4-2018

Development of a Static Aeroelastic Database Using NASTRAN SOL 144 for Aircraft Flight Loads Analysis from Conceptual Design Through Flight Test

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Development of a Static Aeroelastic Database Using NASTRAN SOL 144 for Aircraft Flight Loads Analysis from Conceptual Design Through Flight Test

by

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A thesis submitted to the College of Engineering at
Florida Institute of Technology
in partial fulfillment of the requirements
for the degree of

Master of Science
in
Aerospace Engineering

Melbourne, Florida
April 2018

We the undersigned committee hereby approve the attached thesis, "Development of a Static Aeroelastic Database Using NASTRAN SOL 144 for Aircraft Flight Loads Analysis from Conceptual Design Through Flight Test " by Ryan M. Haughey.

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Abstract

Title: Development of a Static Aeroelastic Database Using NASTRAN SOL 144 for Aircraft Flight Loads Analysis from Conceptual Design through Flight Test

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This paper describes a method for predicting aircraft aerodynamic and inertial loading on a structural finite element model, (FEM) based on static aeroelastic coefficients. These coefficients are computed via interpolated spline methods within NASTRAN Solution 144 (static aeroelastic solution) and the "TRIM" module to connect the doublet-lattice model (DLM) and the structural finite element model for a coupled solution. The database is created by selecting key breakpoints where linear interpolation techniques can be utilized to develop and predict static aeroelastic coefficients for the prediction of any aircraft state for a given transient solution. This method is applicable from conceptual design through flight test.

The methodology described in this paper is essential for aircraft design and analysis from initial stages of conceptual design through flight test and beyond. The procedure for analysis varies slightly between the analysis types. However, the premise is, in general constant. The variation stems most significantly from the source and reliability of the data.

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Acknowledgement

I would like to acknowledge the Northrop Grumman Corporation for assisting in funding my education. Their continuous investment in my growth both financially and in experience and education has helped to develop my skills.

I would also like to acknowledge my advisor, Dr. Razvan Rusovici for his continued support over the past eight years working on multiple design projects and assisting in undergraduate and graduate research.

Finally, I'd like to thank my co-workers, many of whom I had countless discussions on this particular topic, and whose ideas helped to aid the foundation this paper and who dealt with my early departures from the work day in the later portion of the development of this thesis.

Dedication

I would like to dedicate this paper to my parents, whose continued love and support is by far the largest contributing factor to my success. They have supported my dreams and have helped to guide me to where I am today.

Motivation

Current Aircraft Design Practices

The field of flight loads analysis is over seemingly a small community of specialized engineers, as compared to fields such as structural analysis or design, however the data and analysis they produce is critical for the success of an aircraft development or modification program. The common adage of aircraft design loads is that the loads are too high. Consider the design of a brand-new airframe. Through the design phases, maturation of data, process and other developments tend to cause iterations or revisions of analysis. Through these cycles, structural analysts are working with structural design engineers to optimize the airframe to lowest positive margins. If iteration i-1 has loads which are greater than iteration i, the program can claim, "The loads from iteration i-1 were too high! Now the structure has to be redesigned so that the aircraft can hit performance requirements." Whereas in the case where iteration i-1 loads are less than that of iteration i, the program can claim, "The loads from iteration i are too high! Now the structure has to be redesigned to meet airframe strength requirements."

It is a continuous balance where the loads engineer must be sure that the changes between cycles are gradual and predictable with aircraft (and data) maturation. For this to occur, quick and computationally inexpensive methods must be utilized so that incremental change impact can occur and the process for which the loads are developed is consistent through each phase. Having consistency in methodology allows the analysts to focus on the development of increasing fidelity, rather than developing new methods based on program maturation. This paper aims to produce a methodology which can be used as a framework for a common method from the aircraft conceptual design phase through flight test.

There is very little documentation on the fundamentals of aircraft loads analysis. One textbook exists by Lomax, titled "Structural Loads Analysis for Commercial Transport Aircraft: Theory and Practice" [1]. There are, however, no textbooks dedicated to the structural loads analysis of military aircraft.

Even still, the Lomax textbook is significantly outdated and does not contain methods which are required for design today's aircraft, commercial or military. The major reasons being that traditional loads development typically neglects aircraft aeroelastic effects and the data produced has turned from external load tables and diagrams to distributed FEM loads for aircraft detail analysis. This thesis aims to tackle both of those "new era" methodologies of aircraft design through the usage of NASTRAN.

"In the design of aircraft, it is important to have an accurate simulation of both the structural characteristics and the aerodynamic characteristics of the vehicle. For [quasi-static] aerodynamic loads, MSC/NASTRAN uses unsteady aerodynamics at zero reduced frequency." [3] The methods in NASTRAN have become accepted practices for many structural analysis methods for major defense contractors and civilian engineering companies. It is assumed, in this paper, that the methods of NASTRAN are valid and that the results of the NASTRAN codes are validated against industry standard.

A concern outlined in "Design Loads for Future Aircraft" [7], is stated as "With the increased use of active control systems on aircraft, there is currently a strong need to revisit some concepts used for conventional aircraft and to identify the correction to be brought forward to existing procedures to compute the several loads affecting a military aircraft and the effect of the active control system. Special attention has been given to cover these items." As well as "During the past few years there has been an increased interest of the aircraft community on design loads for aircraft". Consequently there was a workshop in 1996 SC73 on "Loads and Requirements for Military Aircraft" Elastic effects on design loads were presented at a Workshop: "Static Aeroelastic Effects on High Performance Aircraft." These claims reflect the date of the release of the paper, which was 2002.

It is difficult to claim that the methods presented in this thesis are groundbreaking and novel in that the methods used in industry practice vary greatly from company to company and are typically held behind proprietary limits.

However, it can be said that the methods presented are unique in that there is not literature that has been found to exist in such detail on the topic, and methods presented in textbooks are only consistent with the foundation of the ideas presented by this thesis. The total method presented (the development of the unit component loads database using NASTRAN) is on the whole, a novel method.

Aircraft Design Phases

There are four main phases of aircraft development that this paper focuses on. These phases are:

1. **Conceptual Design** – The initial design phase of an aircraft development program where trade studies are conducted. Crude analysis is conducted, typically with low fidelity models in order to run optimization simulations for initial system level sizing requirements.

2. **Preliminary Design** – The second design phase of an aircraft development program where one of the trade study designs from conceptual phase is chosen to develop system level specification requirements.

3. **Detail Design** – The third design phase of an aircraft development program where detail analysis and design is conducted for final aircraft design and build. Typically at this point in the aircraft development program no significant changes to the baseline design are made, as to not impede development of surrounded and connected systems and structure.

4. **<u>Flight Test</u>** – The fourth phase of an aircraft development program where the vehicle is tested through a flight test program. This paper is particularly interested in the envelope expansion portion of the flight test program, which is the responsibility of the loads team.

Conceptual design and flight test are perhaps two of the most challenging stages of aircraft loads analysis because of the uncertainty of data. As the preliminary design matures, and the detail design phase progresses, the aircraft design data should mature to a point where the amount of conservative assumptions tends to go towards zero. Unfortunately, during the conceptual design phase, the loads analyst must make many assumptions so that the design optimization and trade studies can be efficiently processed.

"Aircraft conceptual design traditionally utilizes simplified analysis methods and empirical equations to establish the basic layout of new aircraft." [4] The methods provided in this paper reflect this theory.

During the flight test phase of a program, good correlation between the predicted loading does not disturb the development of flight test efforts, such as envelope expansion. [10] Typically, real time data prediction is needed as the aircraft is undergoing the envelope expansion phase of the flight test program. Real time data is necessary so that the engineers can safely assess and predict loading at the next test point in the sky, based on correlated predicted data and data from the latest maneuver.

Aircraft Flight Loads Requirements

This paper provides examples based on military specifications, however there are many similarities between commercial and military requirements. The commercial and civilian requirements are defined by the Federal Aviation Regulations Part 23 and Part 25.

Military Specifications

The primary specifications document for fixed wing military aircraft flight loads is known as *MIL-8861(b) Airplane Strength and Rigidity* [2], which is a Department of Defense document published in February of 1986. It is important to note the date of the latest publication of the spec as it is over thirty years old.

