# **Efficient High Resolution Modeling of Fighter Aircraft with Stores for Stability and Control Clearance**

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This paper documents recent advances towards an efficient computational method for accurately determining the static and dynamic stability and control (S&C) characteristics of high-performance aircraft and demonstrates how it may fit into the SEEK EAGLE Store Certification Process:. In contrast to the "brute force" approach to filling an entire S&C database for an aircraft, the present approach is to reduce the number of simulations required to generate a complete aerodynamic model of a particular configuration at selected flight conditions by using one or a few complex dynamic motions and nonlinear system identification (SID) techniques. The approach is demonstrated by gathering high-fidelity computational fluid dynamics (CFD) data for a rigid F-16 in prescribed motion that approximates dynamic wind-tunnel testing techniques and SID input signals. The motions are optimized to minimize the computational expense and to take full advantage of the tighter control of the CFD environment. They are specified interactively using a newly developed, GUI-based maneuver file generation tool. Global nonlinear parameter modeling and other SID techniques are then used to identify parametric models from the computed aerodynamic force and moment data. These compact models are used to predict the aerodynamic response to maneuvers that were computed for validation purposes and that were not used to derive the models. Partial derivatives of the analytical models can be used to determine the corresponding static and dynamic stability derivatives. The models can also be used to perform real time 6-DOF/aeroelastic simulations of the vehicle in conditions susceptible to spin, tumble, and lateral/longitudinal instabilities. The main benefits of this effort are: 1) early discovery of complex aerodynamic phenomena that are typically only present in dynamic flight maneuvers and therefore not discovered until flight test, and 2) rapid generation of an accurate aerodynamic model to support aircraft and weapon certification by reducing required flight test hours and complementing current stability and control testing.

# Nomenclature

$C_L$	=	lift coefficient
$C_{I}$	=	lift contribution due to pitch

 $C_{L_{a}}$ 

= lift contribution due to angle of attack (lift curve slope)  $C_{L_0}$ 

= lift contribution due to angle of attack rate

- $\overline{C}_{L_a}$ = lumped lift coefficient derivative
- $C_m$ = moment coefficient

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$C_{m_q}$	=	moment contribution due to pitch (pitch damping)
$C_N$	=	normal force coefficient
С	=	mean aerodynamic chord
$\Delta t$	=	time step
h	=	pulse amplitude
М	=	Mach number
q	=	pitch rate
$\dot{q}$	=	angular acceleration
t	=	time
α	=	angle of attack
$\dot{\alpha}$	=	rate of change of angle of attack
ä	=	rate of change of angle of attack rate

# I. Introduction

**P**RACTICALLY every fighter program since 1960 has had costly nonlinear aerodynamic or fluid-structure interaction issues discovered in flight test. The main reason for these "failures" is that the predictive methods used were not able to reveal the onset and nature of the problems early in the design phase. To keep the budget overshoot under control, fixes tend to be *ad hoc* and are applied without a sound basis of fundamental understanding of the physics concerned. Unfortunately, in future aircraft designs, the problems will only become more complex as thrust vectoring, active aeroelastic structures, and other related technologies are implemented for stability and control augmentation. Unmanned combat vehicles may operate in flight regimes where highly unsteady, nonlinear, and separated flow characteristics dominate since there are no man-rating requirements<sup>1</sup>. In order to decrease the costs and risks incurred by flight-testing and post-design-phase modifications, it would be helpful to have a tool to analyze and evaluate the stability and control (S&C) characteristics of the aircraft earlier in the development process.

Three traditional methods exist to determine stability and control characteristics. The first, and most accurate method, involves flight-testing the actual aircraft<sup>2,3,4</sup>. These tests are very expensive, time consuming, and require an operational aircraft, which may not exist in the early stages of the design process. There are also safety issues associated with flight testing new aircraft. The second method is to use wind tunnel testing of scale models<sup>5</sup>. This is also a time consuming and expensive process. Additionally, there are blockage, scaling, and Reynolds-number effects together with support interference issues that prevent the proper modeling of the full-scale vehicle behavior. Also, changes to the actual aircraft geometry may invalidate wind tunnel test data. The final method employs a combination of data sheets, linear aerodynamic theory, and empirical relations<sup>6,7</sup>. This method has met with great success due to its simplicity, but its accuracy is limited – while the basic lift and drag characteristics of high performance aircraft such as the F-16 and F-18 may be predicted fairly well at benign flight conditions, it is very difficult if not impossible to accurately capture the unsteady and dynamic aerodynamic effects of maneuvering aircraft with these techniques.

A relatively new tool in this quest is Computational Fluid Dynamics (CFD). Navier-Stokes CFD solvers have reached a level of robustness and maturity to support routine, everyday use on relatively inexpensive computer clusters. The computation of static stability derivatives can be done with present off-the-shelf CFD tools. However, the prediction of dynamic derivatives requires a time-dependent prescribed motion capability in the flow solver as well as prescribed motions ("maneuvers") which adequately excite the desired aerodynamics. Such high-fidelity CFD offers several unique capabilities that complement experimental testing techniques for obtaining these aerodynamic parameters, but without their limitations. The physical limitations and kinematic restrictions of wind tunnel testing including model motion as well as the interference effects of the model support are not factors in the computational analysis. Flight tests are limited by the fact that only "flyable" maneuvers are possible where many of the parameters in common aerodynamic models are lumped together with no way to determine the independent effects. With CFD, it is possible to prescribe any type of aircraft motion in a flow field. CFD is also capable of altering the flow physics associated with a particular testing technique from those observed in a wind or water tunnel test in order to isolate effects<sup>8</sup>. In addition, CFD can increase our understanding of the causes and types of separated flows affecting S&C prediction. CFD has its own limitations, of course, such as turbulence and transition modeling, to name a few. These limitations have been the topic of countless papers and will not be reiterated here. For the purpose of this work, we focus on modeling and predicting S&C characteristics with CFD and assume that the CFD code and its turbulence models are validated.

The present paper seeks to demonstrate how this new method of computing stability and control characteristics of an aircraft with stores can fit in the current aircraft/store certification process called SEEK EAGLE. First, the SEEK EAGLE Office is described, next the Stability and Control Group is described, and then two case studies of past clearances are presented to lay the foundation of how this method may integrate into the current AF SEEK EAGLE Process. Finally, results are presented of an example use of the method and some validation and verification data are presented to show the work is relevant.

# Air Force SEEK EAGLE Office (AFSEO) Overview

At the end of the Vietnam Conflict, virtually every store in existence was certified for use on the F-4 under the SEEK EAGLE program. In the late 1970s, the F-16 exhibited many more weapon incompatibilities than the F-4. As a result, F-16 combat users were not getting the combat capability they expected. Problems with the SEEK EAGLE program culminated in 1986 when HQ TAC/CC challenged HQ USAF to fix the problem. HQ USAF directed the SEEK EAGLE revitalization study which resulted in the AFSEO charter by SECAF in December 1987.

AFSEO's Mission is to ensure new warfighter capabilities through the application and transfer of aircraftstore compatibility expertise. Its vision is the on time, cost-effective delivery of all USAF flight vehicle store compatibility knowledge.

The SEEK EAGLE program is the standard for the aircraft-stores certification process governed by AFI 63-104 for the US Air Force. The SEEK EAGLE program supports aircraft-stores certification of US and foreignorigin weapons and stores. It assures aircraft-store capability, weapon delivery accuracy verification and obtains data needed to verify and update the accuracy of Operational Flight Programs (OFP) and Technical Orders (TO).

AFSEO delivers a variety of products to the warfighter. The Quick Reaction Certification (QRC) is an urgent or mission essential certification, intended to support actual contingency or warfighting requirements. QRCs take precedence over all previously submitted requirements. The Certification Recommendation (CR) is a non-urgent, world-wide aircraft-store certification product that is executed according to the SEEK EAGLE Priority List, which is largely influenced by the needs of Air Combat Command (ACC). The Flight Clearance (FC) is for specific configurations in support of flight test activities or configurations that have limited operational use. FCs are valid for a specified duration only. AFSEO also provides Ballistics and Safe Escape Analyses, Technical Consulting and Expert Support, Computerized Physical Fit (CPF) modeling of stores on multiple aircraft platforms, Computational Fluid Dynamics (CFD) and Mass Property measurements of stores.



The Certification Process is detailed below in Figure 1:

# **Figure 1: SEEK EAGLE Process**

An FC or CR may be required for a variety of reasons, including but not limited to changes in store configuration, external aerodynamic shape, electromagnetic radiation environment, safing or arming design, ballistics or propulsion, a 5%+ change in store weight or a 10%+ change in pitch or yaw moments. During the course of the certification process, SEEK EAGLE management must identify where stores are to be located on the

aircraft, store configuration, any options requested as well as prioritizing configurations for large-scale efforts. The respective engineering disciplines must identify max G limits, Mach number, Airspeed and Altitude, as well as determine the limits at which store(s) will be released and the limits required for emergency jettison. Final recommendations are provided to the System Program Office (SPO), who is the ultimate authority for final certification of aircraft/store compatibility. This information is then used to generate and update Technical Orders.

# **AFSEO Stability & Control Group**

The SEEK EAGLE Stability and Control group determines the impact external stores have on aircraft flying qualities and ensures all configurations meet weight and balance requirements. Most stores are cleared or certified by analogy to previously tested stores with similar aerodynamics and mass properties. The majority of small changes can be analyzed this way. Wind tunnel testing may be performed for stores requiring certification that cannot be cleared by analogy. However, the funding, time and priority required to obtain accurate and timely answers is often prohibitive to meeting rapidly changing warfighter needs. Flight testing is the most expensive, but most robust option available to the AFSEO S&C group. Fully-instrumented monitored tests are available as well as un-instrumented, qualitative, Captive Flight Profile (CFP) tests that rely on pilot assessment in the form of Cooper-Harper and PIO rating systems.

Resources currently available to the S&C group include an extensive library of flight test and contractor analysis reports for 'by analogy' clearance as well as readily available prime contractor support. Tools available include fuel-burn and drag index calculation programs, the Automated Form F weight & balance software, AOA data tables and IADS analysis software for real-time flight test monitoring. A variety of MATLAB/Simulink based tools are currently under development as well.

