as actually occurred in 0.25 sec from 40 deg/sec to 28 deg/sec. Thus, this anomaly was attributed to re-attachment of the downward rolling R.H. wing's flow field as the leading edge flap scheduled from 7 deg to 10 deg.

This original YF-16 and F-16 flight control systems were all analog. The control laws were all designed for smooth operation with analog components. In the late 1980's, the improvements for digital data recording and processing led to the incorporation of an all digital flight control system where all logic was designed to the same functionality as that of the analog system. The prototype for this improvement was successfully developed and tested under the AFTI/F-16 program (Refs. 18, 19, 20). Of the many lessons learned from that program, it was found to be necessary for the digital control laws to account for the implicit lag introduced by the digital sampling of the input aircraft data and feedback to the flight control computers. With final adjustments, the new digital system worked as well as the original analog system and was deemed acceptable for production F-16's.

Aircraft vibration modes can also have effects on flight control sensor response, especially rate sensitive items (Ref. 17). In the initial design of the YF-16 flight control system, roll loop gain was set too high which resulted in violent roll oscillations for the aircraft during its first high speed taxi. The test pilot took off in order to regain control of the aircraft and landed it safely after he diagnosed the problem. As a result of this occurrence during "Flight 0", the roll gain was reduced to 20% of its design value. In addition, the

F-16 flight control system currently has several notch filters installed to de-couple the roll loop from the first anti-symmetric wing bending modes for certain store carriage configurations. For another set of configurations, during aircraft buffeting the pitch rate gyro sensitivity of a later version of the F-16 was found to be affected by coupling with a higher order fuselage vertical bending mode, for which modifications were required.

Store ejection is a different class of transient activity in which both the aircraft and the ejected body interact in ways that can be quite violent and sometimes catastrophic in consequences (Ref. 17). It is well known that release of heavy stores – such as 2000 pound class weapons – results in significant response of the aircraft due to the change in inertial forces alone. For aircraft like the F-16 which have control laws that are configuration sensitive, release of these heavy stores requires a rapid change in the control laws as directed by the on-board stores management system. The dynamic aspects of stores release can and usually do define the envelopes for such operations.

In addition to the effects on the aircraft, the stores trajectories are obviously affected by their own set of forces. The aerodynamic coupling of aircraft and store CFD with store ejection and missile exhaust plumes must be considered for analytical models. Reference 21 is typical of the types of these developments currently under way. A bending beam approach discussed in Reference 22 accounts for the flexibility of slender ejected stores as they respond to the high accelerations due to the shock-like loads of the ejectors. Modeling the effects of fuel sloshing during the jettison is considered in Reference 23 where solutions are vividly compared with and without sloshing.

3.2 LCO, Buffet and Buzz

These problems are non-linear in nature by virtue of the driving phenomena, although buffet can be linearized under conditions where structural response has no effect on the turbulent flow that provides the forcing function. References 24 and 25 cover many examples of LCO, buffet and buzz, which are referred to as "The Aeroelastician's Nightmares." This title is in reference to the fact that they tend to be discovered for a new or modified aircraft during initial wind tunnel or flight testing or at worst in field operations after full scale production is underway. Several F-16 examples will be described in the following discussions.

LCO for the F-16 with certain heavy air-to-ground stores carriage exhibits a trend that is not driven by transonic effects as are the air-to-air configurations (Ref. 24). It is very persistent and not explosive; however, structural limits impose restrictions on permissible amplitudes of response. Pilot discomfort is another limiting factor depending on an individual's tolerance. Examples of flight time history recordings are shown in Figures 5 and 6 for the wing tip launcher forward accelerometers for the R.H. and LH wings (Ref. 17). In Figure 5, the persistence of the 5Hz LCO is illustrated before, during and after flutter excitation with the flaperon. The 4 sec bulging during excitation has no effect on the LCO as the amplitudes before and after excitation are essentially the same.

Shown in Figure 6 is a characteristic that has been observed repeatedly, denoted as a "bottoming" or "g-clipping" effect of the 5Hz LCO that occurs at higher g's during wind-up-turn maneuvers (Ref. 17). The peculiar nature of this effect is that although the two traces for the left and right hand forward launcher accelerometers are 180 deg out-of-phase – as expected from the LCO anti-symmetric modal response – but the "clipping" is symmetric. This occurs where the LCO g level hits its positive peak on either side when the tip missile is in the nose-down position. With initial onset of "clipping" the negative g peaks are unaffected.

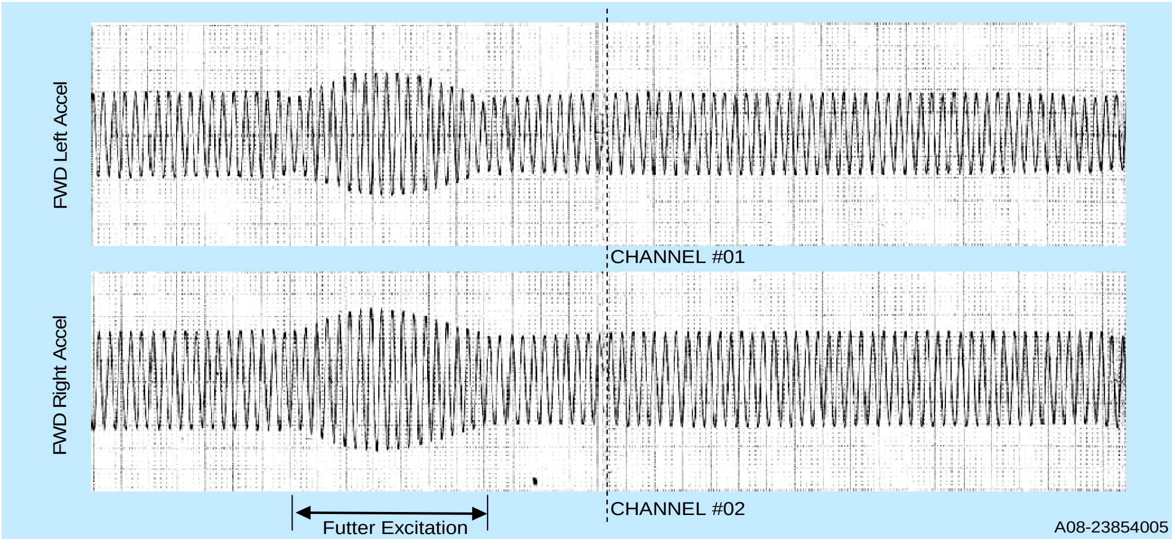

Figure 5 Time History for Launcher Accelerometers for the F-16 5 Hz LCO With Wing Tip Missiles (Ref. 17)

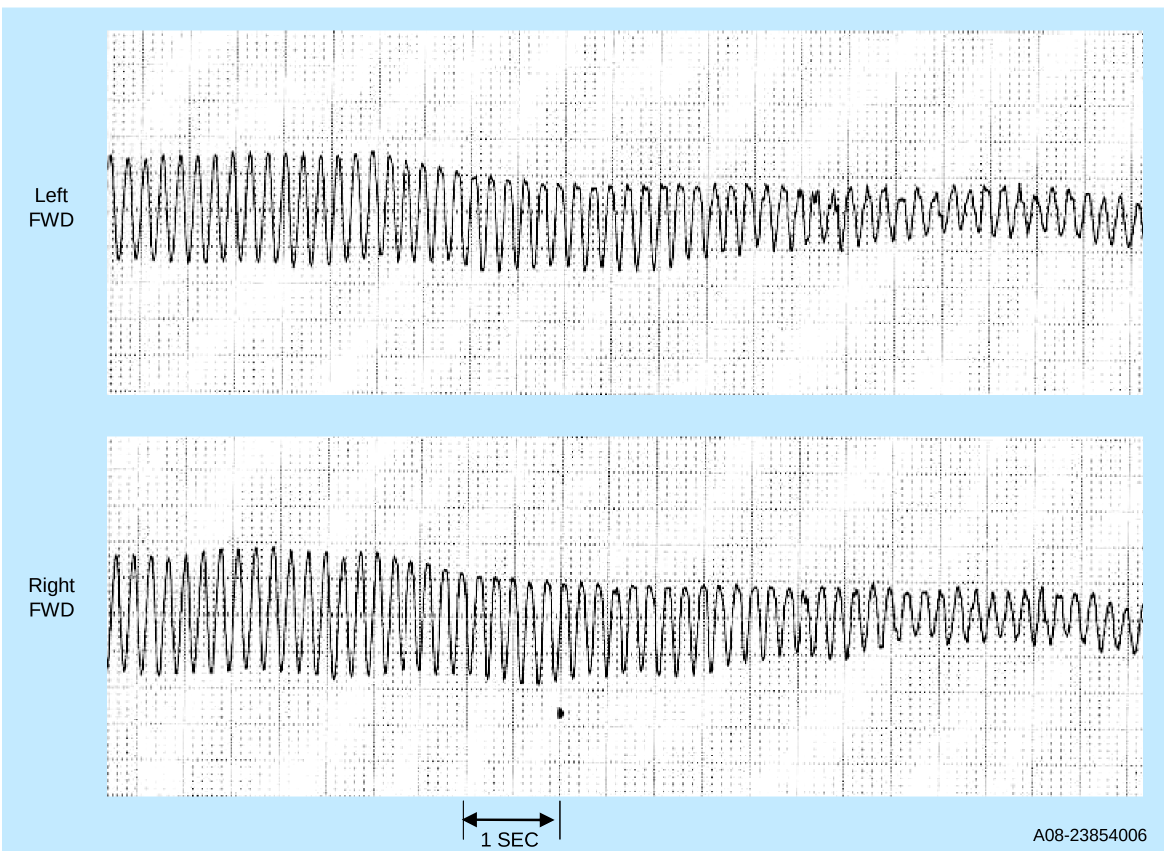

Figure 6 Time History for Left and Right Forward Launchers' Accelerometer Response for F-16 5 Hz LCO, Showing "g-Clipping" Effect (Ref. 17)

The small higher frequency fluctuations seen in the flattened clipped segments of the traces are indicative of a bottom impact response that may occur in the relative motion between the wing-tip and launcher at the attachment fitting. This possibility is backed by the fact that the tip missile CG is forward of the launcher attachment to the wing tip where high positive g's will provide both a high downward shear force and nose down pitching moment to the attachment bolts. These observations are consistent with the existence of relative motion in the attachment that is limited at the extremes of clearance and material strain build-ups. The issue of friction in the wing/launcher interface will be discussed later in this paper.

Buffeting of the F-16 ventral fins has provided a classic example of structural fatigue of these aerodynamic surfaces by an upstream source of severe turbulent wakes (Ref. 24). The location of the key elements on the F-16 is shown in Figure 7 where the two LANTIRN pods are directly upstream of the two ventral fins. During the early 1980's, requirements were imposed to carry two LANTIRN pods just aft of the inlet on the lower fuselage upstream of the ventral fins. Although work was done to reduce drag and wake turbulence from these pods, the first flight with LANTIRNS installed resulted in the loss of the R.H. ventral fin as shown in Figure 8.