The primary difference between Military specifications and commercial and civilian requirements is the flexibility in requirements and deviations from the MIL-Spec is common, while the commercial requirements are not as flexible. Typically, military aircraft requirements are developed and agreed upon between the acquisition branch of the department of defense and the contractor.

Symmetrical flight load factors are prescribed for each class of aircraft defined in the MIL-Spec [2]. These classifications include: Fighter, Trainer, Attack, Transport, Reconnaissance et. al. The symmetrical flight load envelope is described in the spec as well, shown in Figure 1. The points on the diagram are outlined in the spec, and a special factor, designated k, is used to apply uncertainty in regards to unsteady flow and buffet when the aircraft is at stall.

The theory of quasi-steady flight loads typically does not (directly) account for unsteady flow such as buffet at stall. These values can be confirmed during flight test, or if a program can use wind tunnel testing to accurately predict this flow and corresponding normal force coefficients, the value may be reduced or eliminated. Typically however, critical flight loads tend to come at points not along the stall curve of the Vn diagram.

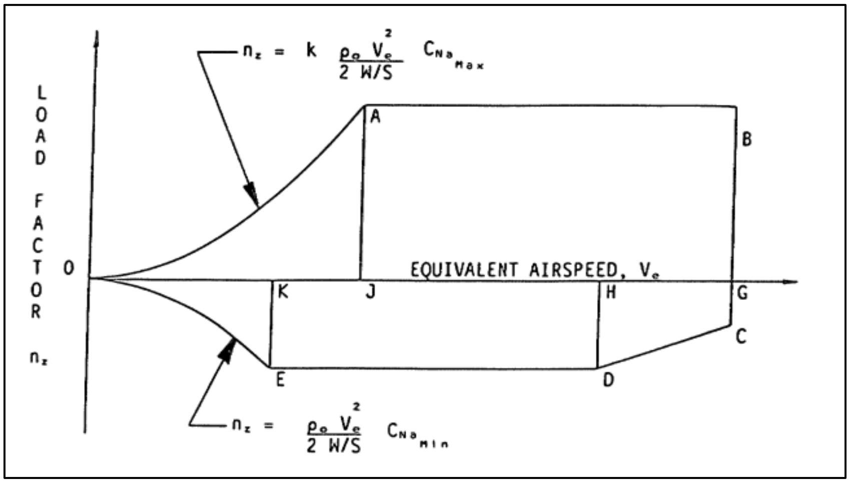


Figure 1 – Vn Diagram for Symmetrical Flight [2]

There are predefined maneuvers in the spec which outline typical flight maneuvers for which the aircraft is designed. Some maneuvers are prescribed for only certain types of aircraft. For example, an accelerated pitch maneuver and recovery is required for all aircraft types, while the level flight roll (360 degree roll) is required only for aircraft of type attack, fighter and trainer.

This paper demonstrates proof of concept for three types of quasi-steady maneuvers prescribed by the MIL-8861(b) Specifications [2]. These maneuver types are accelerated pitch and recovery maneuvers, accelerated roll and recovery maneuvers and abrupt rudder kick maneuvers. All of these are muti-degree of freedom maneuvers. Of course, the same methodology can be appended for three, four and five degree of freedom cases, which include coupled degrees of freedom for maneuvers such as rolling pull-outs (RPO). Three examples of this are also included that consider all 5 degrees of freedom, namely linear accelerations in the Y and Z directions, and roll accelerations about all three principal axes and roll rates about all three principle axes. A more detailed discussion on this is presented in Database Predicted Loads Results.

The three maneuvers are described below in the MIL-8861(b) Specification [2]:

"**3.2.2 Accelerated pitch maneuver and recovery.** The airplane shall be in the basic high-drag, and dive-recovery configurations. The airplane initially shall be in steady unaccelerated flight at the airspeed specified for the maneuver and trimmed for zero control forces at that airspeed. The airspeed shall be constant until the specified load factor has been attained. The load factors to be attained shall be all values on and within the envelope bounded by O, A, B, C, D, and E of Figure 1 (of this document). Except as noted the load factor at each airspeed shall be attained as specified for all center of gravity positions, and also for the maximum-aft center of gravity position, and by a cockpit longitudinal control movement resulting in a triangular displacement-time curve as illustrated by the solid straight lines of Figure 2 (of this document) provided that the specified load factor can be attained by such a control movement; otherwise by the ramp-style control movement illustrated by the dashed straight lines of Figure 2 (of this document) . The time t, is specified in Table I (in reference 2). For the ramp-style control movement, the time t2 shall be the minimum time that the control is held at the stops to attain the specified load factor." [2]

"**3.3.1 Rolling maneuvers**. The airplane shall be in the basic: high-drag and specified store configurations. The airspeeds shall be all airspeeds up to limit speed (V,). During the maneuver, the directional control shall be: a. Held fixed in its position for trim with zero rudder control force in wings-level flight at the speed required, and b. Displaced as necessary to maintain zero sideslip up to limits of the rudder authority. The cockpit lateral control shall be displaced to all the displacements to the maximum available displacement attainable by a pilot lateral control force of 60 pounds (two equal and opposite 48-pound forces applied at the circumference of the control wheel) by application of the control force in not more than 0.1 second for airplanes with stick controls and not more than 0.3 second for airplanes with wheel controls; for automated flight control type systems application of the maximum control surface(s) authority is required. The control force(s) or authority shall be maintained until the required change in angle of bank is attained, except that, if a roll rate greater than 270 degrees per second would result, the control position may be lessened or authority modified, subsequent to attainment of the maximum rolling acceleration, to that position resulting in a roll rate of 270 degrees per second. The maneuver shall be checked by application of the maximum available displacement attainable with a 60-pound lateral control force (two equal and opposite 48-pound forces applied at the circumference of the control wheel) applied in not more than 0.1 second for stick controls and in not more than 0.3 second for wheel controls. For automated flight control type systems, maximum lateral control surface(s) authority shall be used." [2]

"**3.3.3.3 High speed rudder kick**. The airplane shall be in the basic and high-drag configuration at speeds up to V, for VA, VF, and VT airplanes, and up to VH for other type aircraft. The cockpit directional control shall be displaced to the maximum displacement attainable with a 180-pound directional-control force applied in not more than 0.2 second. The control force shall be maintained until the maximum over-swing angle of slide-slip is attained and the airplane attains a steady sideslip. Recovery shall be made by reducing the directional-control displacement to zero in not more than 0.2 second." [2]

The accelerated pitch maneuver and recovery references Figure 2 when stating Figure 3.

These maneuvers were chosen as example maneuvers for this paper due to their dominance on aircraft design sizing. Particular components outlined in this paper are concerned with control surface hinge moment, as there is a direct correlation for the reader to make a connection to loading vs. maneuver. For other structural components, such as integrated fuselage torsion/bending loads, or wing shear/bending loads are not always as directly responsive to these maneuver types.