The primary tool available to the AFSEO S&C group to evaluate the S&C characteristics of the F-16 is Lockheed Martin's Aircraft Trim, Linearization and Simulation (ATLAS) program. The ATLAS program is a generalized, 6-DOF, nonlinear, non real-time simulation. Using ATLAS, the S&C engineer is able to trim the aircraft at a selected flight condition, calculate linear aerodynamic derivatives and simulate time history response from a trimmed condition for a variety of maneuvers. These maneuvers include doublets, rolls, max command pitch maneuvers and pitch rocking during deep stall. In addition to performing the aforementioned trims and maneuver simulations, force and moment coefficient data can be extracted for further analysis.

The ATLAS program contains ten generic subsystems such as equations of motion, air data, pilot model, gust model and mass properties. Six subsystems are F-16 specific which include aerodynamics, propulsion, flight control hardware, flight control laws, control surface actuators and landing gear. Simulations can be run for the digital Block 40-52 aircraft as well as the analog Block 15-32 aircraft. The aerodynamic database is the same data utilized on the F-16 handling qualities simulator and is comprised of data from wind tunnel tests and flight tests. The baseline, clean aircraft aerodynamic package includes breakpoints at nine Mach numbers and six altitudes, as well as breakpoints within +/- 180 degrees AOA and +/- 90 degrees of sideslip. Incremental aerodynamic data for a select number of store configurations is also available for analysis and simulation.

The S&C group's primary limitation is its inability to accurately predict where instabilities occur in the flight envelope, if they occur at all, for new store configurations as well as the type and degree of instability they may cause. The aerodynamic data available in ATLAS does not include more recent weapons, pylons, multicarriage racks and pods. The database is also confined to symmetric configurations when the majority of current store certification work involves asymmetric configurations. The group is also unable to accurately predict departure resistance and boundary layer separation of new store configurations requiring high AOA testing.

To overcome these limitations, AFSEO S&C needs an efficient method of predicting instabilities to continue delivering maximum capability to the warfighter. Configurations that employ a variety of stores and rack systems on a single aircraft to engage multiple targets are in high demand to prosecute the War on Terror. Budgetary pressures and reduced flight test availability are creating an environment that requires doing more with less to meet warfighter needs. Furthermore, new aircraft are entering the USAF fleet such as the F-22 and F-35 that will eventually require substantial aircraft-store compatibility analysis. Developing new modeling and simulation capabilities using the extensive F-16 knowledge base as a foundation is critical to supporting the warfighter who will employ the platforms and weapons of the future.

# Case Study I: Pylon Integrated Dispenser Station (PIDS Pylon)\*\*,<sup>††</sup>

<sup>\*\*</sup> AFDTC-TR-96-33, "Pylon Integrated Dispenser Station (PIDS-3) F-16 Certification Testing," Nov 96.

Manufactured by the Per Udsen Company of Denmark, the Pylon Integrated Dispenser Station (PIDS pylon) was a modified 16S1700 standard weapons pylon designed to improve the survivability of Air National Guard (ANG) F-16s by augmenting chaff and flare payload without negatively impacting carriage of conventional weapons and electronic countermeasures. In 1992, the SEEK EAGLE office was directed to certify carriage of the PIDS pylon on stations 3 and 7 on pre-Block 40 F-16 aircraft. To do so, the S&C group required testing of Loading Category (CAT) III air-to-ground configurations as well as CAT I air-to-air configurations. The two flight test programs outlined below summarize efforts to certify the PIDS pylon on the F-16 without targeting pods (TGP) on pre-Block 40 aircraft.

# CAT III Air-to-Ground Flight Testing

Air-to-ground CAT III flight testing of the PIDS pylon was conducted by the 46<sup>th</sup> Test Wing, 39<sup>th</sup> Flight Test Squadron, Eglin AFB FL at the request of the SEEK EAGLE office. The purpose of the test was to evaluate the effect the PIDS pylon had on aircraft handling qualities. Six qualitative Captive Flight Profile (CFP) missions were flown in support of this effort. Test pilot assessment of aircraft S&C/HQ were critical in comparing the PIDS pylon with the standard weapons pylon.

Historically, left wing heavy configurations are more critical on the F-16 from an S&C/HQ perspective. Given that the PIDS pylon is heavier, longer and wider relative to the standard weapons pylon, the PIDS pylon was mounted on station 3 only. Three PIDS configured loadings and three identical configurations with standard weapons pylons on stations 3 and 7 were flown as baseline configurations as shown below in Figure 2. Each baseline configuration was flown prior to its respective PIDS-equipped configuration. This allowed the pilot to evaluate the effect the PIDS pylon had on aircraft handling qualities when compared to the identical configuration with standard weapons pylons.

Looking Forward					Station Loading				
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Loading	1	2	3	4	5	6	7	8	9
L1 Baseline	Q	Ø	Ö	CLEAN	300	CLEAN	Ö	₿	Q
L1	Ø	Q	Ð	CLEAN	300	CLEAN	0	B	Ò
L2 Baseline	Ø	CLEAN	°. ₽	370	300	370	∎¢°	CLEAN	Q
L2	Ø	CLEAN	°₽°	370	300	370	°. ↓	CLEAN	Q
L3 Baseline			П (84)	CLEAN	300	CLEAN			Ø
L3	Ø		<b>E</b> (84)	CLEAN	300	CLEAN			Ø

<sup>††</sup> AFFTC-TLR-94-60, "F-16 Pylon Integrated Dispenser Station (PIDS/3) Stability and Control Evaluation," Dec 94.



Figure 2: Case Study 1 F-16 PIDS CAT III Flight Test Configurations with a legend.

Mission summaries were generated in accordance with MIL-HDBK-1763A (Test 251) and MIL-HDBK-244A [Paragraphs 6.2.1.7.6.2(a), (b), and (d)]. Maneuvers in the Power Approach (PA) configuration were performed first at approximately 200 KCAS and 15,000 feet. Trim shots, steady heading sideslips, pitch doublets, yaw doublets, roll doublets, loaded rolls, pushovers, unloaded rolls and wind up turns were then performed from 0.8 to 1.2 Mach at 5,000 to 30,000 feet. Each configuration also included an evaluation of air-to-ground handling qualities during tracking (HQDT). This enabled the pilot to assess the PIDS pylon's effect on his ability to accurately employ air-to-ground weapons.

All six CFP missions were flown successfully with no significant anomalies. Test pilots determined that the PIDS pylon had little noticeable effect on F-16 handling qualities. Furthermore, handling qualities during approach and landing were also found to be identical to the respective baseline configuration. The positive results of these six missions enabled SEEK EAGLE to conclude that F-16 S&C/HQ for CAT III air-to-ground loadings with the PIDS and standard weapons pylons were analogous.

# CAT I Air-to-Air Flight Testing (without Targeting Pods)

In 1994, high- $\alpha$  CAT I air-to-air flight testing of PIDS pylon Revert-To loadings without Targeting Pods (TGP) was conducted by the 416<sup>th</sup> Flight Test Squadron at the Air Force Flight Test Center, Edwards AFB CA, at the request of the SEEK EAGLE office.

The objectives of the test were to:

- 1) Determine the change in departure resistance and stability and control characteristics, as well as maneuvering limits, of selected PIDS Revert-To loadings.
- 2) Determine the change in post-departure and recovery characteristics of the same PIDS Revert-To loadings.

Revert To loadings included all external equipment except air-to-ground stores. The test aircraft was a Block 30 F-16 equipped with a flight test air data boom, a flight test instrumentation system and a spin recovery system.

The PIDS Revert-To loadings evaluated included a PIDS pylon on station 3 and a standard weapons pylon on station 7 due to aforementioned F-16 left-wing asymmetries being worst-case from an S&C/HQ perspective. Baseline configurations with standard weapons pylons on stations 3 and 7 were also flown. All baseline configurations had been previously certified to CAT I.

To meet the ANG's requirements, two families of configurations were tested: one with a 300-gallon centerline tank and one with 370-gallon wing tanks and centerline ECM pod. TER-9A loadings were flown since previous testing had demonstrated that the TER-9As were more critical than empty pylons or launchers and to minimize flight test costs. Provided the TER-9A loadings passed CAT I limits, test results would then clear other less critical station 3 and 7 loadings required by the ANG such as empty launchers and empty PIDS pylons. The four PIDS and baseline configurations tested are shown below.



# Figure 3: Case Study 2 F-16 PIDS CAT I Flight Test Configurations with a legend.

Flight testing of the PIDS pylon was broken into two phases. Phase I departure boundary testing consisted of 1g maximum command 360 degree rolls at AOAs of 20 degrees, 22 degrees, and at the AOA limiter; abrupt input slowdown turns to the left and right at 0.6 and 0.9 Mach, and maximum-g maximum command 360 degree rolls at 200, 250 and 300 KCAS. All maneuvers were performed with the throttle at military (MIL) power and at the aft CG limits for centerline tank and wing tank loadings.

Phase II departure boundary testing consisted of an evaluation of post departure and recovery characteristics from intentional departure maneuvers. These maneuvers included upright, inverted and roll-coupled departures initiated at pitch attitudes of 60 degrees and 75 degrees. All intentional departures were performed with the throttle in IDLE. Longitudinal stick force was used to maintain the desired pitch attitude until the aircraft departed controlled flight. By definition, departures occurred when the noseboom AOA exceed +/- 40 degrees. All intentional departures were initiated above 35,000 feet MSL.

In summary, a total of nine test sorties were completed. Wing tank and centerline tank loadings with the PIDS pylon were evaluated and exhibited acceptable post-departure and recovery characteristics. The F-16's departure resistance with PIDS pylon and centerline tank or wing tanks was found to be comparable to equivalent certified loadings with the standard weapons pylon. These test results proved that S&C/HQ for CAT I loadings with the PIDS or standard weapons pylon without TGP were analogous.

Case Study II: MA-31

The MA-31 Target Vehicle is a converted Russian Kh-31 supersonic anti-ship missile capable of sea-skimming at Mach 2.7 and performing 15G multi-axis maneuvers. The MA-31 has historically been launched from the F-4 and QF-4. In recent years, work has been performed to integrate the MA-31 on stations 4 and 6 of the F-16 to allow greater flight test flexibility and to supplement the QF-4. In 2006, the SEEK EAGLE office was directed to complete aircraft-store compatibility work required to clear the MA-31 on the F-16.