Figure 7 LANTIRN Pod Carriage on the F-16 (Ref. 24)

Figure 8 F-16 Ventral fin Damage on First Test Flight With LANTIRN Pods (Ref. 24)

Originally, the primary source of the fin's fatigue and loss was high speed throttle chops that produced severe turbulence from inlet lip spillage during rapid decelerations. A comparison of the ventral fin response to LANTIRN and throttle chop turbulence is shown in Figure 9. The levels are about the same, however, constant buffeting by the LANTIRNS produced much higher fatigue damage per flight hour as compared to that due to the transient throttle chops. Several redesign efforts have been conducted to improve fatigue life of the fins as will be discussed later in this paper.

In 1991, an F-16 test aircraft developed excessive free-play in the rudder that grew with time (Ref. 24). This aircraft was subjected to extreme buffeting loads on the vertical tail as a result of conducting 28 loads flights with test conditions that required 30 second runs with full rudder input for high speed wings-level side-slips. Rudder free-play began to develop during these 28 flights and grew to the point that after these were completed, the free-play continued to develop with further flights. A measure of the free-play was found in a particular "burst signature" of the vertical tail aft tip accelerometer as shown in Figure 10 which correlated well with measured rudder free-play. The similarity is clearly seen between these bursts and LCO in Figure 5. Intermittency of the bursts was a result of neutral rudder position being disturbed by rudder commands and/or atmospheric turbulence.

Figure 9 Ventral Fin Buffet Response to LANTIRN Pod Turbulence and Inlet spillage Turbulence (Ref. 24)

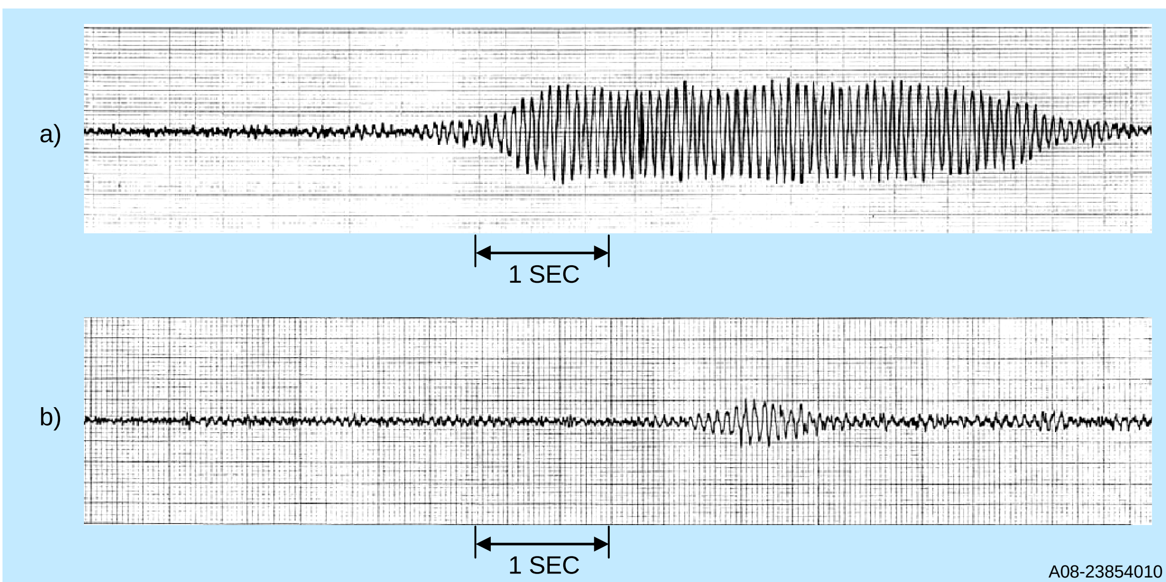

Figure 10 (a) High Amplitude, and (b) Low Amplitude F-16 Rudder Buzz Oscillations for Fin Tip Aft Accelerometer (Ref. 17)

Based on the g-level estimated from the signatures, a history of the free-play growth was constructed and is shown in Figure 11. The high-speed side-slips were conducted during flights 67 through 95 which were accompanied by increases in free-play. After flight 95, the free-play continued to grow exponentially until flight 135, at which time the level was considered to be a flight safety risk. After that flight, rudder actuator bolts were tightened, bushings replaced, etc and free-play was thus reduced to about twice the mil-spec value. As a result, vibration levels from flight 137 returned to the normal range as shown in Figure 10.

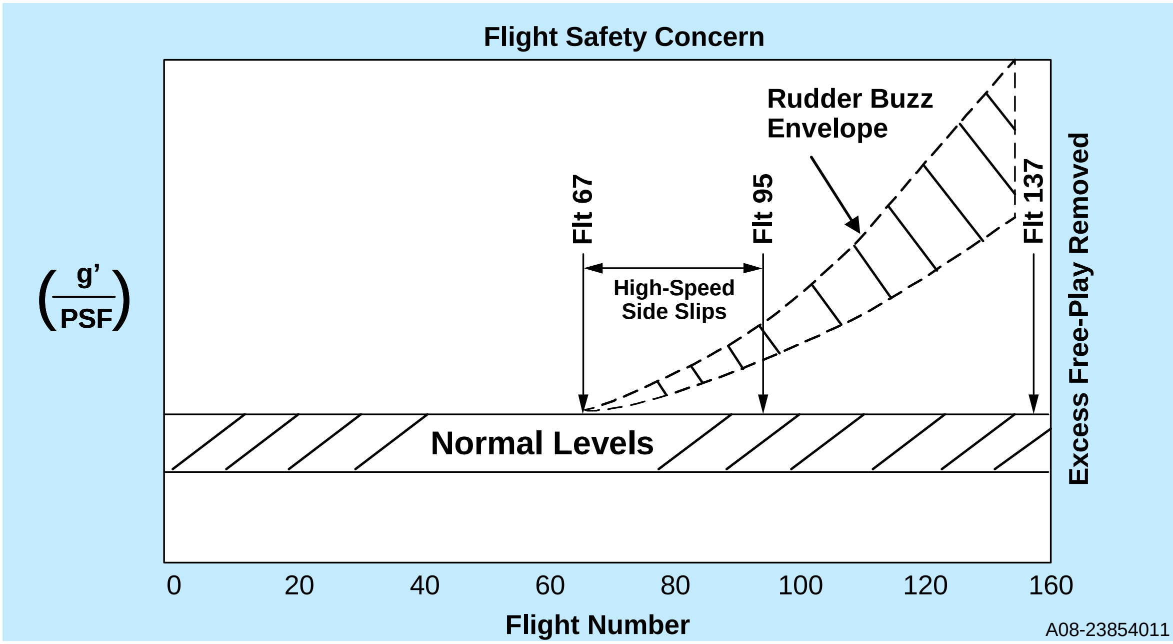

Figure 11 F-16 Rudder Freeplay Development With Flight Time (Ref. 24)

3.3 Wake Vortex Encounters

The encounter of wake vortices can range from an annoying bump to structural damage to catastrophic loss of aircraft. A well known example of the latter case was the loss of an Airbus A300, American Airlines Flight 587, on 17 November 2001. Various newspaper articles have covered this event quoting the NTSB findings that the catastrophic loss of the vertical tail was due to pilot reactions to severe turbulence created by a preceding 747 on climb-out.

The first author of this paper has recently experienced two distinct encounters on commercial aircraft. One was an occurrence in clean air of a rapid RWD roll of about 25 deg from wings level to a LWD roll of about 45 deg back to wings level in about 1.0 sec. This occurred during a downwind run at about 5000 ft in an approach to DFW Airport in 2005. Based on aircraft response, it was concluded that the encounter occurred as schematically shown in Figure 12. A second encounter occurred in 2007 during a return trip from Europe over the North Atlantic at about 35k ft. A hard "double bump" was noted in clean air which, in talking to the co-pilot on the ground after the flight, occurred when the aircraft crossed in another aircraft's wake. The second aircraft was about 20 miles away at the time of crossing which is indicative of the persistence of these wake vortices.

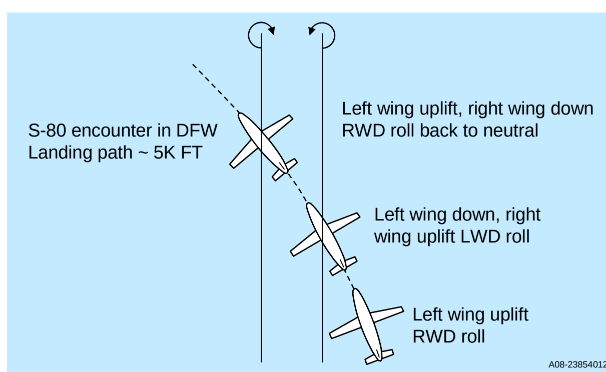

Figure 12 Hypothetical Wake Vortex Encounter (Ref. 17)

Several incidents have occurred over the past 25 years where structural damage was experienced on F-16's during tail chasing air-to-air exercises. One wing tip missile was lost during the early 1980's and four vertical tail tips have been lost since, as typically shown in Figure 13 (Ref. 17). One tip loss case that occurred in the mid 1990's was analyzed for the possibility that wake vortices from the lead aircraft, an F-15 in that case, caused the structural failure. With flight recorder data available for this incident, it was possible to confirm that this was indeed the source of the failure (Ref. 26). Traces for rudder and symmetric horizontal tail motions shown in Figure 14 indicate that the duration of the occurrence was about 0.5 sec. The motions were commanded by the flight control system in response to aircraft reactions to the vortex encounter. The pilot did not have any indication that the failure occurred until his wing man informed him of the damage. This is consistent with the short duration of the encounter and with the mild aircraft response during the incident.

Figure 13 F-16 Vertical Fin Wake Turbulence Encounter (Ref. 17)

An analysis was conducted to determine if wake vortex-induced loads would be high enough and of the proper character to produce the structural failure. With the F-16 flight path data, assumptions were made with regard to the lead aircraft from which a wake vortex structure was calculated. Corresponding loads were developed for the vertical tail, which when applied to a structural model of the tail did indicate a high probability of a failure that was in agreement with that shown in Figure 13 and had also been observed in other similar incidents.