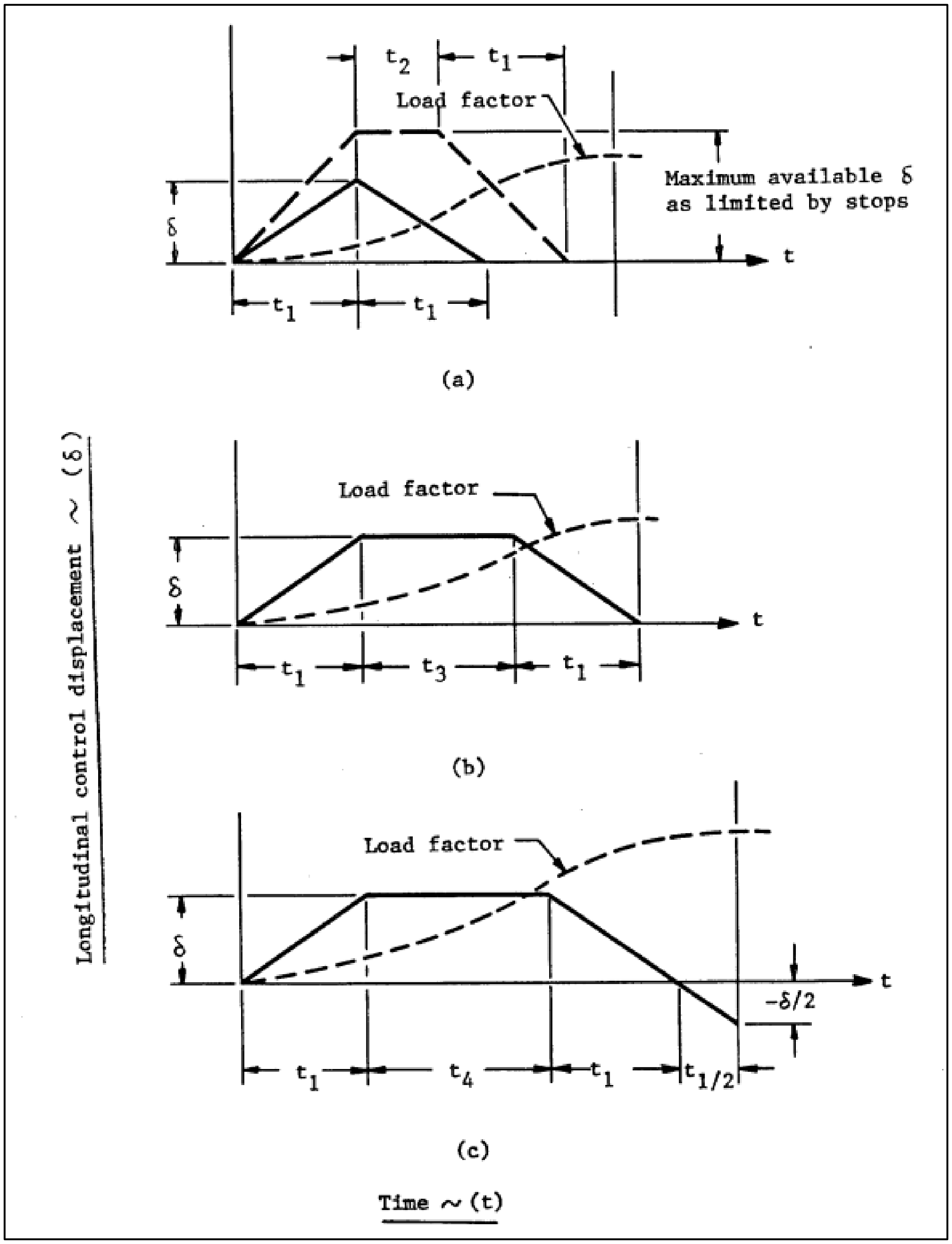


Figure 2 – Longitudinal Control and Load Factor Response for Dynamic Pitching Maneuvers [2]

Aircraft Integration Analysis

Integrated Schedule and Coordination

Perhaps one of the biggest challenges of aircraft design is the integration of the system at the aircraft level. It takes a deep understanding of the interaction of each particular system (including the sciences of flight) to develop an integrated schedule so that the best data available can be used for analysis throughout each of the development phases. While that particular discussion in detail is outside the scope of this thesis, the assumption is that a well planned integrated schedule allows for ideal data to be available to the flight loads analyst through development. This section covers each of the primary disciplines which the analyst needs to be concerned with, and the corresponding data to be expected throughout each of the design phases.

FINITE ELEMENT MODEL (STRUCTURAL ANALYSIS AND DESIGN)

Typically, the customer of the flight loads team is the structural finite element modeling (FEM) team, and this paper will assume that. The FEM team is responsible for, among other things, developing a FEM of the vehicle so that detail analysis and design can be completed, aircraft flexibility can be assessed and may even aid in the development of parametric mass properties. Just as the fidelity of the analysis of the flight loads team tends to increase as the program progresses, so does the finite element model. It is important that during the various phases of development, the stiffness of the global vehicle is tracked. The benefit of this tracking is predicting impacts on flight loads as a function of the effects of static aeroelasticity, which is the effect on the aerodynamic loading due to the deformations of the structure. Typically, as a structure becomes stiffer, the aerodynamic loads will increase. This will be evident as the flexible to rigid aircraft derivatives are presented in this paper.

The FEM is both an input and the end product of the flight loads analysis as the model will have the “critical” loads applied to it so stress analysis can be completed.

FLIGHT CONTROLS

With the advent of fly-by-wire and even direct hydraulic systems, the job of the flight loads analyst becomes increasingly difficult due to the quick and sometimes hard to recognize changes in the flight control system of the vehicle which may have significant effects on the flying characteristics of the vehicle. These changes can be strictly at the software level, where feedback and control gains are modified to meet flying quality standards or may change as aerodynamic data changes or mission requirements change. Typically, flight control development lags behind structural design due to the nature of the schedule of the aircraft development program. It is necessary for the analyst to understand the impacts of potential change and carry relatively significant margin as to protect the vehicle from under-design.

This paper focuses on various assumptions that the analyst will need to make through the different portions of the aircraft development phases. Depending on the program, maximum maneuvering capability may be defined which restrict both the structural design and flight controls. While these values may be good values for initial sizing, if any margin can be reduced by time accurate aircraft state data, i.e. transient time history data, it is desired that these values be used, such that the risk of the outcome is not increased due to potential flight controls updates. In other words, the vehicle should not be overdesigned to maximum maneuvering capability at every point in the sky if reliable time histories are available to the analyst. It is up to the analyst to carry margin in the analysis if this is in fact the case, in addition to the factor of safety.

For the conceptual design phases of the program, it is assumed that no aircraft maneuvering requirements are set, and that general kinematics should be studied. In the later phases of the preliminary design and certainly during detail design, more accurate aircraft flying qualities are set and aircraft maneuvering should be well defined. It is at these stages where the analyst must consider particular data sets for the quasi-steady analysis, whether it be time slices from time accurate data, tabulated maximum rates and accelerations as defined by aircraft control laws, or some other form. The methods selected, however, have no change to the method for developing the loads.

If the usage of the time accurate six degree of freedom simulation is utilized, however, it would be of interest to integrate a method for calculating loads at every time point at specific monitor points of interest. This is important due to the load lag during aircraft maneuvering.

MASS PROPERTIES

The mass properties team is integral in tracking and tabulating aircraft mass properties, both on the total aircraft and detail part level. Through the stages of aircraft development, the aircraft sizing will vary due to the loads that are produced to design the vehicle, along with the development of the stiffness of the FEM. Therefore, the mass properties team will have to track that development. As the program progresses it is necessary to capture updates as mass properties change from parametric and predicted weights to actual weights once the drawing has been released.

Mass properties will also have data involving requirements for payload, such as passengers and cargo for commercial application or weapons and cargo for military applications. The distribution of these payload items can be assumed to have significant impact on the trim of the vehicle and the resulting loads.

Both the total aircraft mass properties and the distribution of mass are important to developing a reliable aeroelastic model. Since the aeroelastic effects are produced from the mass and stiffness matrices, the data from the mass properties team is just as important as the data from the FEM team, i.e. the mass and the stiffness data.

The modeling techniques for the mass distribution may vary throughout the program development. There are many methods which the mass can be represented on the structural finite element model. Within NASTRAN, there are methods of defining material densities, which along with the volumetric properties of the elements will define the mass properties. This method should be used with caution, as the data may not always (and probably won't) sum back to the total aircraft numbers defined by the mass properties team. Other methods include the use of lumped masses. The refinement of the lumped masses is up to the analyst to capture distribution and accurate mode shapes without inducing fictitious modes.

PROPULSION

The integration of the propulsion system into the aeroelastic model can be one of the more challenging disciplines. The propulsion system can have both effects on the total vehicle, and local effects due to inlet and exhaust pressure differentials.

The total vehicle effects of the propulsion system are any off axis thrust terms which cause aircraft pitching moment or loads along the aircraft body axis.

The example used in this paper neglects effects due to propulsion, however some aircraft may have considerable effects on trim due to the propulsion system. Some may be as complex as aircraft such as the Lockheed F-35 which has thrust vectoring, or as simple as the pitching moment due to the low hung wings of a Boeing 737. Ducted inlets, such as those on fighters can cause accelerated flow on the surrounded surfaces which will vary the nominal aerodynamics without propulsive effects.