Due to the lack of available wind tunnel analysis of the MA-31 and its launcher assembly on the F-16 as well as no similar, previously cleared store-launcher assembly available for comparison, air-to-ground CAT III flight testing of the MA-31 was required. Four qualitative Captive Flight Profile (CFP) missions were conducted by the 46<sup>th</sup> Test Wing, 40<sup>th</sup> Flight Test Squadron, Eglin AFB FL at the request of the SEEK EAGLE office. The purpose of the tests was to evaluate the effect the MA-31 had on aircraft handling qualities.

Mission summaries were generated in accordance with MIL-HDBK-1763A (Test 251) and MIL-HDBK-244A [Paragraphs 6.2.1.7.6.2(a), (b), and (d)]. Maneuvers in the Power Approach configuration were performed first at 10,000 feet. Trim shots, pitch doublets, yaw doublets, roll doublets, balanced symmetric pushovers, wind-up turns and bank-to-bank rolls were then performed from 0.8 to 0.95 Mach at 5,000 to 20,000 feet.

The four missions consisted of a symmetric and asymmetric MA-31 loading as well as a similar symmetric and asymmetric baseline configuration. This allowed the pilot to evaluate the effect the MA-31 had on aircraft handling qualities when compared to a similar loading. The symmetric MA-31 loading consisted of an MA-31 on stations 4 and 6, a centerline 300-gallon tank and AIM-9 missiles on stations 1 and 9. The left wing heavy asymmetric MA-31 loading was equipped with an MA-31 on station 4, a centerline 300-gallon tank, and AIM-9 missiles on stations 1 and 9. A symmetric loading of 370-gallon fuel tanks, centerline 300-gallon tank and tip AIM-9 missiles was flown as a baseline for the symmetric MA-31 loading. An asymmetric loading of a single MK-83 on station 3, 370-gallon wing tanks and centerline 300-gallon tank was flown as a baseline for the asymmetric MA-31 loading. The asymmetric moment generated by a single MK-83 was within 1,000 lb-ft. of that generated by an asymmetric MA-31. The four configurations are illustrated below.







Figure 4: Case Study 2 F-16 MA-31 Flight Test Configuration with a legend.

All four CFP missions were flown successfully with no significant anomalies. Test pilots determined that the there were no noticeable handling quality/stability differences between the baseline loadings and the respective MA-31 loadings. All configurations demonstrated solid F-16 responses in the PA and cruise configuration. Handling qualities on the tanker were nominal. The only noticeable difference between the baseline and MA-31 loadings was the increased thrust required during the MA-31 missions to achieve test points due to the high drag of the MA-31 and launcher assembly. The positive results of these four missions enabled SEEK EAGLE to conclude that F-16 S&C/HQ for CAT III air-to-ground loadings with the MA-31 were analogous to carriage of 370-gallon wing tanks for subsonic flight.

#### **Case Study Summary**

In the two case studies presented, a total of 19 sorties were flown for 14 configurations to confirm benign stability and control/handling qualities of new store configurations on the F-16. These missions encompassed the majority of flight testing typically performed by S&C engineers involved with aircraft-store compatibility work, from subsonic to supersonic air-to-ground flight tests to high- $\alpha$  air-to-air combat maneuvering. While flight tests that produce benign results are desirable and beneficial to the warfighter, they are not optimal in terms of resource allocation. Furthermore, the case studies presented here are not isolated instances in the F-16's extensive store compatibility test history. Therefore, accurate and efficient predictive tools capable of identifying configurations susceptible to handling quality instabilities prior to flight test are critical to optimizing flight test funds, minimizing risk to aircrews and delivering maximum capability to the warfighter.

# II. Computational Stability and Control Analysis Approach

The large body of previous work<sup>9,10,11,12,13,14,15</sup> performed by researchers at the US Air Force Academy using the unstructured mesh solver *Cobalt*<sup>16</sup> coupled with a Detached-Eddy Simulation (DES) turbulence treatment, adaptive mesh refinement, six degree of freedom (6 DOF) motion and deforming grids for aero-elasticity has led to a high-fidelity capability for computing S&C characteristics. The proposed approach is to combine the demonstrated capabilities to perform full aircraft simulations at flight Reynolds numbers that include aircraft motion with well-developed flight test techniques for gathering the necessary data across the flight envelope through the use of aircraft maneuvers and post processing of the aircraft response using nonlinear system identification techniques. Although the prescribed aircraft motions and virtual 6 DOF maneuvers will be inspired by flight test techniques, they will take advantage of the tighter control possible with CFD to increase the efficiency of the maneuvers. Also, more complex maneuvers may be envisioned with CFD than is possible with flight test. For example, with CFD the possibility exists to determine damping and cross derivatives individually since arbitrary (non-flyable) dynamic maneuvers are possible. For example, a pitch oscillation can be combined with a maneuver where no pitch rate exists (continuous or pulsed plunge) to remove the effect of  $\dot{\alpha}$  on the combined derivative  $\overline{C}_{L_q} = C_{L_q} + C_{L_{\dot{\alpha}}}$ . In contrast, in both wind tunnel and flight tests, only "lumped" derivatives may be determined due to kinematic constraints. The individual

derivatives are normally determined during post-processing by subtracting the known (empirical) terms away from the combined derivative and gathering the leftovers.

This approach has the advantage of using flight test maneuvers such as a wind-up turn to generate the static and dynamic loads as well as the static and dynamic derivatives including important inertial and aeroelastic effects, which can give rise to non-linear results not easily predicted with the traditional approaches. Further, the approach

may be more computationally tractable than the "brute force" approach since a single calculation can expose the important dynamic effects that would normally need to be examined parametrically.

The current research will culminate in a "virtual flight test" method that may be used to directly examine the classical dynamic aircraft responses, which define flying qualities and have certification requirements. The ultimate goal is to develop a methodology for efficiently and accurately screening for nonlinear aerodynamic phenomena such as spin, tumble, lateral instabilities, limit-cycle oscillations, and tail buffet of full aircraft using a combination of static-steady and unsteady single points, rigid body motion unsteady solutions, and 6 DOF simulations that include aeroelastic effects.

# III. Data Analysis - System Identification Approach

There are many ways to post process the computed aircraft response for the purpose of stability and control analysis. The S&C data can be cast in the form of polynomials, look-up tables, neural networks, or graphs; polynomials are often used because of their inherent smoothness, which is good for robust control design, but discontinuities in the data may require a data partitioning approach. Look-up tables are very common but require an interpolation procedure to be of any practical use. In addition, the entries in the tables have to be generated by numerical differentiation. Even when large amounts of data are available, numerically differentiating the data can introduce very "noisy" results. Using a monotone procedure can help control such spurious oscillations and provides reasonable slope information<sup>Error! Bookmark not defined</sup>. Neural networks are able to approximate any non-linear function to any arbitrary degree of accuracy, but they do not provide any insight into the dynamics of the system. Graphs are good in showing trends qualitatively but are not very useful when it comes to quantitative S&C analysis, flight simulation, or control design. Here, we use system identification to determine parametric models from the computed aircraft response.

System identification (SID) is the process of constructing a mathematical model from input and output data for a system under testing, and characterizing the system uncertainties and measurement noises<sup>17</sup>. The mathematical model structure can take various forms depending upon the intended use. SID is usually applied to wind-tunnel and flight-test data to obtain accurate and comprehensive mathematical models of aircraft aerodynamics, for aircraft flight simulation, control system design and evaluation, and dynamic analysis. A very comprehensive review of SID



Figure 5. System Identification applied to aircraft, inspired by Ref. 19.

applied to aircraft can be found in Morelli and Klein<sup>18,19</sup> and Jategaonkar<sup>20,21</sup>.

Aircraft system identification can be used in cooperative approaches with CFD, to take advantage of the strength of both approaches or having one approach fill in the gaps where the other cannot be used effectively<sup>18</sup>. The wide range of SID tools that have been developed for aircraft system identification can easily be used to analyze CFD data computed for aircraft in prescribed motion. Here we follow the global nonlinear parameter modeling technique proposed by Morelli<sup>22</sup> to describe the functional dependence between the motion and the computed aerodynamic response in terms of force and moment coefficients. The goal is to find a model which has adequate complexity to capture the nonlinearities while keeping the number of terms in the model low. The latter requirement improves the ability to identify model parameters, resulting in a more accurate model with good predictive capabilities. The modeling effort is global because the independent variables ( $\alpha$ ,  $\dot{\alpha}$ , etc.) are varied over a large range. Globally valid analytical models and their associated smooth gradients are useful for optimization, robust nonlinear control design and global nonlinear stability and control analysis.

A range of techniques are implemented in a collection of computer programs called System Identification Programs for AirCraft<sup>19,23</sup> (SIDPAC). SIDPAC was developed at NASA Langley Research Center for analyzing and modeling flight-test and wind-tunnel data. SIDPAC addresses a wide range of system identification problems in a common MATLAB environment. It includes routines for experiment design, data conditioning, data compatibility analysis, model structure determination, equation-error and output-error parameter estimation in both the time and frequency domains, real-time and recursive parameter estimation, low order equivalent system identification, estimated parameter error calculation, linear and nonlinear simulation, plotting, and 3-D visualization.

For identification of aerodynamic models with SID, the aircraft system and its aerodynamics must be adequately excited with an input signal. In flight tests, the input signal may be a control input by the pilot that is distributed to the corresponding control surfaces through the flight control laws, or a sequence of separated control surface excitations that bypass the flight control laws. Flight-test input signals have to meet given maneuver and structural limits. In the wind tunnel, the input signal is typically a change in the angle of attack, the sideslip angle, or the pitch, roll and yaw rate due to a commanded motion of the test rig. The input signals are typically limited by the wind tunnel walls or by the test rig's kinematic and vibrational restrictions.

Such restrictions and limitations do not apply to input signals for CFD. Arbitrary motions may be simulated with relative ease. Only the length of the input signal is limited by the amount of computational resources available. Therefore, the input signal should be optimized to excite as much of the system dynamics as possible in the least amount of computational time. The formulation of such a time- and dynamics-optimized input signal has been the topic of previous and current research<sup>24</sup>. The choice of the optimum input signal also depends on the system to be excited, if a linear or nonlinear model is sought after, and if the flow is steady or unsteady, among other things. The length and type of the input training signals also dictates the prediction performance of the model to be identified. In selecting input training signals for the purpose of this paper, we have followed the work of O'Neill<sup>25</sup>, who developed aerodynamic requirements for input signal design and implementation.