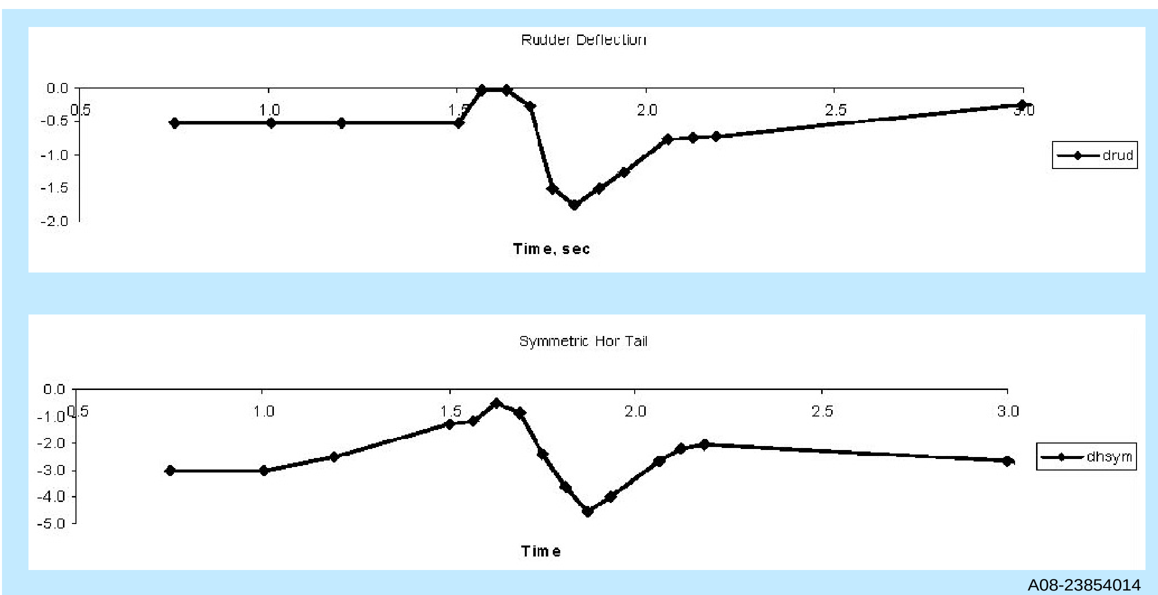

Figure 14 Expanded Response History for Suspected Wake Encounter (Ref. 26)

An example of aircraft and pilot response to encountering wake turbulence during landing is shown as a time history of AOA and roll angle variations in Figure 15 (Ref. 17). This incident occurred as an F-16 touched down on the runway following another F-16 that had landed less than one minute before and was initiated with a loss of lift on the right wing. Pilot compensation with LWD roll input led to over control to the left, countered with RWD roll, etc. as shown in Figure 15 for about 5 cycles after which landing was aborted. A cross-wind of about 2.25 to 3.6 knots from the right was suspected of keeping the right hand wing tip vortex of the lead aircraft from spreading to the right as normally occurs with little or no cross-wind. The downwash that would exist to the left of the right hand vortex was probably the cause of loss of lift on the right wing of the following F-16 which initiated the sequence of events seen in Figure 15.

The above examples of wake vortex encounters illustrate the typical range of severity of such incidents. Unless visualized by water vapor condensation, these vortices are invisible, yet they are highly concentrated and very persistent bundles of energy. The domain of danger is quite small which attests to the infrequent number of events that occur. Landing and take-off events are many times a result of violations of timing requirements for a following aircraft as set by FAA for commercial and various specifications for military aircraft. Tail chasing combat maneuvers are likewise hazardous, but such wake encounters are infrequent again due to the small domain of danger and therefore should not be added to the pilot's work load under air-to-air combat conditions.

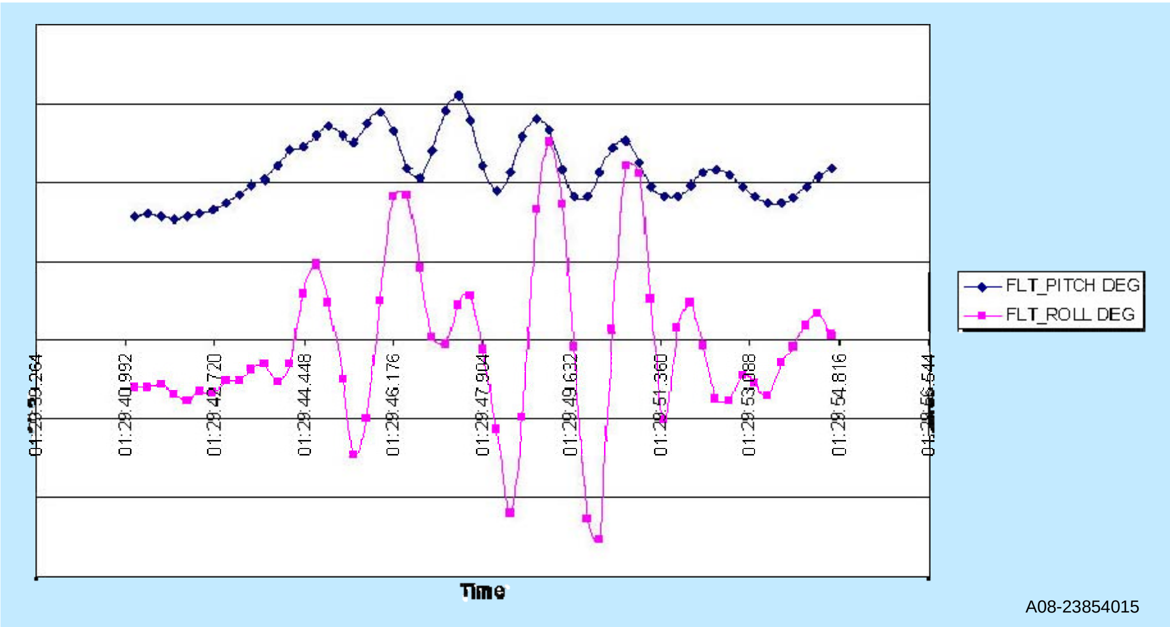

Figure 15 Pitch and Roll Versus Time for F-16 Landing Incident (Ref. 17)

4.0 PATH DEPENDENCY

Although all of the problem area examples presented in the previous section have either path dependency or involve highly non-linear phenomena, wake vortex encounters are probably the most sensitive. This sensitivity is affected by the initial aircraft state prior to encounter and the exact characteristics of the flight path/vortex trail crossing. These factors in turn affect how the flight control system and pilot reactions respond to the encounter, ranging from pilot and flight control system over reacting (A300 loss and F-16 oscillations during landing) to pilot being unaware of the loss of his vertical tail tip.

This section will address the path dependency characteristics common to the problem areas discussed in the previous section as well as some specific examples where path dependency affects non-linear aerodynamics during maneuvers.

4.1 Commonality

The common features of the phenomena discussed in the previous section is that each is manifested in the time domain which may or may not be path dependent, and that all involve some form of aeroelastic effects. Thus, the need for aeroelastic solutions in the time domain currently exists and will become more critical with future aircraft developments. Besides modeling the above phenomena, such time domain solutions are useful in diagnosing flight encountered anomalies as well as assisting in planning and optimizing flight test programs.

The rapid changes in control laws during highly transient maneuvers where both static and dynamic aeroelastic effects are involved, introduce many situations of aircraft dynamics that are not modeled in conventional ASE analyses which are based on stable flight conditions. Lags imposed by flight data sample rates as well as those implicit in various elements in the highly complex control law loops and the mechanical elements (both hydraulic and electrical) become an integral part of the overall actions of the control system. These time lags also become important with regard to structural modes, especially for higher frequency modes due to an increase in effective phase lag. Rate dependent operators in the control laws are even more sensitive to time lags as they are dependent on data that can be affected by structural vibration modes. All of these effects are included only in the form of flex-to-rigid ratios and shifts in aerodynamic centers where as dynamic aeroelastic effects and buffet are not included at all. Again, static aeroelasticity assumes stable flight conditions and does not represent instantaneous conditions where both the structure and aerodynamic forces are transient.

The example of the in-flight anomaly discussed in subsection 3.1 above, falls into the category of highly coupled aerodynamic and ASE aircraft maneuvers. In that case, the longitudinal loop commanded leading edge flap scheduling essentially based on AOA whereas the roll loop commanded flaperon/horizontal tail deflections to achieve pilot instructions. Since this was a high g transonic roll maneuver, wing tip flows on both wings were most likely affected by shock induced separation. With roll induced asymmetric incidence on both wings, the symmetric scheduling of the leading edge flaps would tend to have more influence on the flow separation on the downward rolling wing which as it happened caused a rapid reduction of the roll rate and led to a jump in normal load factor as shown in Figure 4.

The aeroelastic elements involved in this anomaly included “static” and dynamic aeroelasticity under transient conditions with shock induced separation and flow reattachment. These interactions are highly non-linear and it is well known that a sudden flow reattachment at high load levels can induce significant transient dynamic response similar to heavy store ejection. Also, under the conditions of the anomaly, buffeting was most surely involved due to wing tip flow separation. Depending on the wing stores carried, the pilot may or may not have been aware (or bothered) by the wing tip buffeting. However, because data sampling by the flight recorder was only four times per second, dynamic response of the aircraft could not be confirmed.

Store ejection, especially for heavy stores and those that are propelled away from the aircraft by explosive devices, tends to produce airframe response that may be responded to by the flight control system. This response can be either designed into the control laws or not, and can also be a result of changes in control laws dictated by the on-board stores management system. Since these transients are affected by initial state conditions of the aircraft which may be strongly path dependent, time domain solutions are needed that model both aircraft and store motions as well as their aerodynamic and inertial interactions. ASE aspects require that the instantaneous control laws must also be included in these models as well as the vibration modes as they transition from pre-ejection to post-ejection. The vibration modes must include both the aircraft and the store in captive and free-flight states. Obviously, this activity is very path dependent and highly non-linear.

To a certain extent, LCO is not path dependent, however, it does show a tendency to lag in both increasing and decreasing conditions due in part to total system damping. The design of suppression systems to reduce the amplitudes must also consider transient conditions that can become more critical for rapid maneuvers in addition to store ejection. The need for time domain aeroelastic solutions for modeling LCO is primarily driven by the non-linearity of the elements involved in these phenomena which include non-linear aerodynamics and non-linear structural behavior such as friction based damping, both static and sliding.

The most straightforward approach for analytically modeling LCO in the time domain is the use of fully coupled CFD/FEM methods as typical of that presented in Reference 27. However, such solutions currently require extensive computer resources and long run times to achieve stable conditions that are representative of measured data. Semi-empirical time domain methods such as that of Reference 25 provide a more practical approach for current needs which although limited in applicability, do correlate well with experiment. This will be discussed further in the next section with an example of application of the method of Reference 28.

Buffet response of a structure subject to turbulent flows under stable conditions has been modeled successfully over many years as discussed in References 24 and 25. However, buffeting during rapid maneuvers is highly non-linear due to constantly changing flows that produce the turbulent excitation. In these cases, the solution process can be decoupled if the structure response does not affect the source of turbulence. A good example is a downstream surface buffeted by a wake from upstream such as a bluff body (LANTIRN pod) or a burst strake vortex. If the buffeted surface motion does affect the turbulence source, for which an example is leading edge separation on the buffeted surface, then the solutions should be fully coupled in the time domain.

For rapid maneuvers, the time variation of the turbulent flow intensity can be very significant. As will be discussed later in this section, flow separation is highly path dependent and can be virtually non existent during a rapid pitch-up maneuver well beyond the steady buffet onset boundary. However, on pitch-down, the flow is already highly separated and hence it continues with high intensity to well below the steady onset boundary. If measured pressure data are not available for calculation of such response, CFD provides the only other source for the time varying buffeting excitation. These types of computer runs can currently require large resources and very long run times to model a maneuver. Structural response modeling in this approach is almost trivial to calculate.

Since free-play is an important ingredient for control surface buzz such as that described previously for the F-16 rudder, the most accurate and most versatile approach for calculating buzz is in the time domain. The non-linearity of free-play is easily and more physically modeled in this manner. This also permits the external effects of controls input or atmospheric turbulence to bias the control surface to one side or the other or completely outside of the free-play band. By performing these calculations in the time domain, it is also possible to obtain better estimates of fatigue damage rates for operational conditions. Since control surface buzz is very similar to LCO (compare Figure 5 and Figure 10) the same methods can be used for calculating the solutions which includes both the semi-empirical and CFD based methods.