AERODYNAMICS

There are many methods for developing aerodynamics of the vehicle. The term aerodynamics includes both total aircraft aerodynamics, such as those obtained from force balance data from a wind tunnel test, i.e. aircraft pitching and lift coefficients, as well as control surface and door hinge moments and distributed pressure coefficients such as those obtained from computational fluid dynamics (CFD) and pressure sensitive paint wind tunnel tests.

Typically, wind tunnel testing tends to be very expensive and may only occur over certain flight regimes. The point at which wind tunnel testing occurs for a program can vary, however, it can be assumed that during the initial trade studies, no wind tunnel testing exists.

CFD data is typically limited at initial stages of the design phase. The loads analyst should work with the CFD analyst to obtain solutions that will be used as the design refinement goes forward, so that the doublet-lattice methods can be replaced by a complete CFD database of distributed pressure solutions. Note, these CFD results obtained from high order methods such as Euler and Navier-Stokes will be rigid aerodynamics and do not account for flexibility. It will be up to the loads analyst to take these results and correct them for aeroelastic effects using NASTRAN. These methods will be described in more detail.

INTEGRATION

It can easily be seen how interactive each of these disciplines are with each other. The structural design analysts will use the loads developed to size parts. The designed/sized parts will have weights, which the mass properties team will tabulate. These tabulated values are fed into the flight controls analysis to tune gains for flying qualities. The resulting flying quality values are based on aircraft coefficients developed from the aerodynamics team. The fallout aircraft maneuvering capability is fed back into the loads analysis along with mass properties inertial distributions, the FEM aircraft stiffness (based on the aircraft part sizing) and the propulsion system effects.

It is absolutely critical that each of these disciplines are well integrated for the success of the aircraft loads assessments as the process is circular, in that the cycle of analysis is lagged by n-1 for each discipline. Take for example analysis cycle 2 for the aircraft quasi-steady aeroelastic loads. The data the loads analyst will use is based on the stiffness of the finite element model from the previous sizing effort, along with some aircraft maneuvering capability based on the inertia properties from the previous sizing effort. This means that the loads assessment is performed based on the result from the previous loads assessment and the impacts are analyzed on the sizing from the previous assessment.

It is crucial that this inherent lag in design analysis is understood and captured in the loads analysis. The methods described in this paper allow for easy transition of updated data to be positioned into the same database that was previously used in the previous assessment of loads.

Moreover, trade studies and impact analysis can be conducted based on the previously developed database for trends of increased or decrease stiffness and inertial distribution. For example, typically, an increase in wing stiffness tends to lead to increased wing bending loads due to an inherent outboard shift in the center of pressure due to wing flex. A scaled reduction in derivative stiffness in the database can give a quick back of the envelope investigation into wing stiffness changes. Similar studies can be conducted based on the inertial distribution, and maneuvering capability changes. Small mass property changes can be assumed to have no impact on the mode shapes of the vehicle but may have an impact on the inertial distribution and inherent inertia relief. This assessment can be conducted by scaling the inertia relief to mimic what is expected. Similarly, for changes in aircraft maneuvering rates and accelerations, a quick study on the previously defined critical conditions with the existing database at new rates and accelerations can show incremental impacts due to the increases or decrease in aircraft maneuvering capability.

NASTRAN Study Model

NASTRAN Static Aeroelastics

The method used to develop the flexible loads database is derived from solutions of the NASTRAN Solution 144 module, which is the static aeroelastics module in NASTRAN. The method uses "a new stiffness formulation [which] has been developed and added to MSC/NASTRAN to solve the basic trim load problem and estimated aeroelastic stability derivatives at subsonic speeds." [5] There are two primary purposes of NASTRAN Solution 144. These are the study of "Static Aeroelastic Response" and "Aeroelastic Divergence." [5] While the methods described in this paper can aid in the evaluation of static aeroelastic divergence (such as aileron reversal), the intent of this paper focuses on the former, "Static Aeroelastic Response", namely the response of the structural vehicle to the inertial and aerodynamic loading produced at various flight conditions.

This method produces a coupled solution of the structural stiffness response to the quasi-steady aerodynamic loading. It is a method of splines which creates interpolated solution sets on corresponding grid points between the aerodynamic model and the structural model. This coupling allows for structural monitor points and corresponding derivatives for the monitor points.

There are two methods of splines in NASTRAN. Both of which are inherently "two dimensional". They assume that the forces on the coupled models are normal to one or two orthogonal planes and that the aerodynamic geometry consists of collections of points on a plane or along an axis. The two methods are the Harder Desmarais infinite plate spline, also known as SPLINE1 in NASTRAN, and the beam spline, also known as SPLINE2 in

NASTRAN. The spline methods used in the example in this paper are SPLINE2.

1	2	3	4	5	6	7	8	9	10
SPLINE2	EID	CAERO	ID1	ID2	SETG	DZ	DTOR	CID	
	DTHX	DTHY		USAGE					

Figure 3 SPLINE2 Bulk Data Entry

SPLINE2 – Denotes the call in NASTRAN for the spline method in the following bulk data entry

EID – Defines the element ID

CAERO – Interpolated aerodynamic panel ID

ID1/ID2 – First and last box or body element to be interpolated on

SETG – The set of inertia grids to interpolate the aerodynamic forces to

DZ – The linear attachment flexibility.

DTOR – The torsional flexibility ratio = EI/GJ.

CID – The coordinate system (rectangular) for which the y-axis defines the spline.

DTHX/DTHY – Rotational attachment flexibility.

USAGE – Input to define whether force, displacement or both methods are used.

The process of the aeroelastic solution can be formulated into distinct portions. These portions are aerodynamic stability derivatives, aerodynamic pressure distributions (including aeroelastic effects), static trim solutions, external discreet loading, and control surface hinge-moments. These solutions are part of the family of aerodynamic solutions.

Additionally, internal freebody loads, deflections and element stresses can also be calculated as parts of the structural solutions.

For the subsonic solution using CAERO1 and CAERO2, aerodynamic and body elements, respectively, the doublet lattice method is used. Because no license for supersonic flow was available (requires ZONA51 Panel License) all analysis subsonic element with the aforementioned element types. For the purpose of this study, however, that is sufficient as the database generation is not limited in any way by Mach number, and therefore, supersonic flow is in fact unnecessary. However, supersonic analysis is a possible extension of this paper.

NASTRAN solution sequence 144 has multiple methods of deriving aerodynamic stability derivatives. These methods include rigid, unrestrained flexible and restrained flexible. This paper is interested in the unrestrained flexible solutions, which will be what is presented. These solutions allow for the inclusion of inertial effects to be used in the calculation of the aerodynamic stability derivatives, so that trim solutions can be predicted, after an initial set of stability derivatives is developed.

HA-144F NASTRAN Model

The model used for this research is one of the common NASTRAN Aeroelastic study models. Aeroelastic models in NASTRAN consist of two independent finite element model types which are then coupled through the use of splines. One of the models, which is referred to as the structural model, contains classical NASTRAN structural elements, i.e. CROD, CBAR, CQUAD4, etc., while the aerodynamic model contains non-structural elements such as CAERO1 and CAERO2 panels.

The aerodynamic portion of the model is made up strictly of CAERO1 and CAERO2 panels. CAERO1 panels are aerodynamic panels that represent un-interfered panels. CAERO2 panels are aerodynamic panels used to represent wing-body interaction.

The aerodynamic reference data for the model uses a reference area of 400 [ft^2], a reference span of 100 [ft] and a reference chord of 10 [ft].

The structural model, or the portion of the model which contains the stiffness and mass properties, is shown in Figure 4.