Because we are dealing with a high-performance aircraft in this work, the idea is to use large amplitude input signals to excite the nonlinear, unsteady dynamics of the system. For the nonlinear model identification it is important to cover the entire range of the independent variables.

# IV. Flow Solver

Computations were performed using the commercial flow solver *Cobalt. Cobalt* is a cell-centered, finite volume CFD code. It solves the unsteady, three-dimensional, compressible Reynolds Averaged Navier-Stokes (RANS) equations on hybrid unstructured grids. Its foundation is based on Godunov's first-order accurate, exact Riemann solver. Second-order spatial accuracy is obtained through a Least Squares Reconstruction. A Newton sub-iteration method is used in the solution of the system of equations to improve time accuracy of the point-implicit method. Strang *et al*<sup>16</sup> validated the numerical method on a number of problems, including the Spalart-Allmaras model, which forms the core for the Detached Eddy Simulation (DES) model available in *Cobalt*. Tomaro *et al*<sup>26</sup> converted the code from explicit to implicit, enabling CFL numbers as high as  $10^6$ . Grismer *et al*<sup>27</sup> parallelized the code, with a demonstrated linear speed-up on as many as 4,000 processors. The parallel METIS (PARMETIS) domain decomposition library of Karypis *et al*<sup>28</sup> is also incorporated into *Cobalt*. New capabilities include rigid-body and 6 DOF motion, equilibrium air physics and overset grids. A coupled aeroelastic simulation capability is also being developed.

*Cobalt* uses an Arbitrary Lagrangian Eulerian (ALE) formulation to perform grid movement, where the grid is neither stationary nor follows the fluid motion but is reoriented without being deformed. The coordinate values change, but the relative positions between the grid points are unchanged. As a result, terms such as cell volume and face area remain constant and equal to the values in the original grid. The motion can include both translation and rotation. Complex motions can be defined by specifying arbitrary rotations and displacements of the grid in a motion file. This file then forms part of the required input deck for *Cobalt*. The generation of a motion file for the complicated maneuvers of this work can be tedious and error-prone. Analytical functions, e.g. for Gaussian pulses or higher harmonics, cannot be specified directly. Therefore, an interactive GUI has been developed, which takes inputs from the user of what type of maneuver, and a few key parameters, and converts those inputs into the fairly complicated rigid grid movement description in the motion file.

# V. Numerical Grid and Boundary Conditions

The grid used for this research represents a half-span, full-scale model of the F-16. It is identical to the grid used by Squires *et al*<sup>29,30</sup> for the first DES calculation over a full aircraft. The F-16 model includes the forebody bump, diverter, and ventral fin. The engine duct is modeled and meshed up to the engine face (see Figure 6). The wing-tip missile and corresponding attachment hardware and the nose boom are not modeled. The 3D hybrid grid was generated using the grid generation package  $GRIDTOOL^{31}$  and  $VGRIDNS^{32}$ , as well as the Cobalt L.L.C. grid management utility  $BLACKSMITH^{33}$ . The surface grid comprises 167,382 elements and is shown in Figure 6. Off the surface, there are eight prismatic layers. For a Reynolds number of 14,789,444, the height of the first prismatic layer



corresponds to an average wall  $y^+$  value of less than four. In total, there are 790,109 nodes in the volume grid,

Figure 6. Unstructured numerical surface grid for the half-span full-scale model of the F-16 (left) and symmetry plane of the hybrid volume grid showing the meshed inlet duct (right).

corresponding to 3,171,892 cells. Cells are concentrated in the strake vortex region.

The boundary conditions are symmetry, adiabatic solid wall for the surface of the aircraft and the engine duct, and modified Riemann invariants for the far-field boundaries. A source boundary condition based on Riemann invariants is used to create an inflow condition at the engine exhaust. A sink boundary condition is used at the engine face to model the corrected engine mass flow.

# VI. Results

# SIDPAC Results

To date, a full-scale F-16 undergoing the following prescribed motions at a low subsonic Mach number and a Reynolds number of 14,789,444 has been simulated and post processed with system identification software:

- Continuous angle of attack sweeps, 5 different pitch rates
- Plunging motion:
  - o Plunge pulses, 4 different amplitude/pulse length combinations
  - Plunge Schroeder sweep
  - Plunge DC chirp
- Pitching motion:
  - o Sinusoidal pitching motion, 3 different frequencies
  - Schroeder sweep in pitch
  - o DC chirp in pitch
- Coning motion:
  - Conventional coning motion
  - Oscillatory coning motion

The motions were defined using a newly developed interactive GUI<sup>Error! Bookmark not defined.</sup> They represent typical wind-tunnel and advanced flight-test techniques for stability and control testing as well as input signals optimized for system identification. The unsteady maneuvers were simulated using the DES version of the Spalart-Allmaras one-equation turbulence model with rotation correction (SARC-DES) to predict the effects of fine scale turbulence. Fully turbulent flow was assumed. The outer (physical) time step was set to  $\Delta t$ =0.0004s, corresponding to a non-dimensional time step of  $\Delta t^*$ =0.01. Based on previous experience<sup>34</sup> with simulating moving grids and 6DOF motion with *Cobalt*, the number of Newton sub-iterations was set to 5. The temporal damping coefficients for advection and diffusion were set to 0.05 and 0.0, respectively. The unsteady numerical simulations were initialized by steady-state solutions computed with the Spalart-Allmaras turbulence model with rotation correction.

The computations were run on up to 128 CPUs on supercomputing systems at several HPC sites, including the Arctic Region Supercomputing Center (ARSC), the U.S. Air Force Academy's Modeling and Simulation Research Center (USAFA M&SRC), the Maui High Performance Computing Center (MHPCC), and the Aeronautical Systems Center Major Shared Resource Center (ASC MSRC) at Wright Patterson Air Force Base. The majority of computations were performed on 'Iceberg', which is being operated by ARSC, and 'Blackbird', a USAFA asset.

'Iceberg' is an 800-processor IBM Power4 system comprising a combination of 92 p655+ servers, each with 8 processors and 16 GB of shared memory, 2 p690+ servers each with 32 processors and 256 GB shared memory, and 4 p655 I/O servers. The entire system has 25 TB of disk and a theoretical peak performance of five TFlops.

'Blackbird' is a 200-processor Intel Xeon and AMD Opteron beowulf-architecture cluster. Blackbird is comprised of nodes with 2-CPU 2.8Ghz Intel Xeon processors, with 2-4Gb RAM per node and a smaller number of Opteron nodes with 8-16Gb RAM. Nodes are interconnected via gigabit Ethernet. Shared I/O storage consists of approximately 110Tb.

#### A. Continuous Angle of Attack Sweeps vs. Steady and Unsteady Static Computations

In order to characterize the effects of angle of attack on the lift of an aircraft, one would normally accomplish a series of static solutions over a sufficient range of angles of attack. With prescribed grid motions, however, it would be more appealing to perform quasi-static sweeps through a range of angles of attack at a particular pitch rate. This technique would provide continuous lift curve data and would also be more efficient. The results of two or more angle of attack sweeps with different pitch rates could not only be used to determine the lift curve slope,  $C_{L_{\alpha}}$ , but

also the pitch rate derivative,  $C_{L}$ .

Here we set out to determine the maximum pitch rate (i.e. minimum solution time) that would still give quasisteady results. The angle of attack was increased from  $-5^{\circ}$  to  $60^{\circ}$  at constant angular velocities ranging from  $5^{\circ}$ /s to  $40^{\circ}$ /s using *Cobalt*'s rigid-grid motion capability. Note that the initial angle of attack of  $-5^{\circ}$  was held for 500 iterations to give the flow some time to develop. In Figure 7, the  $\alpha$ -sweep results are compared to a series of steadystate and time-averaged unsteady computations at various static angles of attack in  $10^{\circ}$  intervals. The static results are represented in Figure 7 by interconnected turquoise circles and green squares, respectively, whereas the results of the sweeps are represented by continuous lines in the colors orange, blue, red, black and magenta, corresponding to angular rates of  $5^{\circ}$ /s,  $10^{\circ}$ /s,  $20^{\circ}$ /s,  $30^{\circ}$ /s and  $40^{\circ}$ /s, respectively.

Figure 7 suggests that there is no true "steady" run using a non-zero pitch rate. Even for 5°/s – the slowest angular rate simulated here – dynamic effects exist. This is obvious at  $\alpha$ =12.5° and the angle of attack corresponding to maximum lift. However, at angles of attack below 12.5°, the lift curve slope is the same for the steady-state, unsteady and  $\alpha$ -sweep results. Thus,  $C_{L_{\alpha}}$  can be determined equally well from a "fast"  $\alpha$ -sweep and a

series of steady-state computations. Looking at the results for the different  $\alpha$ -sweeps below 12.5° angle of attack, one can also observe that the dynamic lift due to a constant pitch rate increases linearly with the pitch rate. Thus, the pitch rate derivative  $C_{L_{\alpha}}$  can be determined very efficiently by performing two "fast"  $\alpha$ -sweeps.

Note, however, that beyond 12.5° angle of attack, the flow regime is unsteady and nonlinear, rendering these two derivates meaningless. The unsteadiness is inherent to the flow and is not caused by the grid motion but rather by massive separation, vortex breakdown and vortex interactions. Here, we model this nonlinear/unsteady behavior using other, more complex dynamic motions and nonlinear system identification (SID) techniques. The results for



Figure 7. Lift coefficient vs. angle of attack for continuous angle-of-attack sweeps (different pitch rates), steady-state, and time-averaged unsteady CFD results.

this advanced modeling approach are presented now.

# **B.** Plunge Pulses

Plunge pulses were simulated numerically to quantify the effect of  $\alpha$  and  $\dot{\alpha}$  on the lift curve with a single motion. The calculation of the  $\dot{\alpha}$  effect, or its measurement, involves unsteady flow<sup>35</sup> and cannot be determined on the basis of steady-state aerodynamics, justifying the time-dependent CFD approach taken here. A pulse gives considerable detail in the frequency domain with significant cost reduction over alternative methods of calculating multiple oscillatory responses or a series of plunges with constant  $\dot{\alpha}$ . It can be used to extract pitch rate derivatives when combined with a pitch pulse. A similar procedure can be used for lateral and directional derivatives due to roll rate and yaw rate, respectively.