Path dependency for wake vortex encounters is driven by the aircraft state prior to the encounter and the exact characteristics of the flight path/vortex trail crossing. If the affected aircraft is in a rapid maneuver condition, then its flow fields and structural dynamics – including ASE states – all act as input to the eventual final response that results from the encounter. Likewise the exact nature of the vortex crossing is very path dependent and can result in either a very mild response or a catastrophic loss of the aircraft. This phenomenon is probably the most difficult to model due to the large uncertainties associated with both the geometric aspects of the aircraft path and vortex field as well as the associated aeroelastic and ASE properties of the aircraft.

4.2 Specific Examples

Path dependency is especially vivid for high AOA maneuvers where the flow fields progress from attached flows through vortex development and bursting into the post-stall region. Experimental data obtained during M=0.2 low-speed testing of a straked delta wing under such conditions were reported in References 29 and 30. The model approximated a 1/9-scale version of an F-16 with a simple 76 deg delta strake and a 40 deg wing as illustrated in Figure 16. It was oscillated in pitch at various mean AOA's up to large amplitudes of 36 deg, peak-to-peak, with a total incidence range from -8 deg to 50 deg. The model support was mounted on a turntable as shown in Figure 16 which permitted testing in body-axis pitch at various side-slip angles. The non-dimensional frequency parameter, k, was defined as

$$k = \frac{2\pi f C_r}{2V}$$

where f is frequency in Hz, C_r is wing root chord in ft, and V is free steam velocity in fps.

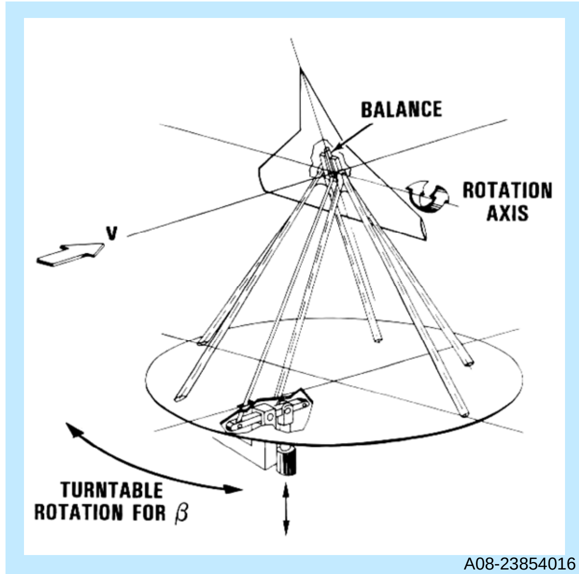

Figure 16 Straked Wing Model Dynamic Support System (Ref. 30)

Data obtained from this testing that best demonstrate path dependency are those for the model pitching with a side-slip angle, β of -5 deg. These data are also more relevant to real aircraft high AOA maneuvers where 0.0 deg side-slip rarely occurs. Variations of steady normal force, $C_{N,}$ and roll moment, C_l, are shown in Figure 17. Pertinent flow regimens are identified and data for both β = 0.0 deg and β = -5 deg are included.

The more interesting results shown in Figure 17 are for the rolling moment, C_l. For the sign conventions in Reference 30, β = -5 deg places the nose to the right and positive C_l is stable which forces the windward wing to roll up to reduce β. In the linear range up to 8 deg, the development of a stable (positive) C_l with AOA is consistent with the lower sweep windward wing experiencing higher lift than the leeward wing. With the onset of vortex flows, 8 to 18 deg, a destabilizing trend develops (negative) C_l where strake-induced vortex lift increases at a more inboard location on the windward wing. Also, the higher sweep of the outboard panel on the leeward wing begins to develop more vortex lift as well. Thus, the leeward wing rolling moment and destabilization increases faster than that for the windward wing. This effect is greatly intensified with the onset of vortex bursting on the windward wing which is fully burst by about 25 deg. The unstable "bucket" from 25 deg to 33 deg is caused by bursting on the leeward wing which tends to balance that on the windward wing as it approaches fully separated flow. Abrupt termination of the "bucket" occurs when the leeward wing rapidly stalls and returns C_l to a stable level.

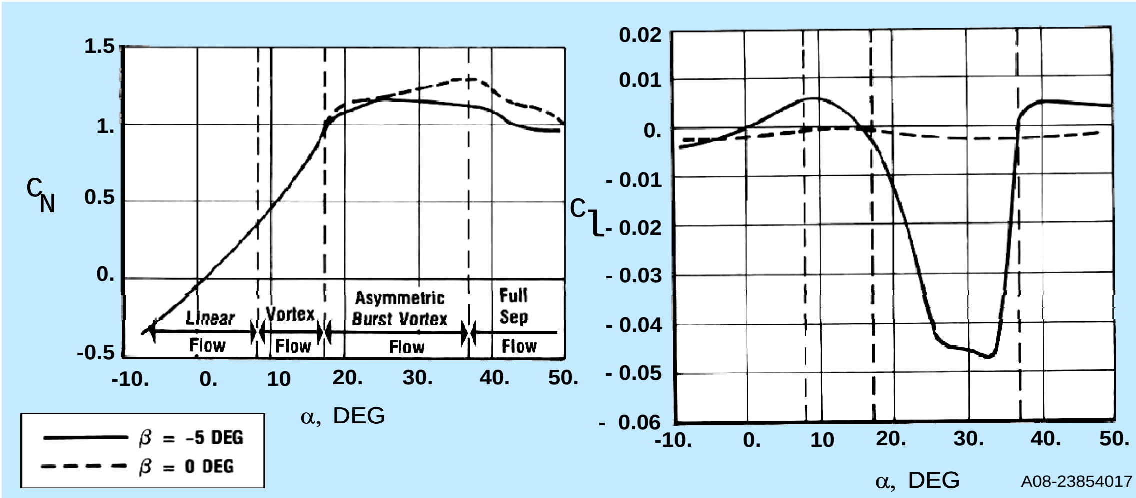

Figure 17 Steady Normal Force and Roll Moment Results for the Straked Wing Model (Ref. 30)

The sweep difference between the windward and leeward wings is -10 deg for β = -5 deg, thus flow transition points occur at different AOA’s which, as just described, leads to the rolling moment characteristics shown in Figure 17. Likewise, lags associated with these flow transitions become important under dynamic conditions. Unsteady C_l variations in the form of hysteresis loops for model pitching at β = -5 deg are shown in Figure 18 where the model AOA is oscillating between 8 deg and 38 deg at two frequencies of k = 0.09 and k = 0.15. Direction of time in the loops is denoted by arrows. For the low frequency case, k = 0.09, the hysteresis loop appears to be a simple lag distortion of the steady “bucket” characteristic. For the higher frequency, k = 0.15. However, the distortion is not so simple and it appears that the leeward wing stalling did not occur as discussed in more detail in Reference 30. Although the dynamic results shown in Figure 18 are for the model oscillating as opposed to performing a transient maneuver, the trends would be very close to each other. Also shown in Figure 18 is a hysteresis loop for a smaller amplitude motion from 12 deg to 29 deg at k = 0.15 which does follow a simple lag distortion of the downward leg of the “bucket”.

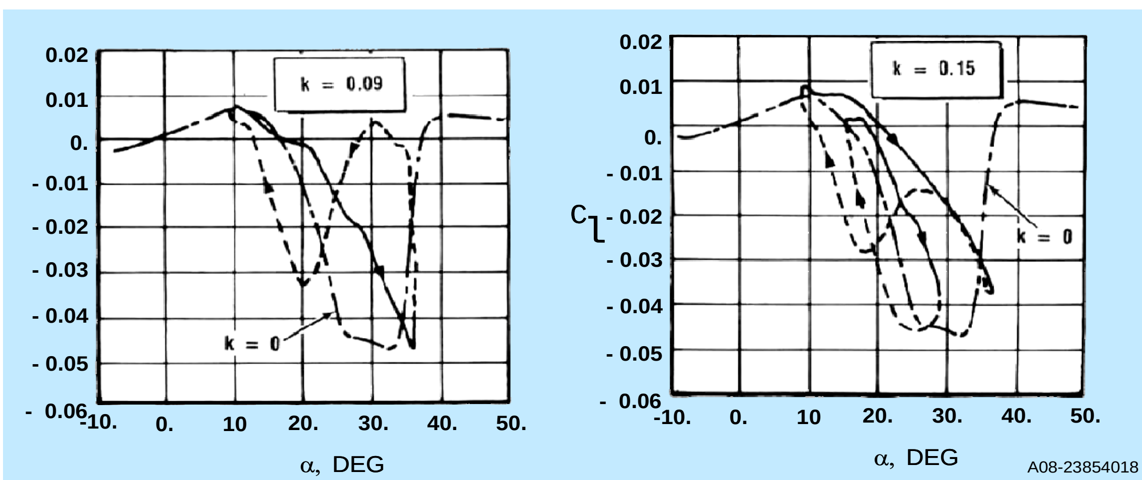

Figure 18 Effect of Frequency and Amplitude on Unsteady Roll Moments for Oscillations from 8 deg to 38 deg AOA for the Straked Wing Model (Ref. 30)

Dynamic effects on C_N from Reference 30 are illustrated in Figure 19 for the same data set at β = -5 deg and k = 0.15. For the incidence range of -4 deg to 24 deg, the two hysteresis loops for the two oscillation amplitudes show very little dynamic effect and closely follow the nearly linear static data. The mid AOA range data for 8 deg to 38 deg clearly illustrate the effect of dynamic overshoot for conditions past the static vortex burst angle of 18 deg. Both hysteresis loops are very similar in shape and have a common path during pitch-up until vortex burst occurs. During pitch-down, both loops follow paths that produce an under-shoot of the static data. At the highest AOA range of 22 deg to 50 deg, both loops enclose the maximum C_N point where the outboard wing becomes fully separated. These two loops are also very similar and exhibit significant over-shoot on pitch-up and under-shoot on pitch-down relative to the static data.

The dynamic data shown in Figures 18 and 19 clearly demonstrate path dependency when major flow transitions are encompassed during a maneuver. The equivalent maximum pitch rates for a full scale fighter are in the 15 to 25 deg/sec range (k = 0.09 and 0.15) and are well within realistic rapid maneuver conditions. The effect on C_l in Figure 18 for the same maneuver performed at 15 and 25 deg/sec would require significantly different control laws where state conditions would not be sufficient to define input data for the flight control computers. This would be especially true for the higher pitch rate. Similar effects of pitch rate on C_N illustrated in Figure 19 lead to the same conclusion. Other forces such as pitching moment and axial drag are affected as well during dynamic conditions as also discussed in References 29 and 30.

Static aeroelastic deflections at high AOA where the flow fields are mostly separated, have minimal impact even at high g's. However, due to persistence of separated flows on pitch-down as seen in Figure 19 for the mid- and high- range oscillations, high intensity buffeting extends to conditions well below buffet onset, depending on the path taken in the maneuver. Control laws that govern ride equality and load alleviation could be affected by such "un-natural" buffet occurrences.