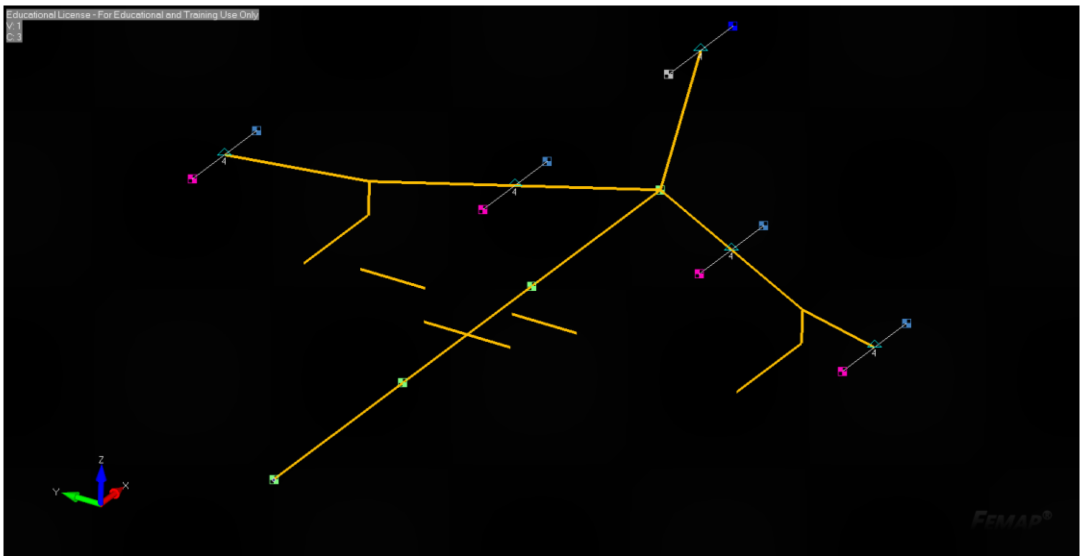


Figure 4 NASTRAN Structural Model

The model consists of 19 bar elements (CBAR), 14 concentrated mass elements (CONM2), and 10 RBE3 rigid elements. The 19 bar elements are colored yellow in the model shown in Figure 4. Some CONM2 elements are shown in pink, light blue and dark blue. They are held at the poles of the rigid RBE3 elements which are colored white. The rest of the CONM2s are colored green along the fuselage. Notice, the mass elements are lumped and are not located on the control surfaces themselves but along the hinge line. The underhanging pylons also have no mass attached to them.

It would be expected that further along in aircraft development, this model will become more refined with two dimensional elements such as CQUAD4s and CTRIA3s along with beam, rod and bar elements. The inertial distribution should increase significantly as well.

The quick look at aircraft mode shapes can be a good indication on the aeroelastic effects of the vehicle. As the vehicle FEM and inertial properties change, identifying key mode shapes can be critical in predicted aeroelastic effects as the design matures. The first three mode shapes (really these are the 7^{th}, 8^{th} and 9^{th} mode shapes, given 6 unrestrained rigid body modes), which are shown in Figure 5, Figure 6 and Figure 7. They occur at 7.56 Hz, 9.79Hz and 18.18Hz, respectively. The first mode is a fuselage yawing mode. The second is a combined fuselage bending and symmetric wing bending mode. The third is second combined fuselage bending and symmetric wing bending mode.