Four unsteady simulations were run for the half-span model of the F-16. Each simulation used a forced translational input in the direction normal to the free stream that varied temporally as a Gaussian pulse, i.e. the plunging input, h, was given by

$$h(t) = \overline{h}e^{-\ln(0.5)(2\cdot(t-t_0)/t_{12})^2}$$
(1)

where  $\overline{h}$  is the amplitude of the pulse,  $t_0$  determines the time at which the peak input occurs, and  $t_{12}$  is the pulse width at half amplitude. The parameter  $t_{12}$  determines the sharpness of the pulse and, therefore, the range of frequencies excited in the system. The first three cases considered here used  $\overline{h} = 0.01c$ , 0.1c and 1.0c (where *c* is the mean aerodynamic chord),  $t_0 = 0.17s$ ,  $t_{12} = 0.05s$ , and  $t = 0.0 \dots 0.6s$ . A total of 1,500 time steps, corresponding to 0.42 s of simulation time, were computed for each case. Each simulation took about 356 CPU hours on 64 IBM Power4 CPUs. The forth case was for  $\overline{h} = 1.0c$ ,  $t_0 = 0.34s$ ,  $t_{12} = 0.1s$ , and  $t = 0.0 \dots 1.2s$ .

Figure 8 shows a sequence of snapshots of the plunge maneuver with  $\overline{h} = 1.0c$  and  $t_{12} = 0.05s$ . The time interval between images is 0.03 s. The angle of attack is shown as a function of time in the lower left corner of each image. The snapshots depict an instantaneous vorticity iso-surface colored by velocity magnitude and a set of instantaneous



Figure 8. DES of configuration plunge pulse,  $\overline{h} = 1.0c$ ; snapshots of instantaneous iso-surface of vorticity colored by velocity magnitude and instantaneous streamlines.

streamlines.

The figure shows that during the maneuver, the flow over the wing and tail separates due to high induced angles of attack. For the chosen combination of pulse parameters, the angle of attack range is  $-50^{\circ}$  to  $+50^{\circ}$ . Note that higher amplitudes result in higher angles of attack and angle of attack rates, assuming that the pulse length is kept constant. If the amplitude is kept constant instead, longer pulses result in lower angles of attack and angle of attack rates. Part of the problem is to find combinations of the pulse parameters that minimize the computational time but also adequately excite the needed aerodynamics within reasonable limits. Ideally, the pulse should be as sharp as possible to minimize computational time. This, however, may cause numerical problems and/or an inaccurate aerodynamic response.

# 1. Nonlinear parametric modeling of plunge pulse with SIDPAC

In a first step, the lift coefficient time history computed for  $\overline{h} = 1.0c$  and  $t_{12} = 0.05s$  was low-pass filtered using a function in SIDPAC<sup>23</sup> that decomposes a signal into a deterministic signal and non-deterministic noise, using Fourier analysis. Signal and noise are separated using an optimal Weiner filter in the frequency domain. The cut-off frequency (54.2 Hz) was chosen automatically based on the data. The raw data, the filtered data, and the residual are shown in Figure 9.



Figure 9. Computed lift coefficient time history (top) compared to low-pass filtered time history (middle) and residual (bottom).

Next, the filtered lift coefficient time history was modeled in SIDPAC<sup>23</sup> using multivariate orthogonal modeling functions generated from  $\alpha$ ,  $\dot{\alpha}$  and  $\ddot{\alpha}$  as independent variables. The independent variables were computed by repeated differentiation of the displacement *h*. The maximum independent variable order for each individual orthogonal function was set to four. The maximum order for each orthogonal function was set to six. For example, an orthogonal function similar to  $\alpha^4 \cdot \dot{\alpha}^2$  would be an allowed term, but not  $\alpha^5 \cdot \dot{\alpha}$  (which exceeds the maximum independent variable order).

The number of orthogonal functions that correspond to the minimum prediction error turned out to be 11. All 11 orthogonal functions were retained. The identified multivariate polynomial model for the 11 retained orthogonal functions is:

$$C_{L_{\text{MOF}}} = C_0 + C_1 \alpha + C_2 \dot{\alpha} - C_3 \ddot{\alpha} - C_4 \alpha \dot{\alpha} - C_5 \alpha^2 + C_6 \alpha^3 + C_7 \alpha \ddot{\alpha} - C_8 \dot{\alpha}^2 + C_9 \alpha^2 \dot{\alpha}$$
(2)

Note that the identified model contains linear and higher-order static and dynamic terms, as well as nonlinear crossterms. The higher-order and nonlinear cross terms are required to capture the dynamics of this violent plunge pulse. The model coefficients of the linear terms correspond to the classic static and dynamic stability derivatives.

Figure 10 compares the multivariate polynomial model fit (black line) to the CFD data used to build the model (blue line). The red and green lines represent model fits obtained with linear and stepwise regression, respectively. The linear regression model is based on the three independent variables,  $\alpha$ ,  $\dot{\alpha}$  and  $\ddot{\alpha}$ . The stepwise regression model contains an additional cross term:

$$C_{L_{\text{swp}}} = -C_0 + C_1 \alpha + C_2 \dot{\alpha} - C_3 \ddot{\alpha} - C_4 \alpha \dot{\alpha} \tag{3}$$

It can be seen that the multivariate polynomial model fits the CFD data much better than the other two models, which is due to the inclusion of the higher-order and nonlinear cross terms.



Figure 10. Lift coefficient time history computed for a plunge pulse with  $\overline{h} = 1.0c$  and  $t_{12} = 0.05s$ , compared to different model fits.

#### 2. Model validation with dissimilar plunge pulse

Data from the other maneuver computations were used to test the predictive capability of the model identified from the large amplitude plunge pulse input. Accurate prediction for maneuvers with dissimilar inputs is a strong indicator of good modeling results.

The models were validated by using them to predict the lift coefficient time history for a plunge pulse that was different from the one used to build the models. Figure 11 compares model predictions for a pulse with  $\overline{h} = 0.1c$  and



Figure 11. Lift coefficient time history for a plunge pulse with  $\overline{h} = 0.1c$  and  $t_{12} = 0.05s$  predicted by different analytical models trained with CFD data for a dissimilar plunge pulse, compared to validation data.

 $t_{12} = 0.05$ s to CFD data that was computed for validation purposes. The lift coefficient history predicted by the multivariate polynomial model (black line) follows the trend of the validation data (blue line), although with a slight offset. This indicates that the polynomial modeling was successful in capturing the nonlinear aerodynamic functional dependence but overestimated the constant model term. This problem could potentially be solved by removing the DC bias from the  $C_L$  time history before the SID analysis and adding it back afterwards. Both the linear regression model (red line) and the stepwise regression model (green line) overestimate the peaks and slopes,

and underestimate the constant model term, demonstrating their limited predictive capabilities. The same trend is seen in Figure 12, which shows model predictions and the corresponding validation data for a wider plunge pulse



Figure 12. Lift coefficient time history for a plunge pulse with  $\overline{h} = 1.0c$  and  $t_{12} = 0.1s$  predicted by different analytical models trained with CFD data for a dissimilar plunge pulse, compared to validation data.

with  $\overline{h} = 1.0c$  and  $t_{12} = 0.1s$ .

# C. Schroeder Sweep Plunge

In a next step, even more complex motions were investigated to improve upon the model identified above. Following the experimental work of Murphy and Klein<sup>36</sup>, a wide-band forced oscillation, or Schroeder sweep, was simulated to excite the nonlinear unsteady aerodynamics of the F-16 model for system ID purposes. The Schroeder excitation signal is based on the multisine class of signals. This class of signals allows for the specific excitation of a specified bandwidth through the summation of discrete frequencies. A phase-shift presented by Schroeder<sup>37</sup> allows for minimizing the signal's peak factor. The Schroeder form is based on a sum of cosine terms with a specified phasing. An expression for displacement *h* is given below. The form is analytic in time but not smooth in frequency.

$$h(t) = \sum_{k=1}^{N} \sqrt{\frac{1}{2N}} \cos\left(\frac{2\pi kt}{T} - \frac{2\pi k^2}{N}\right)$$
(4)

For overall performance, Young and Patton<sup>38</sup> found that for a specific helicopter identification problem, a multisine gave slightly better results than the corresponding frequency sweep (chirp). This improvement appeared to be due to improved low frequency excitation with the multisine. However, Simon and Schoukens<sup>39</sup> found that the Schroeder sweep is sensitive to the signal excitation length.

The Schroeder sweep in plunge simulated with *Cobalt* started after 0.2s of time-dependent simulation and had a length of 5s. It was generated by combining equally-spaced frequencies between 0.0 Hz and 5.0 Hz. The sample rate corresponded to the time step size of 0.0004s. The phases of all sinusoidal components were shifted to obtain a zero crossing or resting condition. The resulting angle of attack range was from  $-45^{\circ}$  to  $+45^{\circ}$ .

### 1. Nonlinear parametric modeling of Schroeder sweep plunge with SIDPAC

The computed lift coefficient time history was analyzed with SIDPAC to generate yet another analytical model. As before, a number of independent variables were specified (in this case angle of attack and angular rate) and multivariate orthogonal functions were automatically generated. Based on statistic metrics, a subset of these functions were retained and expanded into a multivariate polynomial model for the lift coefficient. The model fit to the original training CFD data is shown in Figure 13.



Figure 13. System ID of data from plunge Schroeder sweep: training data (from CFD) and model fit. Independent variables are angle of attack,  $\alpha$ , and angular rate,  $\dot{\alpha}$ .

The training data is represented by a blue line, whereas the model fit is shown in red. Only minute differences between the model fit and the training data are evident, demonstrating that SIDPAC could model the data with a nonlinear function of the independent variables. The model fit shows a trend opposite to that of the training data where the sweep was suddenly and discontinuously excited from a resting condition. This is a typical problem with discontinuous input signals that are used for system identification purposes. In how far this inverse model fit has an impact on the predictive capabilities of the identified model remains to be investigated as part of the validation.