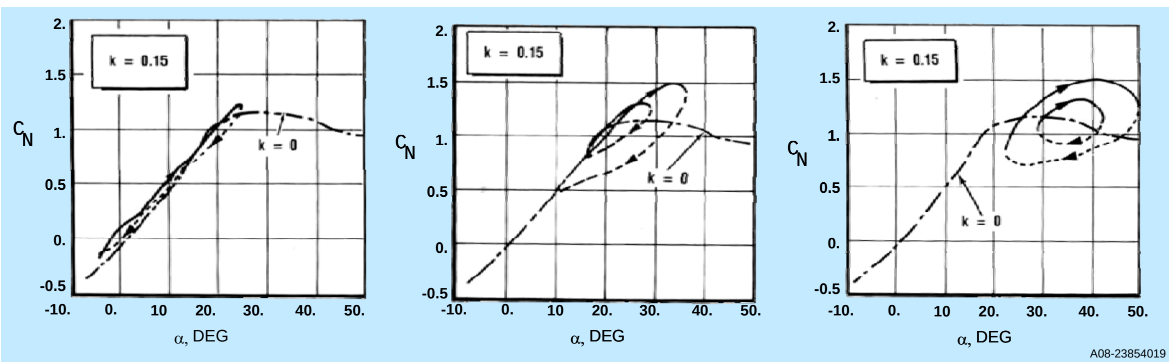

Figure 19 Effects of Amplitude and Range of Oscillation on Unsteady Normal Force for the Straked Wing Model (Ref. 30)

For high g transonic maneuvers that are more sustained, such as wind-up-turns, static aeroelastic effects can be quite non-linear due to first order modifications of mixed transonic flows over the wings. These modifications result from wing twist that is produced by static aeroelastic interactions. If the wing is a wash-out design, then nose-down twist will have a beneficial effect on wing tip shock induced separation. For a wash-in design the opposite will be true. These effects are discussed in Reference 31 where both CFD and experimental results are compared. For a wash-out wind tunnel model wing, the wing tip flows remained attached at M = 0.9 and 8-9 deg AOA. CFD/FEM solutions with an Euler method provided good agreement with the wash-out model data. A solution shown in Reference 31 for an F-16 pulling a 9-g transonic maneuver is illustrated in Figure 20 where the wing deflections are quite visible and indicative of actual aircraft response. The wash-in wind tunnel model at the same conditions was separated at the wing tip for which the Euler based CFD/FEM solution was inadequate due to inability to treat separated flows.

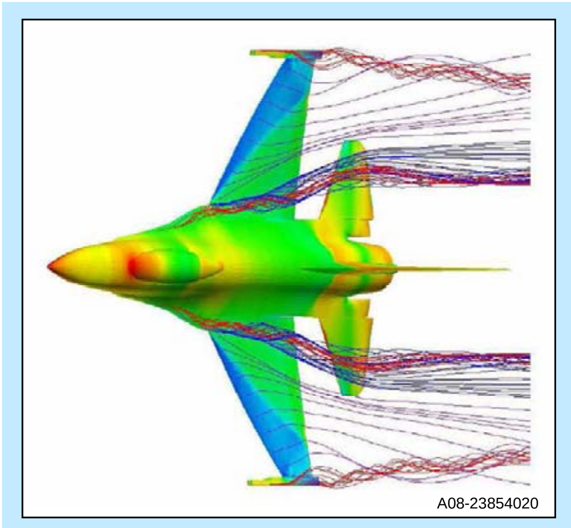

Figure 20 Pressures and Streamlines Obtained from a Computational Aeroelastic Maneuver Simulation (Ref. 31)

Wing tip flow separation at transonic speeds plays a key role in the problem area of abrupt wing stall where one wing stalls earlier than the other and results in an abrupt un-commanded roll of the aircraft. The abrupt wing stall (AWS) program was initiated by such problems encountered with the pre-production version of the F/A-18E (Ref. 32). Cooperative efforts by many contributors were conducted and reported in two parts in the AIAA Journal of Aircraft, May-June 2004 and May-June 2005. An historical review for AWS was presented in Reference 33 where it was concluded that the phenomenon is not new. Because wing tip shock induced flow separation is involved in AWS, static aeroelastic and ASE effects as well as maneuver dynamics may all be contributors to the phenomenon as discussed in the previous paragraph relative to Reference 31. Thus, time domain aeroelastic solutions that include ASE effects would be very beneficial in identifying and understanding AWS on both new and existing aircraft. It would also provide guidance for flight testing as well as reduce the scope of a diagnostic flight test program. More importantly, fixes could be evolved analytically and critical conditions could be identified for flight verification in a manner similar to what is now done for flutter.

5.0 IMPACTS ON AIRCRAFT DESIGN AND DEVELOPMENT

The impacts on aircraft design and development by the availability of time domain aeroelastic solutions is multi-faceted. These impacts include, but are not limited to: early detection of potential ASE and transient problems; improved aircraft maneuver capability; better identification of maximum design loads; improved buffet and fatigue assessments; and flight test support. Currently, time domain aeroelastic solutions are applied to LCO and buffet as well as to transonic flutter for limited demonstration cases. Linear and semi-empirical aerodynamic based time domain solutions are more feasible in the near term. Non-linear CFD based solutions are not yet practical for design purposes due to the long turn-around times; however, with the rapid development of computing power, such methods will become more attractive and practical within the next 5 to 10 years.

In this section, several examples will be described that illustrate the relevancy as well as the potential benefits of employing time domain aeroelastic solutions to aircraft design and development. Included will be a survey of representative state-of-the-art methods as well as specific applications to various problems associated with the F-16 and F-22 aircraft.

5.1 Examples of Current State-of-the-Art Methods

Much progress has been made over the past 30+ years with regard to conducting aeroelastic analyses in the time domain. References 24, 27, 28, 31 and 34-38 are representative of current on-going efforts to conduct aeroelastic analyses in the time domain. These examples are not meant to be a detailed historical perspective as given in Reference 27, but to provide a brief survey of the latest developments and trends.

Calculation of stability derivatives for the preproduction F/A-18 E aircraft was accomplished with CFD for the rigid aircraft and compared very well with wind tunnel data (Ref. 34). Although these results were obtained for a rigid aircraft, the calculation of lateral directional derivatives is not a trivial matter. Static aeroelastic effects for real structures at transonic conditions calculated by a coupled CFD/FEM approach, were presented in Reference 31 which demonstrated that such an approach is practical and should lead to a standard procedure for aircraft design and development. An evaluation of application of CFD to maneuvering type motions was discussed in Reference 35 where realistic non-linear high AOA aerodynamics were involved for a free-to-roll delta wing responding to scheduled elevon control inputs. The long term goal of this research is to achieve time domain ASE simulations for high-performance fighter aircraft.

Application of a CFD/FEM coupled method was discussed in Reference 36 where a loosely coupled approach was taken for calculating transonic flutter characteristics for the AGARD 445.6 wing. Agreement with measured flutter data was very good. Wing-store LCO calculations in the time domain were performed on an ideal rectangular wing with a tip store with several CFD methods coupled with the FEM (Ref. 27). The CFD methods include CAPTSDv and ENS3DAE with both viscous and inviscid effects.

Examples of methods to obtain less resource intensive non-linear aeroelastic solutions vary from semi-empirical methods to reduced order methods. Semi-empirical methods are quite fast but limited in applicability and they require input data bases that contain the non-linear aerodynamic and structural characteristics necessary for solution. The method of Reference 24 and 28 is a time domain method that is applicable to both LCO type problems and buffet for wing like surfaces. Reference 37 demonstrates a numerical continuation approach that can rapidly identify the bifurcation behavior of non-linear aerodynamic systems. Examples included both a non-linear aerodynamic driven case and a free-play generated instability. Extensive efforts are being devoted to the reduced order model (ROM) as typified by Reference 38 as a means of eliminating extensive point specific unsteady CFD solutions through the extraction of "aerodynamic modes". It is believed that this approach is limited to conditions in which flow states do not undergo major transitions such as shock induced separation, vortex burst and leading edge separation.

An interesting paper on dynamically modeling the human pilot's arm (Ref. 39) suggests the possibility of including pilot dynamics into the time domain aeroelasticity. It is expected that a large number of papers and reports exist as seen in the reference list for that paper. The addition of human pilot modeling would certainly require support from specialists in that area. One problem area discussed in Section 3 of the current paper concerning modeling of wake vortex encounters would be an obvious candidate for such added capabilities.

The following subsections describe the use of the semi-empirical method of References 24 and 28 to solve or better understand various non-linear aeroelastic problems for the F-16 and F-22.

5.2 F-16 Ventral Fin Re-Design

The severe fatigue problems resulting from LANTIRN pods' carriage for the F-16 ventral fins were not adequately solved during the late 1980's and early 1990's as discussed in subsection 3.2. Constant high levels of buffeting response due to the pods as illustrated in Figure 9, were not effectively countered with increased strength and stiffness of the ventral fins and their supporting structures. Therefore an investigation was conducted in the mid-1990's to redesign the ventrals and supporting structure where both structural and aerodynamic modifications were evaluated. Four fin configurations were evolved on the basis of wind tunnel and water tunnel testing, and application of a standalone version of the closed loop time domain buffet analysis tool (which was later embedded into the NONLINAE code, Reference 28). Because leading edge separation from the sharp leading edge of the fin occurred at about 3 deg side-slip, standard analysis runs of the fin designs included side-slip variations up to 5 deg. Details of this design development are given in Reference 25.

The four fin configurations were tested at Lockheed Martin in Fort Worth on an upgraded late Block 40 test F-16 on loan from Edwards AFB (Ref. 40). Three of these four fins, plus several early block ventral fins, were tested on an early Block 15 F-16 by the RNLAF in The Netherlands. The four fins consisted of: (1) the baseline fin, "BSLN," the standard block 40 ventral fin; (2) the "MMC" fin, the Block 40 fin with 40% stiffer skins of MMC aluminum material; (3) the "MMCNC" fin, the MMC fin with an added rounded "nose cap" glove with a NACA 0012 airfoil section of 5 inch chord; and (4) the "NACA" fin, the Block 40 modified to have a full span, full chord airfoil section that eliminated the sharp leading edge and sharp tip section of the fin.

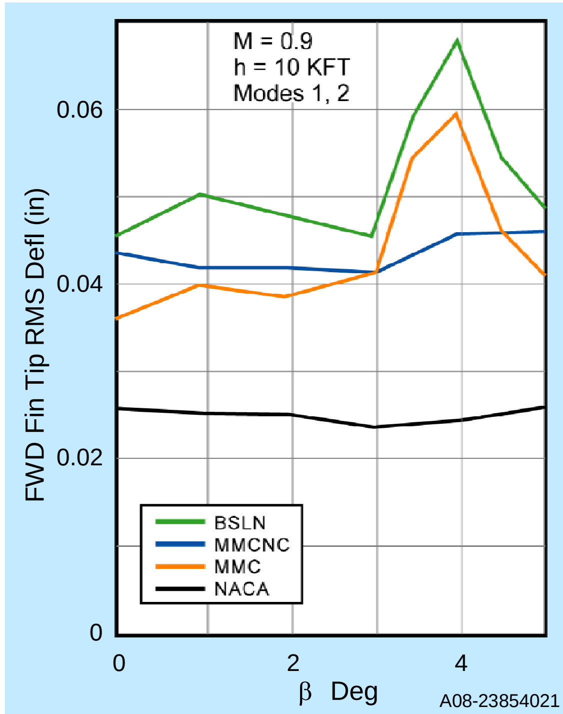

Figure 21 F-16 Ventral Fin Calculated NONLINAE Response for Four Fin Designs With LANTIRN Turbulence (Ref. 25)

The distinguishing characteristics of the four fin designs are best illustrated with the analytical buffet predictions as shown in Figure 21. This increase in buffet response in the 3 to 5 deg range was due to leading edge separation on the fins which acted as an amplifier of the turbulence flowing over the fin. Elimination of this effect was achieved with rounding of the fin leading edge as demonstrated in Figure 21 by the flat response of the "MMCNC" and "NACA" fin designs.