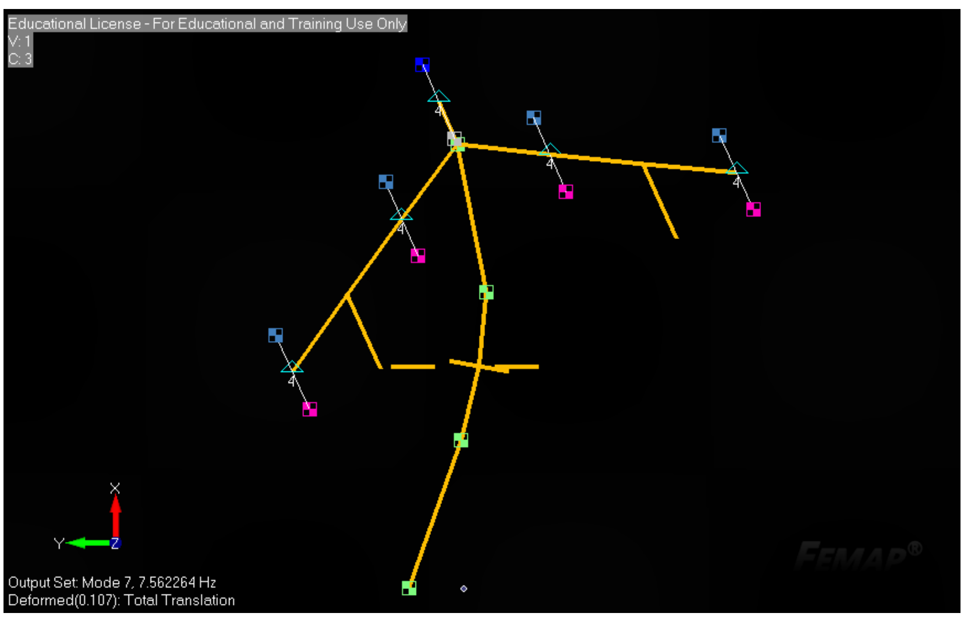


Figure 5 First Mode - Fuselage Yawing at 7.56Hz

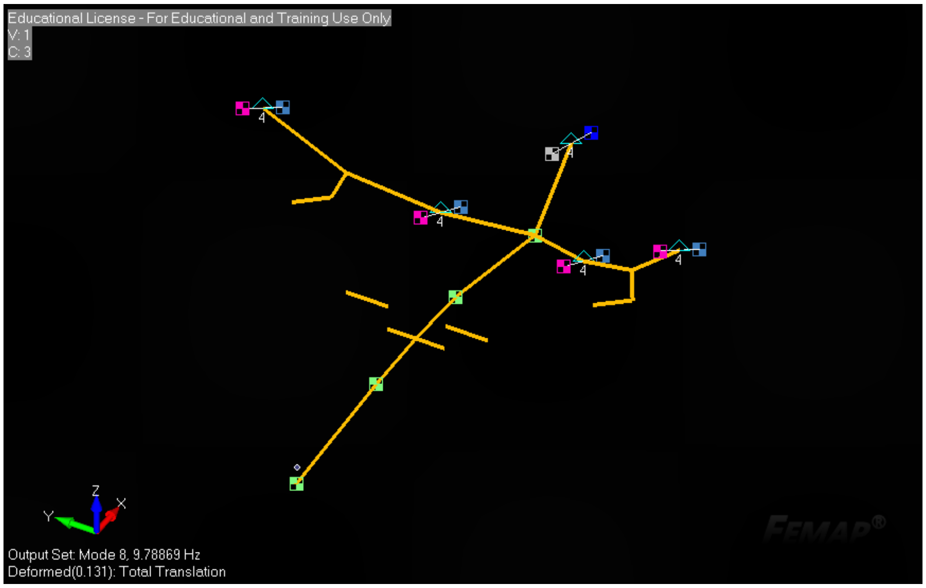


Figure 6 Second Mode - First Fuselage and Wing Bending Mode at 9.79Hz

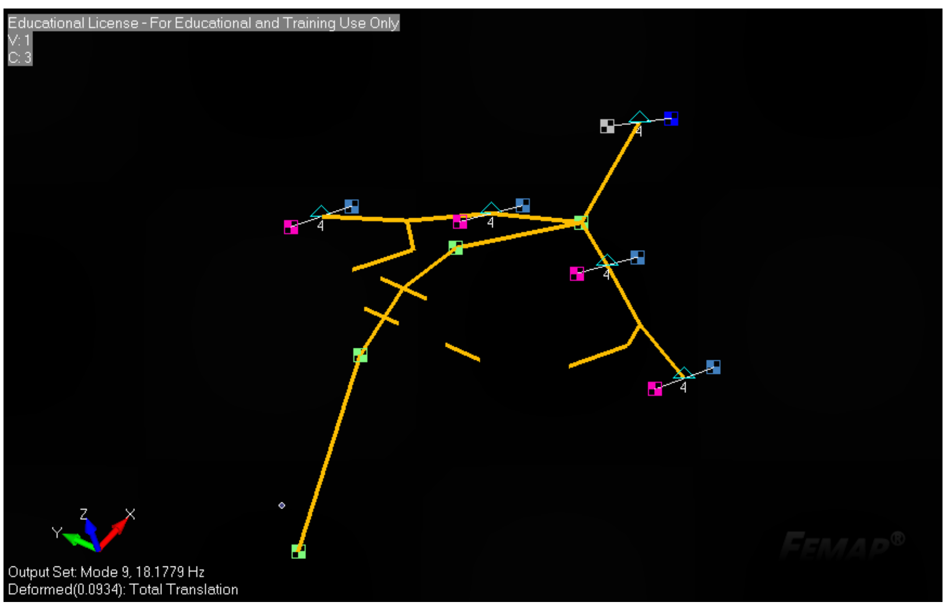


Figure 7 Third Mode - Second Fuselage Bending and Wing Bending at 18.78Hz

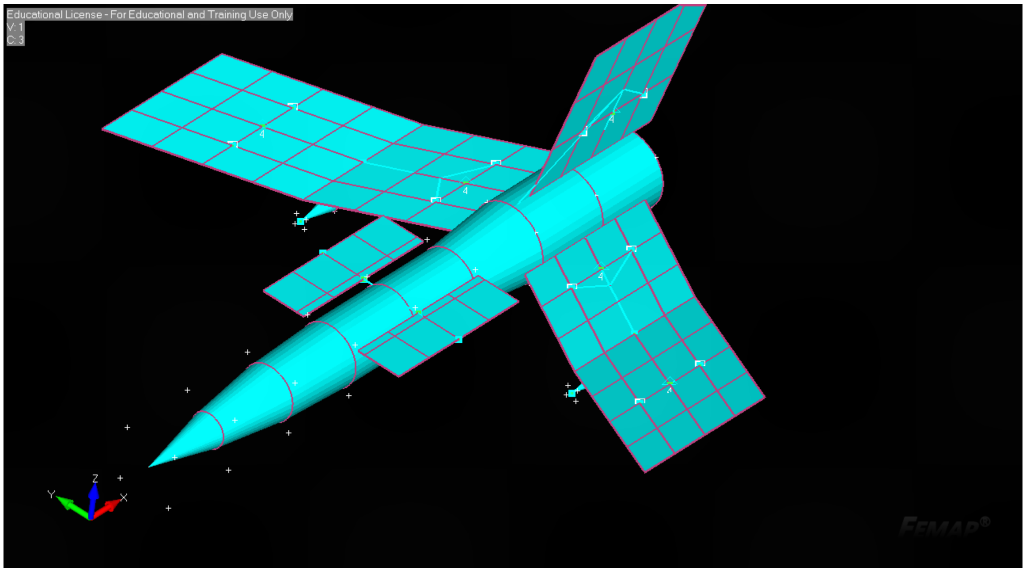


Figure 8 NASTRAN Aero Model Mesh with Structural Model

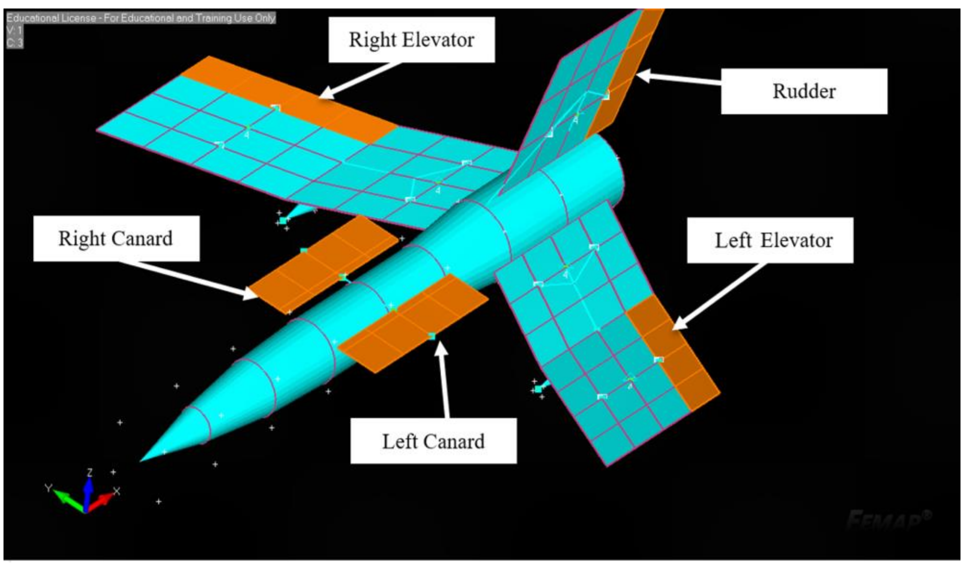


Figure 9 NASTRAN Aero Model with Control Surfaces and Structural Model

The NASTRAN Aero Model and the NASTRAN Structural Model on connected through a spline, or interpolation method which "connects" specific grid points of both models to transfer aerodynamic loading to the structural model and provide aircraft stiffness and mass properties to the aero model for aeroelastic effects. These combined models can be seen in Figure 8 and the modeled control surfaces can be seen in Figure 9.

The bulk data file input for the spline is shown in Appendix M
HA-144F Model Data. For reference, the bulk data entry format can be seen in Figure 3.

The splines used were the same that were existing on the NASTRAN HA144F model and no modifications were made. The aerodynamic modeling and the structural modeling are also consistent with what the initial example had defined, and no modifications were made. The details of the modeling and the exact model itself are independent of the procedure described.

In theory, the methods described in this paper are able to be used on any fixed wing aircraft type, including forward swept wings, wings with positive and negative dihedral, V-tail aircraft, variable sweep aircraft, traditional swept wings, high wing, low wing, canard, blended wing body, and even flying wing body. Basically, if the aircraft can fly, it can be modeled with these methods.

The HA 144F model was selected due to its particularly unique properties and most notably the forward swept wings and canards. Forward swept wings tend to have issues with static aeroelasticity, namely divergence. The particulars of that discussion are outside the scope of this paper; however, the flexible effects of wing torsion can be seen in Appendix D

Flexible vs. Rigid Stability Derivatives. The aircraft has a higher d_{Cz}/d_{α} value when flexibility is considered. This is due to the fact that as the aircraft increases in alpha, the wing tends to torque to increases alpha, which in some cases can lead to divergence issues. Portions of the processes described in this paper can be used to set requirements for wing torsion stiffness due to static aeroelastic divergence. One of the most well-known examples of this type of aircraft was the Grumman X-29 demonstrator.

It is interesting to note the first mode shape of the vehicle. The first mode is a fuselage yawing mode which is typically uncharacteristic of flight vehicles. This indicates that perhaps the fuselage design needs to be stiffer. Typically, the first modes are wing bending or torsion modes before fuselage modes. Studies of the mode shapes can help identify static aeroelastic effects due to stiffness properties of the vehicle.

Database Generation Process

Static Aeroelastic Trim Theory

The theory for the development of the database generation used in this thesis is reliant on linear assumptions. The doublet-lattice model used in the aerodynamic generation (if chosen) assumes a constant stability derivative as a function of Mach only. Therefore, the linearity of the control derivatives is assumed across all variations of the control variables, at a given Mach. This theory, however, though limited in direct application can be used to capture non-linear effects. This can be accomplished via a piecewise-linear database generation. This method is shown in the way the database is generated, piecewise linear, in Mach, neglecting non-linearity in any other terms, such as stall effects at extreme alphas, or betas. Additional non-linearities can occur due to downwash effects, propulsive effects, etc. It is up to the analyst to understand these non-linearities and develop the breakpoints of the database to capture them.

First, a database of aerodynamic stability derivatives must be developed. These stability derivatives are a function of the control variables used in the trim analysis. These control variables include alpha and beta, which are the relative flow angles, and control surface deflections, i.e. aileron, elevator and rudder. Additionally, set parameters which need to be considered are roll, pitch and yaw rates. Neglected aerodynamic effects are acceleration terms, which describe the inertial loading.

A summary of the resulting rigid database can be seen in Appendix A
Tabulated Rigid Stability Derivatives at Incremental Mach Values and the corresponding flexible database can be seen in Appendix B
Tabulated Flexible Stability Derivatives at Incremental Mach Values.

Note, the aerodynamic stability derivatives in X are always zero. This is because the database assumes only 5 degrees of freedom for trim, Y, Z, MX, MY, and MZ. In order to satisfy the determined system for static aeroelastic trim, the number of free variables must equal the number of degrees of freedom for the equation of motion. In other words, there must be n-number of unknowns to be solved for, given n-number of equations. The solutions being sought are in the form **Ax=B** where **B** is the inertial loading, **A** is the dimensionalized stability derivatives and x is the state vector of the free and prescribed variables.