### 2. Model validation with DC Chirp Plunge

The model that was identified from the Schroeder sweep plunge response in the previous section is validated with a DC chirp plunge. A DC Chirp signal contains a static offset to alleviate the low frequency problems inherent in the traditional chirp signal<sup>25</sup>. The result is a non-symmetrical signal with a similar form but with improved performance when compared to the basic chirp excitation signal. The signal is a linear frequency sweep from zero frequency at time zero. Like the traditional chirp, the signal is analytic everywhere. The functional form is:

$$h(t) = \overline{h} (1 - \cos \omega t^2)/2 \tag{5}$$

Here, a DC chirp plunge with duration of 5s was simulated numerically for validation purposes. The frequency was ramped up from 0 Hz at time zero to 5 Hz at the end of the chirp. The amplitude was  $\overline{h} = 1.0c$ . As a result of the plunging motion, the induced angle of attack varied from -30° to +30°. The simulation of this motion required the computation of 12,500 physical time steps.

The results of the CFD validation exercise in the form of the lift coefficient time history are compared to model predictions for the same motion input signal in Figure 14. The validation data from CFD corresponds to the blue line, the model predictions to the red line. It is evident that the predictive capabilities of the model that was trained with the dissimilar Schroeder sweep plunge, can accurately predict the response of the F-16 to a DC chirp plunge. Nonetheless, the model under predicts the peak lift coefficient towards the end of the time history. These minute differences may be due to high-frequency unsteady effects that the model was not trained for, that were filtered out in the model identification procedure, or that were not excited by the Schroeder sweep training signal. However, the inverse model fit described before did not impact the present predictions because the DC chirp signal does not contain any discontinuities. Lastly, the angle of attack range excited with the DC chirp was well within the bounds of the range that was used to train the model, extrapolation beyond this range was not necessary here.



Figure 14. Validation of plunge Schroeder sweep trained model for lift coefficient with plunge DC chirp data.

# 3. Model validation with plunge pulse

Next, the same model is used to predict the response of the F-16 to one of the plunge pulses from above. The analytical model predictions are compared to the validation CFD data in Figure 15. As before, the blue line denotes the validation data and the red line the model predictions. The model predictions follow the trend of the validation data throughout the entire lift coefficient time history; however, the model slightly over-predicts the peak values. A small phase shift is also apparent. It is remarkable, however, that the unsteady overshoot of the aircraft lift coefficient response after 2s is well predicted by the model. Clearly, the signal that was used to train the model excited a wide range of the system dynamics.



Figure 15. Validation of plunge Schroeder sweep trained model for lift coefficient with plunge pulse data.

### **D.** DC Chirp in Pitch

All of the previous motions were in plunge. In the following, we investigate motions in pitch. This time, a dc chirp signal is used to train a multivariate polynomial model and other, dissimilar motion signals are used to validate the model.

# 1. Nonlinear parametric modeling of DC chirp in pitch with SIDPAC

As before, the structure of the model to be identified from the DC chirp in pitch training data was determined using multivariate orthogonal functions generated by Gram-Schmidt orthogonalization, ordered by dynamic programming. The independent variables used were the angle of attack, the pitch rate and the angular acceleration. Figure 16 shows how the model structure was determined using statistical measures. The number of orthogonal functions is increased until the prediction error of the model reaches a minimum. This is associated with a certain coefficient of determination,  $R^2$ , which is a measure of "goodness" of the model fit. Note that  $R^2$  does not have a maximum where the prediction error has a minimum, however, for the sake of robustness and compactness of the model, the number of orthogonal functions retained is not further increased as the benefits in terms of a higher  $R^2$ 



Figure 16. System ID of DC chirp in pitch with multivariate orthogonal functions: prediction error (top), and (bottom) vs. number of orthogonal functions. The red diamond indicates the number of orthogonal functions for which the prediction error has a minimum.

value are minimal.

In the present case, 11 orthogonal modeling functions were retained and were expanded into an ordinary multivariate polynomial model for the normal force coefficient. The free parameters of the models were estimated using ordinary least-squares regression. The final nonlinear multivariate polynomial model took the following form:

$$C_{N_{\text{MOF}}} = C_0 + C_1 \alpha + C_2 q - C_3 \dot{q} - C_4 \alpha^2 - C_5 \alpha^4 + C_6 \alpha q^2 - C_7 \alpha \dot{q} + C_8 q \dot{q} + C_9 q^2 \dot{q} + C_{10} q^3 - C_{11} \dot{q}^2$$
(6)

For illustrative purposes, the same data was used to estimate the parameters of a least-squares linear model:

$$C_{N_{\rm LESO}} = C_0 + C_{N_a} \alpha + C_{N_a} q + C_{N_a} \dot{q}$$
(7)

The parameters of the model resemble the classical stability derivatives. The model fit of both models to the training data is shown in Figure 17, which shows the normal force coefficient as a function of time. The blue curve represents the training CFD data, the red and magenta curves denote the nonlinear multivariate polynomial model and linear model, respectively. The nonlinear model fit to the training data is significantly better for the low frequency excitation response in the beginning of the motion. The high-frequency response is modeled almost equally well by both models. Note that despite its complexity the nonlinear model is not able to fit the fluctuations in the normal force coefficient at around 1s into the motion. These fluctuations are associated with the inherently unsteady vortical flow field and are not due to the dynamic motion. Below, both models are validated by using them to predict the response to sinusoidal pitch motion at three different frequencies.



Figure 17. System ID of DC chirp in pitch: Training data (from CFD) and model fit. Independent variables are angle of attack,  $\alpha$ , pitch rate, q, and angular acceleration,  $\dot{q}$ .

# 2. Model Validation with Sinusoidal Pitching Motions

A conventional wind-tunnel technique used for many years to estimate dynamic damping derivatives such as  $C_{m_q}$  is to perform a 1 DOF, planar, forced-oscillation test. For this method the model is placed at various angles of attack in a wind tunnel and allowed to undergo forced sinusoidal oscillations at different frequencies and usually relatively small amplitudes. Oscillations are usually done about the pitch, roll and yaw axes. This technique produces combined, or lumped, derivatives. This occurs because the angle-of-attack rate and pitch rate are kinematically constrained to be equal.

Inspired by this experimental technique, forced pitch oscillations were simulated numerically. The initial angle of attack of  $30^{\circ}$  was held for 0.2 s to give the flow some time to develop. Thereafter, it was varied according to the following relation:

$$\alpha(t) = 30^\circ + 15^\circ \times \sin(2\pi \times f \times t), \qquad (8)$$

where f is the frequency. Three different frequencies were simulated, 2.0 Hz, 1.0 Hz, and 0.5 Hz. For both the 2.0 Hz and 1.0 Hz case, a total of 5,500 time steps, corresponding to 2.2 s of simulation time or four and two complete cycles, respectively, were computed. These simulations took about 1,860 CPU hours on 64 IBM Power4 CPUs of 'Iceberg'. For the 0.5 Hz case, a total of 10,500 time steps, corresponding to 4.2s of simulation time, were computed. This computation was performed on 64 processors on 'Blackbird' and took 95h of wall clock time, corresponding to 6,100 CPU hours.

Figure 18 is a sequence of images visualizing the flow computed for this sinusoidal pitching motion with imposed amplitude and angular frequency. Each image depicts an instantaneous vorticity iso-surface colored by velocity magnitude. The angle of attack is shown as a function of time in the lower left corner of each image. Note that the flow field is seen to be unsteady even at the beginning of the simulation where the angle of attack is static. This is due to the strake vortex experiencing vortex breakdown, and the massive flow separation over the main wing. At dynamic angles of attack, the flow field undergoes drastic changes, including the appearance and disappearance of a forebody vortex, the burst and reformation of the strake vortex, and the formation of a burst



Figure 18. DES of F-16 in sinusoidal pitching motion; instantaneous vorticity iso-surface colored by magnitude of velocity,  $\alpha(t) = 30^{\circ} + 15^{\circ} \times \sin(2\pi \times 1.0 Hz \times t)$ .

main-wing vortex.

These nonlinear phenomena give rise to nonlinear behavior of the aerodynamic forces and moments. The blue line in Figure 19 shows the normal force coefficient  $C_N$  as a function of the angle of attack computed for f=2.0 Hz. For this case, the dynamic lift curve features a wide "hysteresis" loop that occurs because the flow at increasing angle of attack features different characteristics to that at decreasing angle of attack. The "jump" in the lift coefficient at the beginning of the first cycle is due to the infinite acceleration that occurs when the sinusoidal motion starts at t = 0.2 s. The infinite acceleration is due to a discontinuity in the second derivative of the angle of attack. As a result, the lift coefficient increases from its static value to the dynamic value that corresponds to the angle of attack rate given by the sine wave. The associated transient is seen to have disappeared in the second cycle.

Also shown in Figure 19 for validation purposes are the predictions of the nonlinear model identified earlier. The model predictions compare very favorable with the validation data throughout the entire pitch cycle. The width of



Figure 19. Validation of DC chirp in pitch trained models for normal force coefficient with sinusoidal pitch data, *f*=2.0 Hz.

the hysteresis loop is matched.

The situation is not as favorable for the f=1.0 Hz case, which is shown in Figure 20. The red and green lines correspond to nonlinear model predictions, the magenta line corresponds to predictions of the linear model. As before, the blue line represents the validation CFD data. The nonlinear model predicts the correct trends; however, in the pitch-down part of the cycle, the model over-predicts the normal force coefficient. The linear model predictions are by far inferior, showing the largest discrepancies to the validation data at the high angles of attack. It



Figure 20. Validation of DC chirp in pitch trained models for normal force coefficient with sinusoidal pitch data, f=1.0 Hz.

cannot predict the nonlinear loss of lift do to its linear nature.

Figure 21 shows the situation for the lowest frequency, f=0.5 Hz. For this case, the hysteresis loop is rather flat. Again, the nonlinear model does predict hysteresis, however, it does not predict the pitch down part of the hysteresis loop as well as it did for f=2.0 Hz. Also, it cannot predict the high-frequency fluctuations of the normal force coefficient, which are due to the inherent flow field unsteadiness. Recall that this inherent unsteadiness was also not



Figure 21. Validation of DC chirp in pitch trained models for normal force coefficient with sinusoidal pitch data, *f*=0.5 Hz.

modeled during the training of the model.