Flight test results for the four fin designs at Fort Worth on the Block 40 F-16 are shown in Figures 22-24 (Ref. 25). The quantity shown in these figures is a fatigue "damage parameter" rate that is generated directly from time history test data. It uses Minor's rule to accumulate incremental fatigue damage that occurs with each reversal pair of data points relative to the mean of those points (Refs. 25 and 40). The damage rate is calculated by dividing the total damage for a time history sample by the number of seconds for the sample. Since fatigue damage due to a reversal pair is represented as a half cycle raised to the sixth power, single large spikes in the response can produce as much damage as many lower amplitude cycles.

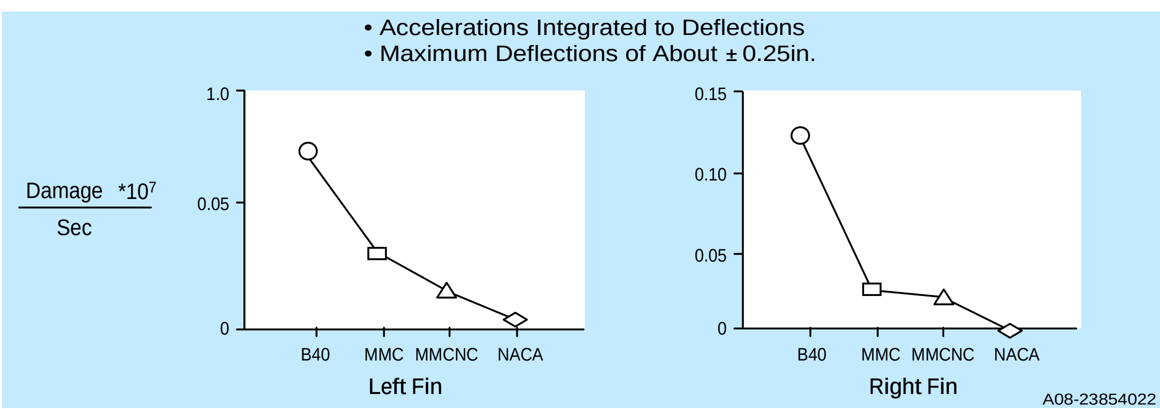

Figure 22 Accelerometer Response for Four Fin Designs to LANTIRN Pod Turbulence, M=0.90, 10K FT, Flight Test Data Normalized to 600 KCAS (Ref. 25)

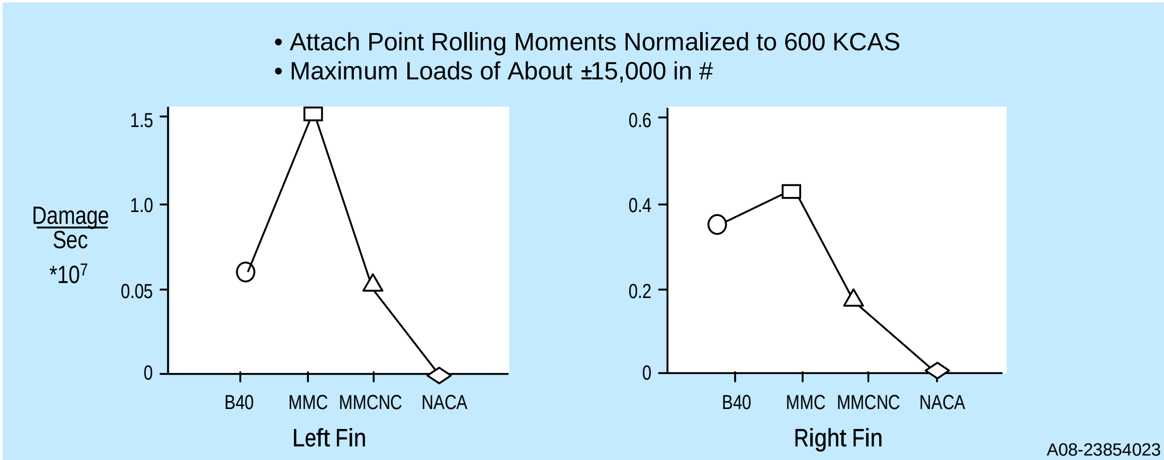

Figure 23 Rolling Moment Response for Four Fin Designs to LANTIRN Pod Turbulence, M=0.90, 10K FT, Flight Test Data Normalized to 600 KCAS (Ref. 25)

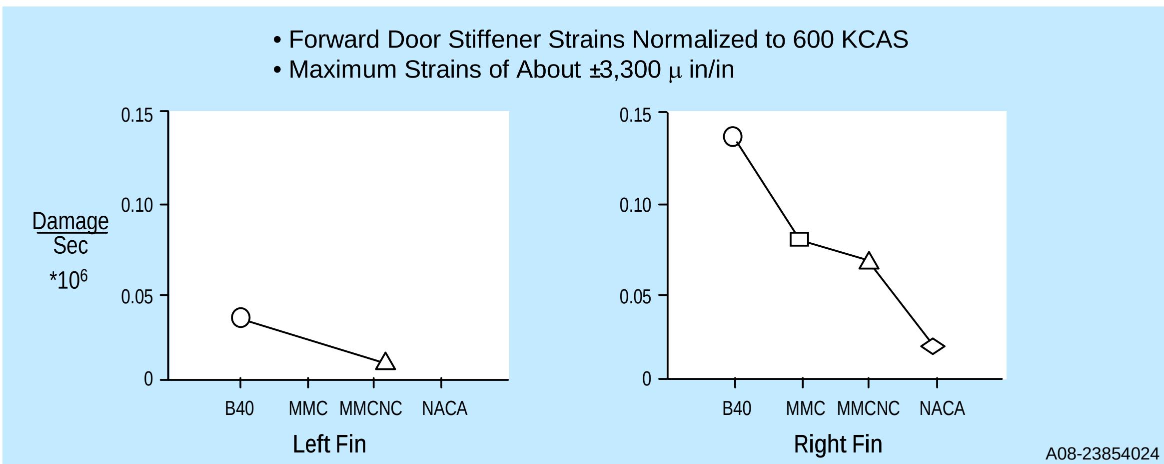

Figure 24 Stiffener Strain Response for Four Fin Designs to LANTIRN Pod Turbulence, M=0.90, 10K FT, Flight Test Data Normalized to 600 KCAS (Ref. 25)

Damage rate variations due to fin tip deflections are shown Figure 22, where the "MMCNC" rate is consistently lower than that for the "MMC." Thus, addition of the aerodynamic nose cap modification to the same fin is effective. This difference is even more striking for the fin's rolling moment response in Figure 23 where the "MMC" fins produce higher damage rates which are significantly reduced by the nose cap addition. Effectiveness of the nose cap is further demonstrated in Figure 24 for the fuselage panel forward stiffener strains.

Although the NACA fins design was far superior to the others in terms of decreasing fatigue damage rates, it required significant re-tooling. Hence, the "MMCNC" fin, which only required new skin material and a bonded nose cap, was chosen as a preferred spare for all F-16 models. With large reductions in failure rates for the "MMCNC" fins, this design is now the only spare fin that is provided for all F-16's world wide.

5.3 F-16 Ventral Fin Modeled Response To Throttle Chops

During the late 1970's and 1980's, the primary source of ventral fin buffet was high speed throttle chops where the throttle was quickly returned to idle thrust at transonic speeds. This action resulted in dumping high pressure air in the engine duct out of the inlet face. Following the initial dumping, inlet spillage developed which persisted until either the speed was low enough to match thrust or the throttle was increased sufficiently to match the speed. As shown in Figure 9, the buffet response due to high speed throttle chops is quite severe and is comparable to that due to the LANTIRN pods.

Modeling of the calculated response of the ventrals to LANTIRN turbulence was accomplished with a constant excitation as was representative of the usage (Ref. 25). Since throttle chops are highly transient, the fatigue damage rates would have to be changed to an event damage value where division by time of the event was not used. Analysis of flight test data for throttle chops in Reference 40 was conducted in such a way as to be consistent for each fin design based on standard deceleration profiles for the test points.

An example of a means for modeling these types of transients is shown in Figure 25 where the turbulence model for the LANTIRN pods was quickly ramped up and ramped down over a period of time representative of aircraft deceleration. A qualitative comparison of the response envelopes for this approach with flight test data illustrates that transient buffet analysis is possible. If this methodology was available and applied during early design stages of the F-16, then the ventrals would have been designed to better fit the environment.

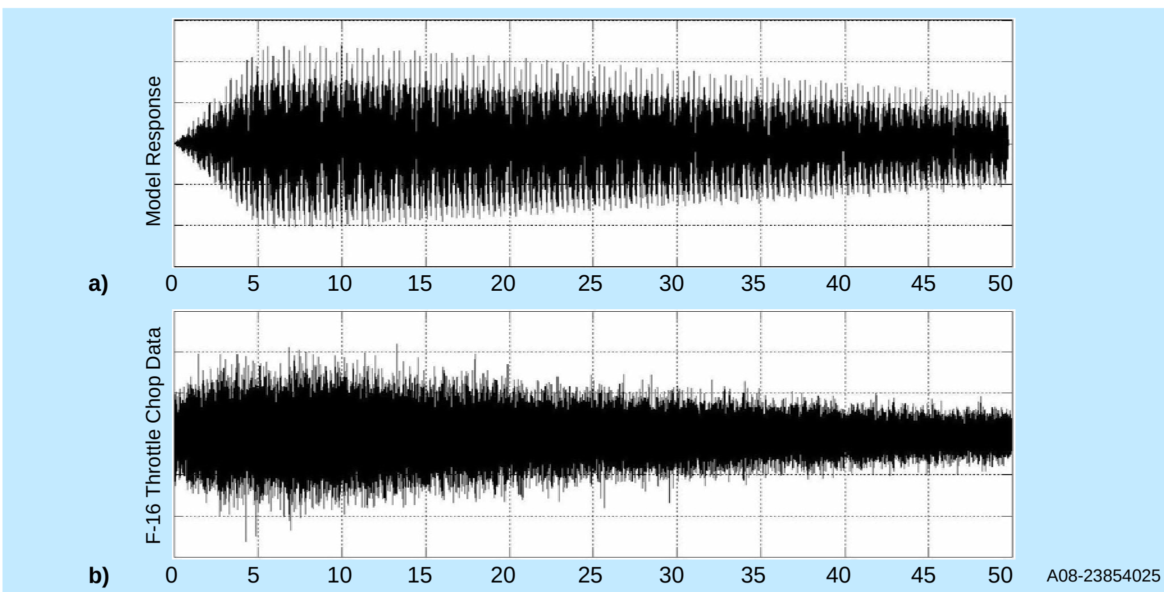

Figure 25 Transient Buffet Modeling With NONLINAE Code, (a) Simulated Transient Response; and (b) F-16 Response to High Speed Throttle Chop

5.4 F-16 Air-to-Ground Configuration 4.4 Hz LCO Modeling

Reference 24 presented some examples of predictions of LCO for an F-16 air-to-air configuration and for a B-1A wind tunnel model. Both problems involved non-linear aerodynamic forces where the first was transonic shock induced trailing edge separation and the second was unstable vortex flows on highly swept wings. The semi-empirical non-linear aeroelastic method, NONLINAE (Reference 28) was used to make these predictions which compared quite well with test data for both cases.