It is common to neglect the drag term for two particular reasons. The first is that the doublet-lattice method does not consider viscous drag, which can only be predicted through higher order CFD methods such as Navier-Stokes. More importantly, though, is that the aircraft body axis is always assumed to be aligned to the aircraft inertial axis, and that the aircraft can be in the same state whether thrust is at maximum power or it is at idle. For example, in flight test a pilot may be steady level and pull back abruptly on the stick at full throttle. Similarly, the pilot can be at the same airspeed and same altitude at idle power by diving into the maneuver from a higher altitude, so that the aircraft has state data that is (exactly) the same except for the thrust setting for that maneuver.

In the following derivations, roll, pitch and yaw rates are represented by the letters “p”, “q” and “r”, respectively and their time derivatives, or accelerations are represented with dot notation. The true airspeed is represented by “V’. The aircraft body forces are represented by “X”, “Y” and “Z” and the aircraft body moments are represented by “L”, “M”, and “N”, each representing the 3 forces and moments on the principle body axis, in the order of roll, pitch and yaw. The relative flow angles of angle of attack and sideslip angles are represented by “α” and “β”, respectively, and the control surface deflections are represented by “δ”.

The general form of the aerodynamic loading can be represented as

Equation 1 CY Aero Loading

$$C_{Y_{aero}} = C_{Y_\alpha}\alpha + C_{Y_\beta}\beta + C_{Y_{\delta_a}}\delta_a + C_{Y_{\delta_e}}\delta_e + C_{Y_{\delta_r}}\delta_r + C_{Y_p}\frac{p\bar{b}}{2V} + C_{Y_q}\frac{q\bar{c}}{2V} + C_{Y_r}\frac{r\bar{b}}{2V}$$

and

Equation 2 CZ Aero Loading

$$C_{Z_{aero}} = C_{Z_\alpha}\alpha + C_{Z_\beta}\beta + C_{Z_{\delta_a}}\delta_a + C_{Z_{\delta_e}}\delta_e + C_{Z_{\delta_r}}\delta_r + C_{Z_p}\frac{p\bar{b}}{2V} + C_{Z_q}\frac{q\bar{c}}{2V} + C_{Z_r}\frac{r\bar{b}}{2V}$$

The terms can be combined and cast into matrix form yielding:

Equation 3 General Matrix Aerodynamic Loading

$$\begin{Bmatrix} \dfrac{Y}{\bar{q}S} \\ \dfrac{Z}{\bar{q}S} \end{Bmatrix} = \begin{bmatrix} C_{Y_\alpha} & C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_e}} & C_{Y_{\delta_r}} & C_{Y_p} & C_{Y_q} & C_{Y_r} \\ C_{Z_\alpha} & C_{Z_\beta} & C_{Z_{\delta_a}} & C_{Z_{\delta_e}} & C_{Z_{\delta_r}} & C_{Z_p} & C_{Z_q} & C_{Z_r} \end{bmatrix} \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \\ \dfrac{p\bar{b}}{2V} \\ \dfrac{q\bar{c}}{2V} \\ \dfrac{r\bar{b}}{2V} \end{Bmatrix}$$

Each of the stability derivatives are non-dimensionalized by the freestream dynamic pressure, $\bar{q}$, and the reference area, S. For trim, it is assumed that the free trim variables, or variables used to trim the aircraft are α, β, and control surface deflection only. It is convenient to rewrite Equation 3 in the form of a linear combination of free variables and prescribed variables, where the prescribed variables are the aircraft rotational rates. These rotational rates contribute to the aircraft roll, pitch and yaw damping terms. In some cases, these terms may be negligible for inertial effects, but must be included for aerodynamic effects. The roll, pitch and yaw rates induce local flow angles of alpha and beta on the panels, resulting in local reductions or increases of aerodynamic pressures and may, such as in the case of this example, be significant due to the high rates (>100 deg/s) of a fighter type aircraft.

Equation 4 Modified Coefficient Aerodynamic Loading g

$$\begin{Bmatrix} \frac{Y}{\bar{q}S} \\ \frac{Z}{\bar{q}S} \end{Bmatrix} = \begin{bmatrix} C_{Y_\alpha} & C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_e}} & C_{Y_{\delta_r}} \\ C_{Z_\alpha} & C_{Z_\beta} & C_{Z_{\delta_a}} & C_{Z_{\delta_e}} & C_{Z_{\delta_r}} \end{bmatrix} \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \end{Bmatrix} + \begin{bmatrix} C_{Y_p} & C_{Y_q} & C_{Y_r} \\ C_{Z_p} & C_{Z_q} & C_{Z_r} \end{bmatrix} \begin{Bmatrix} \frac{p\bar{b}}{2V} \\ \frac{q\bar{c}}{2V} \\ \frac{r\bar{b}}{2V} \end{Bmatrix}$$

Since the aircraft rates are prescribed of roll rate, p, pitch rate, q and yaw rate, r, there are three possible conditions that will exist; either the terms will be non-zero, zero or some combination of both. The system can be solved for directly in either scenario. Therefore, the second term on the RHS of the equation can be moved to the LHS of the equation.

Equation 5 Modified Coefficient Aerodyanamic Loading, Ax = B form

$$\begin{Bmatrix} \frac{Y_A}{\bar{q}S} \\ \frac{Z_A}{\bar{q}S} \end{Bmatrix} - \begin{bmatrix} C_{Y_p} & C_{Y_q} & C_{Y_r} \\ C_{Z_p} & C_{Z_q} & C_{Z_r} \end{bmatrix} \begin{Bmatrix} \frac{p\bar{b}}{2V} \\ \frac{q\bar{c}}{2V} \\ \frac{r\bar{b}}{2V} \end{Bmatrix} = \begin{bmatrix} C_{Y_\alpha} & C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_e}} & C_{Y_{\delta_r}} \\ C_{Z_\alpha} & C_{Z_\beta} & C_{Z_{\delta_a}} & C_{Z_{\delta_e}} & C_{Z_{\delta_r}} \end{bmatrix} \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \end{Bmatrix}$$

Similarly, the aircraft roll, pitch and yawing moments can be expressed:

Equation 6 Aeodynamic Moment Coefficients, Matrix Form

$$\begin{Bmatrix} \frac{L_A}{\bar{q}S\bar{b}} \\ \frac{M_A}{\bar{q}S\bar{b}} \\ \frac{N_A}{\bar{q}S\bar{c}} \end{Bmatrix} - \begin{bmatrix} C_{L_p} & C_{L_q} & C_{L_r} \\ C_{M_p} & C_{M_q} & C_{M_r} \\ C_{N_p} & C_{N_q} & C_{N_r} \end{bmatrix} \begin{Bmatrix} \frac{p\bar{b}}{2V} \\ \frac{q\bar{c}}{2V} \\ \frac{r\bar{b}}{2V} \end{Bmatrix} = \begin{bmatrix} C_{L_\alpha} & C_{L_\beta} & C_{L_{\delta_a}} & C_{L_{\delta_e}} & C_{L_{\delta_r}} \\ C_{M_\alpha} & C_{M_\beta} & C_{M_{\delta_a}} & C_{M_{\delta_e}} & C_{M_{\delta_r}} \\ C_{N_\alpha} & C_{N_\beta} & C_{N_{\delta_a}} & C_{N_{\delta_e}} & C_{N_{\delta_r}} \end{bmatrix} \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \end{Bmatrix}$$

Attention is turned towards the inertial portion of the problem to create a system of equations to solve for the statically determinant trim analysis.