The conclusion from this exercise is that the multivariate modeling approach is capable of modeling the response to sinusoidal oscillatory motion, however, the predictive capabilities of the model is limited by the dynamics that were excited with the training input signal.

### Validation and Verification

It is important to build confidence in the previously demonstrated method for it to become a part of aircraft/store certifications. An initial look at validation and verification (V&V) is accomplished by comparing CFD and SIDPAC solutions to solutions obtained using ATLAS. This section presents comparisons between CFD, SIDPAC, and ATLAS for steady state solutions and a DC Chirp maneuver. Then the SIDPAC nonlinear model resulting from the DC Chirp maneuver is used to create sinusoidal pitching maneuvers at three frequencies with results compared to ATLAS. It is important to realize that this is just an initial attempt at V&V since there is no grid sensitivity study and the configuration is slightly different than the ATLAS configuration. The ATLAS configuration includes wing tip and under wing pylons whereas the CFD model is a clean configuration. Also, the propulsion system is only an approximation obtained from the open literature, whereas the ATLAS propulsion system is a Lockheed proprietary model.

As in the previous section, all solutions were computed using Cobalt version 3.0 from Cobalt Solutions LLC. Steady-state solutions and initiation of time-accurate solutions were computed using the Reynolds-Averaged Navier-Stokes (RANS) turbulence model of Spalart-Allmaras with Rotation Corrections (SARC), 1st order-accuracy in time, and a time step commensurate with a CFL number of one million. Time-accurate solutions were computed with the Detached-Eddy Simulation hybrid RANS-Large Eddy Simulation turbulence model with SARC as the underlying RANS model. A time-step size of 0.0005 seconds was chosen as a conservative estimate for all of the time-accurate simulations.

All of the computations were run on 128 to 256 CPUs on two different supercomputing systems. All of the static solutions were accomplished on "jvn" at the US Army Research Lab in Aberdeen Proving Ground, Maryland. This machine is a Linux Networx Evolocity II with 2048 Intel Xeon EM64T processors running at 3.6 GHz and connected via Myrinet. The dynamic solutions were accomplished on "falcon", a 2,048-processor AMD Opteron (2.8 GHz) cluster with 1,024 XC Compute Nodes (2 processors/node) connected with Infiniband Interconnect.

Single-point solutions were computed for a range of angles of attack from 0 to 30 degrees at a Mach number of 0.6 and an altitude of 5,000 ft. An initial steady-state solution was accomplished followed by 3,000 time-accurate iterations with 2<sup>nd</sup>-order temporal and spatial accuracy and 3 Newton subiterations per time step. From this converged solution, an additional 4000 time steps were computed at each angle of attack up to 15 degrees. The converged solution at 15 degrees was used to initialize the remainder of the static runs. Each time-accurate iteration took approximately 5.7 seconds using 128 processors on "jvn". Typically, the last 2,000 iterations of each run were time-averaged to compute the aerodynamic coefficient values reported in the results. The unsteady bounds shown in the results were taken as the minimum and maximum values observed over the same number of iterations. All coefficients of this section are presented without a defined scale to allow presentation in the open literature.

Figure 22 below shows the variation in  $C_L$  and  $C_M$  as a function of  $\alpha$ , as well as  $C_D$  as a function of  $C_L$  for CFD time-averaged solutions, CFD unsteady maximum and minimum values, and ATLAS. The left hand side of Figure 22 depicts the  $C_L$  and  $C_M$  versus  $\alpha$ .  $C_L$  and  $C_M$  values resulting from solutions up to an  $\alpha$  of 15 degrees compare very well to ATLAS and exhibit essentially no unsteady effects as measured by the difference between the minimum, maximum, and average CFD solutions.  $C_L$  and  $C_M$  values resulting from solutions above an  $\alpha$  of 15 degrees exhibit significant unsteadiness and the  $C_L$  resulting from time-averaged CFD has a measurable difference from the ATLAS solutions above 20 degrees  $\alpha$ . This difference may be due to either an insufficient refinement of the grid to capture some relevant physics or the difference in configuration between the CFD grid and ATLAS. The drag coefficient,  $C_D$ , as a function of  $C_L$  is presented in the right hand side of Figure 22. The CFD solutions compare very well for low values of CL but overpredict the drag for the mid-range of CL and underprict the drag for the solutions computed at the highest  $\alpha$ 's. In general, the comparisons are reasonable and demonstrate the usefulness of CFD for computing static solutions.



Figure 22: Lift, drag, and moment coefficients for CFD simulations compared to ATLAS.

A chirp input signal was used to drive the pitch motion of the grid for the dynamic solution. The chirp signal was defined at a time step of 0.0005 seconds (identical to solver time step) for 5 seconds, and linearly spanned the frequency band from 0 to 5 Hz. A total of 10,000 time-accurate time steps were required to accomplish the entire pitch chirp maneuver. Five subiterations were accomplished at each time step for this maneuver. Using 128 processors on "falcon", each iteration took 6 seconds of wall-time.

The output data from the CFD solution of the DC Chirp was processed using the SIDPAC software allowing terms up to fourth order. The resulting nonlinear equation for  $C_L$  as a function of  $\alpha$ , q, and q is expressed in equation (9) and required 15 terms to fit.

$$C_{L}(\alpha, q, \dot{q}) = C_{1} + C_{2}\alpha + C_{3}q + C_{4}\dot{q} + C_{5}\alpha q^{2} + C_{6}\alpha q + C_{7}\alpha \dot{q} + C_{8}\alpha^{3} + C_{9}q\dot{q} + C_{10}q^{3} + C_{11}\alpha q\dot{q} + C_{12}\dot{q}^{2} + C_{13}q^{2}\dot{q} + C_{14}\alpha \dot{q}^{2} + C_{15}q^{4}$$
(9)

The terms are ordered in importance to the numerical fit of the data (i.e.  $\alpha$  is more important than q which is more important than q, etc.). The resulting CM equation required 25 terms and is expressed in equation (10) also ordered by importance.

$$C_{M}(\alpha, q, \dot{q}) = C_{1}q + C_{2} + C_{3}\alpha^{2} + C_{4}q^{4} + C_{5}\dot{q} + C_{6}\alpha q\dot{q} + C_{7}\alpha\dot{q}^{2} + C_{8}\alpha + C_{9}q^{2} + C_{10}\alpha^{4} + C_{11}\alpha^{2}q^{2} + C_{12}\alpha q^{2} + C_{13}q^{3}\dot{q} + C_{14}q\dot{q}^{2} + C_{15}\alpha^{3}q + C_{16}\alpha q\dot{q}^{2} + C_{17}\dot{q}^{3} + C_{18}\dot{q}^{2} + C_{19}\alpha q + C_{20}\alpha q^{3} + C_{21}\alpha\dot{q} + C_{22}\alpha\dot{q}^{3} + C_{23}q\dot{q}$$
(10)

Figure 23 depicts  $C_L$  and  $C_M$  as a function of time for the DC Chirp maneuver. The ATLAS values were only shown for the range of validity of the model resulting in the data drop out at the peaks and valleys of  $C_L$  and in between the peaks of  $C_M$ . In general, there is a qualitative match between the unsteady CFD, SIDPAC non-linear model of the CFD, and ATLAS. In the  $C_L$  versus time plot presented in the left hand side of Figure 23 one can see the match in frequency between the three data sets but ATLAS solutions exhibit an additional behavior between the peaks and valleys not observed by either the CFD or SIDPAC solutions. The  $C_M$  versus time results of the right hand side of Figure 23 again show a good match in the frequency but the valleys show an underprediction in the negative  $C_M$  values corresponding to the static solution discrepancies in  $C_M$  at  $\alpha$ 's above 15 degrees observed in Figure 22.



Figure 23: Lift and moment coefficient as a function of time for CFD, SIDPAC nonlinear model of CFD data, and ATLAS.

After obtaining the SIDPAC model of the F-16, solutions were computed with the model for sinusoidal pitching about a 15 degree  $\alpha$  with frequencies of 1Hz, 2Hz, and 3Hz and compared to ATLAS. Figure 24 depicts C<sub>L</sub> and C<sub>M</sub> for the 1Hz pitching maneuver. The C<sub>L</sub> as a function of  $\alpha$  is presented in the left hand side of Figure 24. The simulation starts at an  $\alpha$  of 15 degrees, follows the lower portion of the right hand C<sub>L</sub> loop to the right during the up stroke, follows the upper portion of the right hand loop during the down stroke until 15 degrees  $\alpha$  at which point it crosses to the lower portion of the left hand loop until a 0 degrees  $\alpha$ . The up stroke from 0 degrees  $\alpha$  follows the upper portion of the left hand curve until reaching 15 degrees  $\alpha$  again. As one can see, the orientation of the two loops are matched with the ATLAS simulation showing a collapsed loop on the lower angles of attack and a loop with a lower C<sub>L</sub> on the upper end of  $\alpha$  values. These differences correspond to the differences seen in the time history of C<sub>L</sub> presented in Figure 23. On the other hand, the C<sub>M</sub> values are in quite good agreement with a match in the orientation of the loops and even a match in values as well. This is surprising since the C<sub>M</sub> time history from CFD show measurable differences.



Figure 24: Lift and moment coefficient data resulting from a sinusoidal pitching maneuver symmetric about 15 deg at 1Hz. SIDPAC nonlinear model of CFD data trained by a DC Chirp maneuver compared to ATLAS.

Figure 25 and Figure 26 depict  $C_L$  and  $C_M$  for the 2Hz and 3Hz pitching maneuvers, respectively. The results are very similar to the 1Hz case with a slightly larger difference in the down stroke and up stroke values of  $C_L$  on the left hand loop. As in the 1Hz case the CM comparisons are very good.



Figure 25: Lift and moment coefficient data resulting from a sinusoidal pitching maneuver symmetric about 15 deg at 2Hz. SIDPAC nonlinear model of CFD data trained by a DC Chirp maneuver compared to ATLAS.



Figure 26: Lift and moment coefficient data resulting from a sinusoidal pitching maneuver symmetric about 15 deg at 3Hz. SIDPAC nonlinear model of CFD data trained by a DC Chirp maneuver compared to ATLAS.