In addition to non-linear aerodynamic forces, the LCO case for an F-16 air-to-ground configuration with wing-tip missiles required other non-linear forces to achieve what is referred to as the "4.4 Hz LCO" for one type of missile and the "5.0 Hz LCO" for another type of missile (see Figures 5 and 6). Through numerical experimentation aided by physical insight into the phenomena, the additional force was identified as static Coulomb friction associated with a mode that consisted of almost pure pitching motion of the tip missile on the wing. The additional generalized force was applied to this mode at points in the time history solution process where the tip store velocity went to zero.

Results for two calculations are shown in Figure 26 to demonstrate the effect of applying the Coulomb friction force using the NONLINAE code of Reference 28. The top time history was obtained with this new force with initial 4.4 Hz excitation to represent flight flutter practices. The bottom time history was identical but did not have the new force included. Response during excitation is clearly shown for 5 sec, but damps out after termination of the excitation.

This application of a non-linear frictional force in the time domain clearly demonstrates the value of such an approach. The main advantage is that discontinuous physical mechanisms are easily implemented without resorting to continuous mathematical approximations. Free-play is also more physically modeled in the time domain and is thus more directly evaluated in parametric studies.

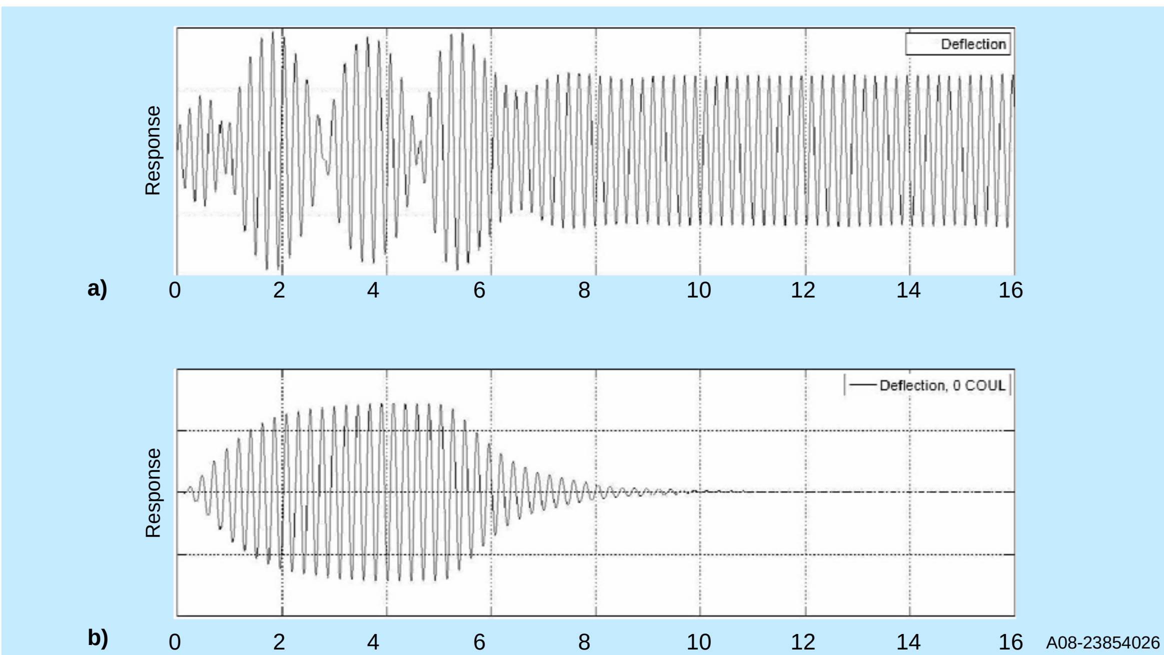

Figure 26 F-16 4.4 Hz LCO Modeling With NONLINAE for Coulomb Friction Model (a) On; and (b) Off

5.5 F-22 Vertical Tail Buffeting

In Reference 25, results were presented for predictions of F-22 vertical tail buffeting that were obtained with the NONLINAE code where it was shown that the predictions agreed well with measured flight data. Also, time domain and frequency domain solutions using the same unsteady pressure data produced nearly identical response PSD's. The unsteady RMS pressure levels were generated with time accurate CFD and the spectral distribution shape was estimated from measured wind tunnel pressure data.

Statistical modeling of the F-22 vertical tail flight buffet data base was presented in Reference 41. Early flight testing at low *q*'s showed that the ratio of measured maximum peak loads relative to the RMS were very high, exceeding values of 4. For projected maximum peak loads scaled with *q*, subsequent flight buffet testing was approached with caution by a slow build-up as the exceedance value of 4 or higher indicated that unacceptable loads would be encountered within the design envelope. However, as higher *q* test points were conducted, the exceedance values reduced in a nearly linear fashion. At completion of vertical tail buffet testing, the exceedance value was reduced to the 2.0 – 2.5 range with an increase of RMS up to 3.5 times the initial values of early testing where the exceedance was in the 4.5 to 5.0 range.

Although this result seemed mysterious, an examination of the flight test time histories indicated that the bursting of high buffet response remained about constant, but the bursts were separated by periods of low response for which the length of time decreased as the RMS levels increased. An example of this characteristic is shown for a side-slip condition in Figure 27 (Ref. 41) which also illustrates the asymmetry with side-slip where the right rudder actuator is more active than the left actuator.

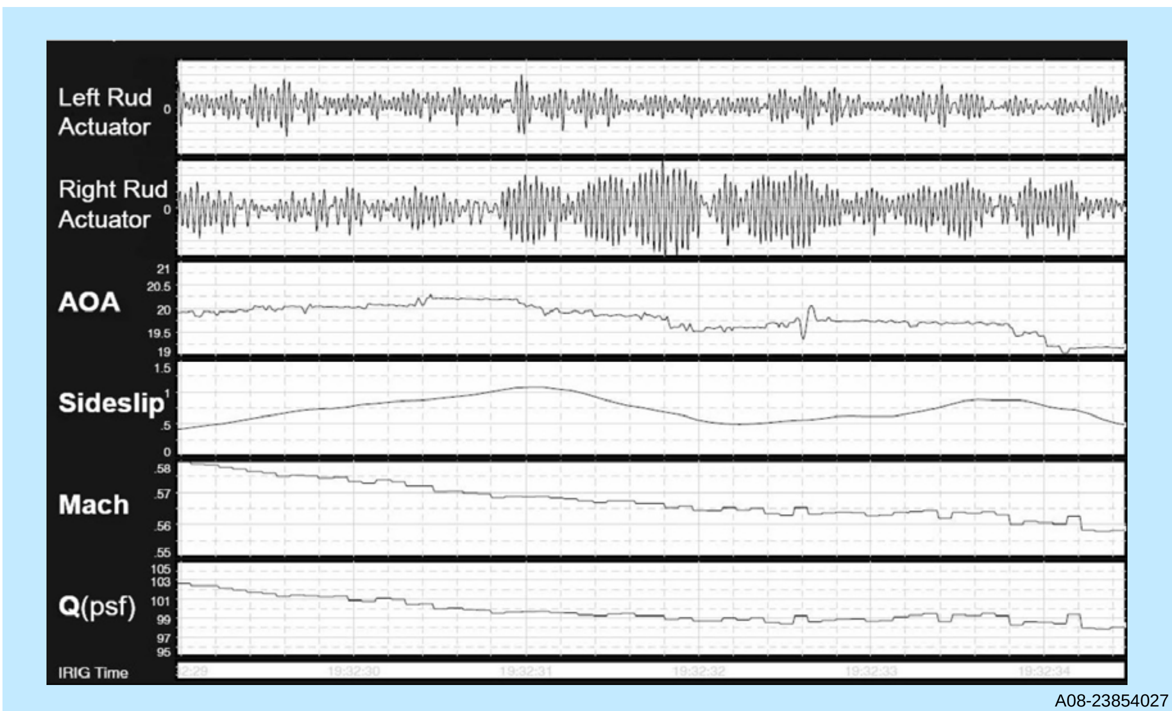

Figure 27 Flight Test F-22 Rudder Actuator Buffet Response at Nearly Constant Conditions (Ref. 41)

The buffet features embedded in the NONLINAE code (Ref. 28) have the capability to model non-stationary spectra for the buffet turbulence. This is accomplished by allowing the frequency of each spectral component to oscillate at an amplitude range equivalent to the frequency spacing of the spectrum (that is, ±0.5 Hz for a delta frequency of 1.0 Hz, etc.). In addition, a set of randomly varying phase angles is employed to provide random phasing for each of these components. The frequency of oscillation for the component frequencies is constant and is determined according to constraints that preserve overall characteristics of the spectral function (Ref. 42).

The same random phase angle set may also be employed to provide spatial phasing for each chordwise row of panels used to apply the unsteady pressures. Thus, by choosing any starting phase angle number in the set (1, 3, 10, etc.), different sequential sets of spanwise phase angles can represent different levels of spatial chaos in the calculation of the generalized unsteady aerodynamic forces.

These random features in the NONLINAE code were used to design the F-16 ventral fin configurations discussed in subsection 5.2 above. They have also been applied to the F-22 vertical tail buffeting model analysis discussed in Reference 25. Further development of the statistical aspects of the F-22 time domain solution have demonstrated the potential for realistic characteristics that track those discussed in Ref. 41.

Calculated 5 second time history responses are shown in Figure 28 for two different sets of spanwise phase angles. The top plot is for the set that produced the maximum response RMS and the lower plot is for the set that produced the minimum response RMS. The similarity between the top plot and the right rudder actuator time history in Figure 27 is remarkable. The bursting characteristics in Figure 28 are a direct result of the non-stationary spectral modeling capability in NONLINAE. The different levels in the upper and lower plots are due to the two sets of spanwise phase angles chosen.

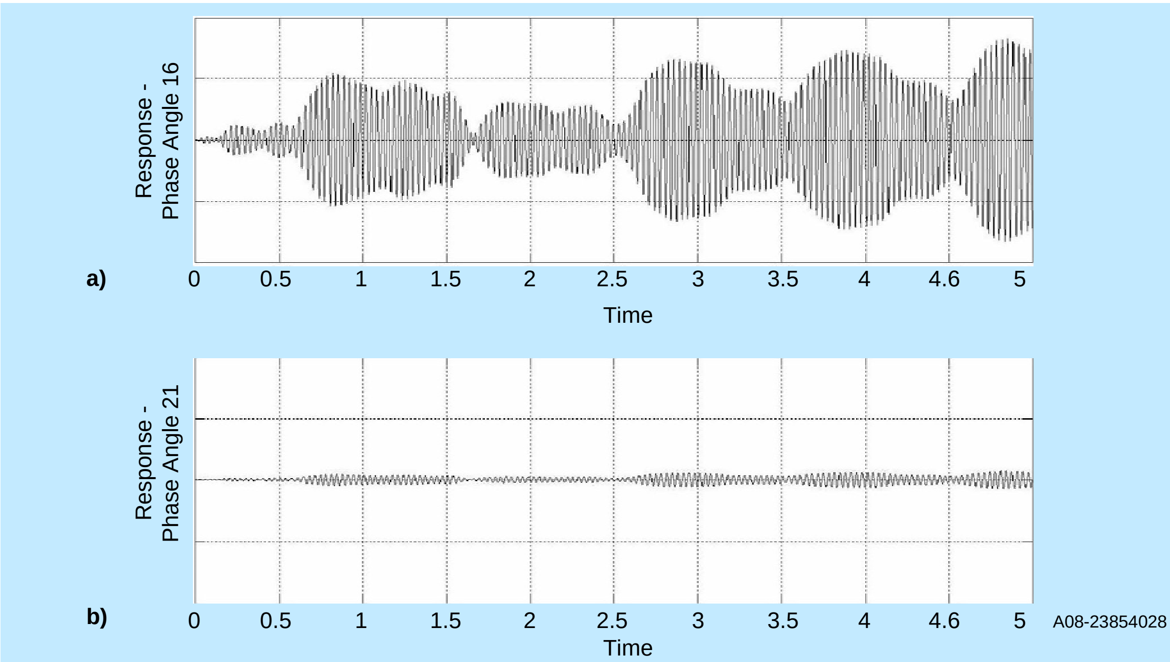

Figure 28 F-22 Rudder Buffet Response as Calculated With NONLINAE for Two Sets of Spanwise Phase Angles, (a) Maximum Response (Phase Angle Set 16); and (b) Minimum Response (Phase Angle Set 21)

In order to access the effect of spanwise chaos, 21 different sets of spanwise phase angles were used in 21 time history solutions with the same input buffet spectral data set. RMS levels for each of these spanwise phase angle set runs are shown in Figure 29. Referring to decreasing RMS at higher chaos levels, four bands of response were defined as chaos levels 4, 3, 2, 1 where each contained approximately the same number of RMS values. The distributions of the 21 values in Figure 29 also emphasize the randomness in this modeling method.

Finally, exceedance distribution plots are shown in Figure 30 where each distribution represents the combination of all time histories above a noted chaos level. As examples, all 21 time histories are used for chaos level 5 (0.0) whereas only time histories for phase angle sets 4, 6, 7, 14 and 16 are used for chaos level 1. The phase angle sets shown in Figure 28 are 16 (highest) and 21 (lowest). The tightening of the distributions are clearly seen as the level of chaos is reduced. Comparisons of these shapes with those distributions in Reference 41 show a strong similarity relative to the Rayleigh Distribution (solid line).

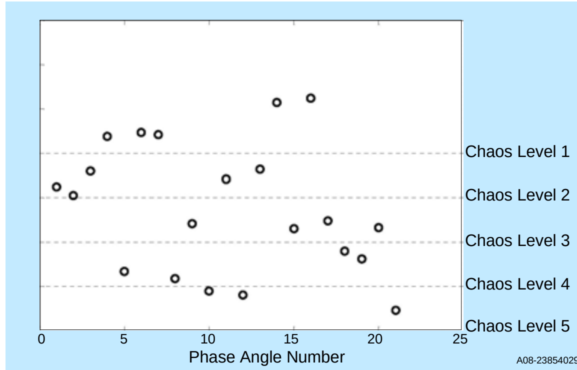

Figure 29 NONLINAE-Calculated F-22 Rudder Buffet RMS Values for All 21 Sets of Spanwise Phase Angles as Shown in Five Chaos Levels

With regard to the maximum exceedance levels shown in Figure 30, chaos level 5 yields values of +4.4 and -4.3 whereas chaos level 1 values are +2.9 and -2.9. The upper limit is +2.5 and -2.6 for phase angle set 16 for which the time history was shown in the upper plot in Figure 28. These values are all in line with those obtained during the F-22 flight buffet testing discussed in Reference 41 where the early data at low q (high chaos) were in the 4.5 to 5.0 range and the high q (low chaos) in the 2.0 and 2.5 range.

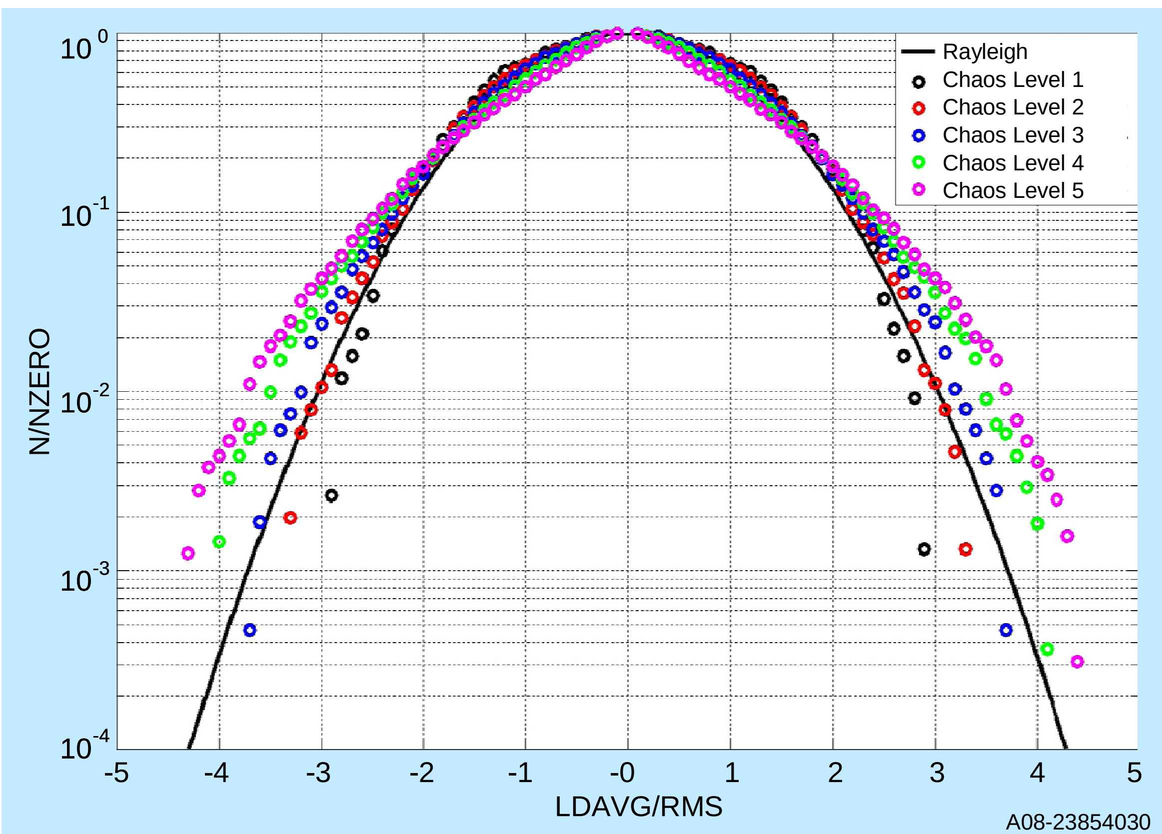

FIGURE 30 NONLINAE-CALCULATED F-22 RUDDER BUFFET RESPONSE STATISTICAL DISTRIBUTIONS FOR THE FIVE CHAOS LEVELS

6.0 CONCLUDING REMARKS

The design, development and testing of today's aircraft involves the close integration of many diverse technologies and systems which is expected to become more and more complex in the future. This is particularly true for fighters which must be light weight, highly maneuverable and multi-roll in order to meet customer requirements. In addition, these aircraft routinely operate under very transient conditions at the edges of their envelope which are highly non-linear and unsteady states, that can only be effectively modeled in the time domain. Thus, the purpose of this paper was to present a strong case for increasing the efforts to more fully develop the capabilities to perform time-domain aeroelastic simulations that can bring together the many diverse technologies which are employed in today's and future aircraft. To support this case, the paper presented four major sections covering historical perspective, problem areas, path dependency and impact on aircraft design and development.

It was shown that the projected increased complexity in the close integration of diverse technologies and systems is based on historical trends where integration complexity is driven by growth of demands for higher aircraft performance as well as both greater on-board data processing power and computational modeling capabilities for engineering designs. These trends extend all the way back in time to the beginning of aeroelasticity, where major changes in aircraft performance demands have driven major changes in aeroelastic modeling capabilities as listed below:

1. high performance mono-wing aircraft – 2-D aeroelasticity
2. supersonic flight – 3-D aeroelasticity; and
3. active controls – aeroservoelasticity.

Since the early 1980's, the appearance of LCO and other non-linear aeroelastic phenomena have motivated significant research efforts into time domain modeling capabilities. These efforts have included both semi-empirical and CFD based methods and are pointing the way to future aeroelastic developments.

Problem areas covered were high rate transient maneuvers, LCO, buffet, buzz and wake vortex encounters. Various examples of high rate transients were described which included a high-g/roll anomaly, analog to digital flight control upgrades, vibration modes/sensor interactions and store ejection loads as well as peak maneuver loads and ASE stability where flight control laws were rapidly changing. Several examples of non-linear aeroelastic problems were presented that included F-16 LCO, high amplitude buffet and rudder buzz. Wake vortex encounters were described which included personal experiences, F-16 vertical tail tip losses and an F-16 landing incident caused by the wake of a preceding aircraft.

Path dependency was described as a feature common to the above problem areas where each was manifested in the time domain and all involved some form of aeroelastic effects. The rapid change in control laws, flow separation and buffeting during highly transient maneuvers, where both static and dynamic aeroelastic effects are involved, introduce many situations of aircraft dynamics that are not modeled in conventional ASE and static aeroelastic analysis that are based on stable flight conditions. Ejection of heavy stores is an even more rapid change in aircraft dynamic states for which the ASE aspects require that the instantaneous control laws must also be included in modeling these phenomena as well as the vibration modes as they transition from pre-ejection to post-ejection configurations. Path dependency for wake vortex encounters is driven by the aircraft state prior to the encounter and the exact characteristics of the flight path/vortex trail crossing.

The impacts on aircraft design and development by the availability of time domain aeroelastic solutions are multi-faceted. These impacts include, but are not limited to: early detection of potential ASE and transient problems; improved aircraft maneuver capability; better identification of maximum design loads;

improved buffet and fatigue assessments; and flight test support. Examples were discussed where this

analysis capability provided: guidance for developing improved F-16 ventral fins; better understanding of F-16 tip missile LCO phenomena; and the role of chaos in F-22 vertical tail buffeting statistical characteristics. Currently, time domain aeroelastic solutions are applied to LCO and buffet as well as to transonic flutter for limited demonstration cases. Linear and semi-empirical aerodynamic based time domain solutions are more feasible in the near term. Non-linear CFD based solutions are not yet practical for design purposes due to the long turn-around times; however, with the rapid development of computing power, such methods will become more attractive and practical within the next 5 to 10 years.

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