One important note is that it took only seconds to build the SIDPAC model and produce the figures shown above once the CFD simulations were complete. The CFD simulations can be accomplished in a day depending on the number of processors available on the machine for each condition and configuration.

# VII. Conclusions

The SEEK EAGLE Process has been described and two case studies were presented to present a baseline of store certification processes and then a novel new approach was presented for developing a simulation tool for additional configurations resulting from new weapons or new combinations of weapons on an aircraft. The new method of developing an efficient computational method for accurately determining static and dynamic stability and control (S&C) characteristics of a high-performance aircraft as well as the aircraft response to pilot input has been presented. The approach presented herein uses a high-fidelity CFD code to simulate the response of an F-16 to prescribed motions, which are specifically designed to take full advantage of the unique capabilities of the CFD environment while minimizing the computational time. The dynamic motions simulated so far include forced pitching oscillations, coning motion, oscillatory coning motion, plunge pulses, DC chirps in plunge and pitch, Schroeder sweeps in plunge and pitch, and continuous angle-of-attack sweeps. The results of the simulations in the form of non-dimensional force and moment coefficients have been analyzed with aircraft system identification techniques that have previously been used successfully with data from many different flight test programs and wind tunnel experiments. The identified nonlinear parametric models provide an excellent fit to the CFD training data and show good predictive capabilities when subjected to motions that were not used to train the models. It was also demonstrated that time-accurate  $\alpha$ -sweeps offer a faster and more efficient approach to predicting the linear lift curve slope and pitch rate derivative than the conventional point-by-point steady-state approach. The sweeps provide high-density, continuous data which may not be practical to obtain by steady-state CFD. Comparison of the numerical results with experimental data will be the subject of a forthcoming paper as approval to use an available experimental database is still pending at the time of this writing. Additionally, work is also underway to include the effects of moving control surfaces into the simulations in order to incorporate control effects into the system models.

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# References

<sup>1</sup>M. Withrow, "Dr. Raymond Gordnier Discusses the Research Direction of Advanced Computational Methods," in *Air Force Research Laboratory Horizons*, April 2005.

<sup>2</sup>D. T. Ward, "Introduction to Flight Test Engineering," Elsevier, 1993.

<sup>3</sup>R. D. Kimberlin, "Flight Testing of Fixed-Wing Aircraft," AIAA Educational Series, 2003.

<sup>4</sup>D. Stinton, "Flying Qualities and Flight Testing the Airplane," AIAA Educational Series, 1996.

<sup>5</sup>AGARD Advisory Report 305, Fluid Dynamics Panel Working Group 16, "Cooperative Programme on Dynamic Wind Tunnel Experiments for Maneuvering Aircraft," 1996.

<sup>6</sup>B. N. Pamadi, "Performance, Stability, Dynamics, and Control of Airplanes," AIAA Educational Series, 1998.

<sup>7</sup> B. L. Stevens, F. L. Lewis, "Aircraft Control and Simulation," second ed., John Wiley & Sons, 2003.

<sup>8</sup>L. L. Green, A. M. Spence, and P. C. Murphy, "Computational Methods for Dynamic Stability and Control Derivatives," *AIAA Paper 2004-0015*, 2004.

<sup>9</sup>J. R. Forsythe, K. A. Hoffmann, and F. F. Dieteker, "Detached-Eddy Simulation of a Supersonic Axisymmetric Base Flow with an Unstructured Flow Solver," *AIAA Paper 2000-2410*, June 2000.

<sup>10</sup>J. R. Forsythe, and S. H. Woodson, "Unsteady CFD Calculations of Abrupt Wing Stall using Detached-Eddy Simulation," *AIAA Paper 2003-0594*, Jan 2003.

<sup>11</sup>S. A. Morton, M. B. Steenman, R. M. Cummings, and J. R. Forsythe, "DES Grid Resolution Issues for Vortical Flows on a Delta Wing and an F-18C," *AIAA Paper 2003-1103*, Jan. 2003.

<sup>12</sup>J. R. Forsythe, K. D. Squires, K. E. Wurtzler, and P. R. Spalart, "Detached-Eddy Simulation of Fighter Aircraft at High Alpha," *Journal of Aircraft*, Vol. 41, No. 2, pp. 193-200, 2004.

<sup>13</sup>J. R. Forsythe, C. M. Fremaux, and R. M. Hall, "Calculation of Static and Dynamic Stability Derivatives of the F/A-18E in Abrupt Wing Stall Using RANS and DES," in *Proceedings of International Conference for CFD*, Toronto, Canada, 2004.

<sup>14</sup>M. Claus, S. A. Morton, and R. Cummings, "DES Turbulence Modelling on the C-130 Comparison Between Computational and Experimental Results," *AIAA Paper 2005-884*, 2005.

<sup>15</sup>S. A. Morton, and J. R. Forsythe, K. D. Squires, R. M. Cummings, "Detached-Eddy Simulations of Full Aircraft Experiencing Massively Separated Flows," *Computational Fluid Dynamics Journal ISCFD Japan*, Vol. 13, No. 3, January, 2005.

<sup>16</sup>W. Z. Strang, R. F. Tomaro, and M. J. Grismer, "The defining methods of Cobalt<sub>60</sub>: a parallel, implicit, unstructured Euler/Navier-Stokes flow solver," *AIAA Paper 1999-0786*. 1999.

<sup>17</sup>L. Ljung, "System Identification. Theory for the User," Prentice Hall Information and System Sciences Series,

<sup>18</sup>E. A. Morelli and V. Klein, "Application of System Identification to Aircraft at NASA Langley Research Center," *Journal of Aircraft*, Vol. 42, No. 1, January-February 2005, pp. 12-25.

<sup>19</sup>V. Klein and E. A. Morelli, *"Aircraft System Identification – Theory and Practice,"* AIAA Educational Series, 2006.

<sup>20</sup>R. V. Jategaonkar, "Flight Vehicle System Identification – A Time Domain Methodology," Progress in Astronautics and Aeronautics, Vol. 216, F. K. Lu, editor, AIAA, 2006.

<sup>21</sup>R. V. Jategaonkar, D. Fischenberg, W. Gruenhagen, "Aerodynamic Modeling and System Identification from Flight Data – Recent Applications at DLR," *Journal of Aircraft 2004*, 0021-8669 vol.41 no.4 (681-691), 2004.

<sup>22</sup>E. A. Morelli, "Global Nonlinear Parametric Modeling with Application to F-16 Aerodynamics," ACC Paper WP04-2, Paper ID i-98010-2, American Control Conference, Philadelphia, Pennsylvania, June 1998.

<sup>23</sup>E. A. Morelli, "System IDentification Programs for AirCraft (SIDPAC)," AIAA Paper 2002-4704, 2002.

<sup>24</sup>E. A. Morelli, "Practical Input Optimization for Aircraft Parameter Estimation Experiments," NASA CR 191462, May 1993.

<sup>25</sup>C. R. O'Neill and A. S. Arena, "Comparison of Time Domain Training Signals for CFD Based Aerodynamic Identification," *AIAA Paper 2004-209*, 2004.

<sup>26</sup>R. F. Tomaro, W. Z. Strang, and L. N. Sankar, "An implicit algorithm for solving time dependent flows on unstructured grids," *AIAA Paper 1997-0333*, 1997.

<sup>27</sup>M. J. Grismer, W. Z. Strang, R. F. Tomaro, and F. C. Witzemman, "Cobalt: A Parallel, Implicit, Unstructured Euler/Navier-Stokes Solver," *Adv. Eng. Software*, Vol. 29, No. 3-6, pp. 365-373, 1998.

<sup>28</sup>G. Karypis, K. Schloegel, and V. Kumar, "*Parmetis: Parallel Graph Partitioning and Sparse Matrix Ordering Library, Version 3.1*," Technical report, Dept. Computer Science, University of Minnesota, 2003.

<sup>29</sup>K. D. Squires, J. R. Forsythe, K. E. Wurtzler, W. Z. Strang, R. F. Tomaro, M. J. Grismer, P. R. Spalart, "Towards Prediction of Aircraft Spin," in Proceedings of the 2001 DoD High Performance Users Group Conference, Biloxi, 2001.

<sup>30</sup>K. D. Squires, J. R. Forsythe, S. A. Morton, D. C. Blake, M. Serrano, K. E. Wurtzler, W. Z. Strang, R. F. Tomaro, P. R. Spalart, "Analysis of Full Aircraft with Massive Separation using Detached-Eddy Simulation," in Proceedings of 2002 DOD

High Performance Computing Users Group Conference, Austin, TX, 2002.

<sup>31</sup>S. Samareh, "GridTool: A Surface Modeling and Grid Generation Tool," *Proceedings of the Workshop on Surface Modeling, Grid Generation, and Related Issues in CFD Solutions*, NASA CP-3291, May 9-11, 1995.

<sup>32</sup>S. Pirzadeh, "Progress Toward A User-Oriented Unstructured Viscous Grid Generator," AIAA Paper 96-0031, 1996.

<sup>33</sup>Cobalt Solutions, "Blacksmith User's Manual," Version 3.0, 2005.

<sup>34</sup>J. Forsythe, W. Strang, K. Squires, "Six Degree of Freedom Computation of the F-15E Entering a Spin," *AIAA Paper 2006-*858, 2006.

<sup>35</sup>B. Etkin, "Dynamics of flight: stability and control", 3<sup>rd</sup> ed., Wiley, New York, 1996.
 <sup>36</sup>P. C. Murphy, and V. Klein, "Validation of Methodology for Estimating Aircraft Unsteady Aerodynamic Parameters from Dynamic Wind Tunnel Tests," AIAA Paper 2003-5397, 2003.

<sup>37</sup>M. Schroeder, "Synthesis of Low-Peak Factor Signals and Binary Sequences with Low Autocorrelation," IEEE Transactions on Information Theory, Vol. 16, Jan 1970, pp. 85-89.

<sup>38</sup>P. Young, and R.J. Patton, "Comparison of test signals for aircraft frequency domain identification," AIAA J. Guidance, Dynamics and Control, vol. 13, no. 3, pp. 430-438, May-June 1990.

<sup>39</sup>G. Simon, and J. Schoukens, "Robust Broadband Periodic Excitation Design," IEEE Transactions on Instrumentation and Measurement, Vol. 49, No. 2, April 2000, pp. 270-274.