It is assumed that the aircraft inertial loading has small deformations such that Euler rigid body motion can used, which is expressed as:

Equation 7 Inertial Linear Acceleration Loading

$$\begin{bmatrix} X_I \\ Y_I \\ Z_I \end{bmatrix} = W\left[N_x\ N_y\ N_z\right] = W[\ 0\ \ N_y\ N_z]$$

and

Equation 8 Euler Rigid Body Mechanics

$$\begin{bmatrix} L_I \\ M_I \\ N_I \end{bmatrix} = \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} \begin{Bmatrix} \dot{p} \\ \dot{q} \\ \dot{r} \end{Bmatrix} + \begin{Bmatrix} p \\ q \\ r \end{Bmatrix} \times \begin{bmatrix} I_{xx} & I_{xy} & I_{xz} \\ I_{yx} & I_{yy} & I_{yz} \\ I_{zx} & I_{zy} & I_{zz} \end{bmatrix} \begin{Bmatrix} p \\ q \\ r \end{Bmatrix}$$

The rigid body moments can be expanded:

Equation 9 Euler Rigid Body Rolling Moments Expanded

$$L_I = I_{xx}\dot{p} - I_{yz}(q^2 - r^2) - I_{zx}(\dot{r} + pq) - I_{xy}(\dot{q} - rp) - \left(I_{yy} - I_{zz}\right)qr$$

Equation 10 Euler Rigid Body Pitching Moments Expanded

$$M_I = I_{yy}\dot{q} - I_{zx}(r^2 - p^2) - I_{xy}(\dot{p} + qr) - I_{yz}(\dot{r} - pq) - (I_{zz} - I_{xx})rp$$

Equation 11 Euler Rigid Body Yawing Moments Expanded

$$N_I = I_{zz}\dot{r} - I_{xy}(p^2 - q^2) - I_{yz}(\dot{q} + rp) - I_{zx}(\dot{p} - qr) - \left(I_{xx} - I_{yy}\right)pq$$

If a symmetric aircraft (about the XZ-plane) is assumed, the inertia matrix reduces to:

Equation 12 Inertia Matrix

$$\boldsymbol{I} = \begin{bmatrix} I_{xx} & 0 & I_{xz} \\ 0 & I_{yy} & 0 \\ I_{zx} & 0 & I_{zz} \end{bmatrix}$$

Equation 13 Symmetric Euler Rigid Body Pitching Moments

$$\boldsymbol{L_I = I_{xx}\dot{p} - I_{zx}(\dot{r} + pq) - \left(I_{yy} - I_{zz}\right)qr}$$

Equation 14 Symmetric Euler Rigid Body Rolling Moments

$$M_I = I_{yy}\dot{q} - I_{zx}(r^2 - p^2) - (I_{zz} - I_{xx})rp$$

Equation 15 Symmetric Euler Rigid Body Yawing Moments

$$N_I = I_{zz}\dot{r} - I_{zx}(\dot{p} - qr) - \left(I_{xx} - I_{yy}\right)pq$$

Equation 13 through Equation 15 is the case for the HA144F model and is typically a good approximation for most practical aircraft analysis.

To satisfy static equilibrium, the inertial forces and moments plus the aerodynamic forces and moments must sum to zero. Therefore, the relationship can be made such that (assuming X_I= 0 and X_A = 0 always),

Equation 16 Aircraft Steady State Balanced Condition

$$\begin{bmatrix} Y_I \\ Z_I \\ L_I \\ M_I \\ N_I \end{bmatrix} = -\begin{bmatrix} Y_A \\ Z_A \\ L_A \\ M_A \\ N_A \end{bmatrix}$$

Substituting in the inertia terms via the relationship of inertial and aerodynamic loading, the combined equation becomes:

Equation 17 General Control Trim Solution for Balance Flight Maneuver

$$-\begin{Bmatrix} \frac{Y_I}{\bar{q}S} \\ \frac{Z_I}{\bar{q}S} \\ \frac{L_I}{\bar{q}S\bar{b}} \\ \frac{M_I}{\bar{q}S\bar{b}} \\ \frac{N_I}{\bar{q}S\bar{c}} \end{Bmatrix} - \begin{bmatrix} C_{Y_p} & C_{Y_q} & C_{Y_r} \\ C_{Z_p} & C_{Z_q} & C_{Z_r} \\ C_{L_p} & C_{L_q} & C_{L_r} \\ C_{M_p} & C_{M_q} & C_{M_r} \\ C_{N_p} & C_{N_q} & C_{N_r} \end{bmatrix} \begin{Bmatrix} \frac{p\bar{b}}{2V} \\ \frac{q\bar{c}}{2V} \\ \frac{r\bar{b}}{2V} \end{Bmatrix} = \begin{bmatrix} C_{Y_\alpha} & C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_e}} & C_{Y_{\delta_r}} \\ C_{Z_\alpha} & C_{Z_\beta} & C_{Z_{\delta_a}} & C_{Z_{\delta_e}} & C_{Z_{\delta_r}} \\ C_{L_\alpha} & C_{L_\beta} & C_{L_{\delta_a}} & C_{L_{\delta_e}} & C_{L_{\delta_r}} \\ C_{M_\alpha} & C_{M_\beta} & C_{M_{\delta_a}} & C_{M_{\delta_e}} & C_{M_{\delta_r}} \\ C_{N_\alpha} & C_{N_\beta} & C_{N_{\delta_a}} & C_{N_{\delta_e}} & C_{N_{\delta_r}} \end{bmatrix} \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \end{Bmatrix}$$

This equation is of the form **B** = **A**x, where:

$$\boldsymbol{B} = -\begin{Bmatrix} \frac{Y_I}{\bar{q}S} \\ \frac{Z_I}{\bar{q}S} \\ \frac{L_I}{\bar{q}S\bar{b}} \\ \frac{M_I}{\bar{q}S\bar{b}} \\ \frac{N_I}{\bar{q}S\bar{c}} \end{Bmatrix} - \begin{bmatrix} C_{Y_p} & C_{Y_q} & C_{Y_r} \\ C_{Z_p} & C_{Z_q} & C_{Z_r} \\ C_{L_p} & C_{L_q} & C_{L_r} \\ C_{M_p} & C_{M_q} & C_{M_r} \\ C_{N_p} & C_{N_q} & C_{N_r} \end{bmatrix} \begin{Bmatrix} \frac{p\bar{b}}{2V} \\ \frac{q\bar{c}}{2V} \\ \frac{r\bar{b}}{2V} \end{Bmatrix},$$

$$\mathbf{A} = \begin{bmatrix} C_{Y_\alpha} & C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_e}} & C_{Y_{\delta_r}} \\ C_{Z_\alpha} & C_{Z_\beta} & C_{Z_{\delta_a}} & C_{Z_{\delta_e}} & C_{Z_{\delta_r}} \\ C_{L_\alpha} & C_{L_\beta} & C_{L_{\delta_a}} & C_{L_{\delta_e}} & C_{L_{\delta_r}} \\ C_{M_\alpha} & C_{M_\beta} & C_{M_{\delta_a}} & C_{M_{\delta_e}} & C_{M_{\delta_r}} \\ C_{N_\alpha} & C_{N_\beta} & C_{N_{\delta_a}} & C_{N_{\delta_e}} & C_{N_{\delta_r}} \end{bmatrix},$$

and the state vector,

$$x = \begin{Bmatrix} \alpha \\ \beta \\ \delta_a \\ \delta_e \\ \delta_r \end{Bmatrix}$$

Where **B** and **A** are known, x can be found. The vector, x is the solution to the quasi-steady trim solution.

There are only three control surfaces defined in this solution, however, more, and in fact infinite number of surfaces or external control devices can be defined for the aircraft. For example, thrust vectoring may be another consideration. In such a case, the **A** matrix would change from a 5x5 matrix to a 5x6 matrix, where the additional terms would be derivatives of forces and moments with respect to the thrust vectoring device and some variable. It is up to the analyst to determine appropriate development of this matrix and include these in the model.

It should be noted that while there are 5 linear equations, they are in fact not independent equations but the respective position of the trim variable has no bearing on the corresponding degree of freedom. For example, because α and side force, Y are the first terms of the state and output vector, the side-force is not necessarily dependent on alpha alone. In fact, alpha has zero influence on that degree of freedom (in the presented example).

The solution to the quasi-steady trim is necessary for "re-building" the aircraft loading. As mentioned, there are different phases of the program where different data is available. When a six degree of freedom simulation is available, there will be inherent differences between the data for the simulation and the data within the aeroelastic model, unless there are corrections to correlate the models. If the models are well correlated, the time accurate aircraft state data should be able to be represented by the trim solution in NASTRAN, however if the data is uncorrelated, there will be deviations in aircraft control position solutions to the trim state.

It is always necessary to have the aircraft in a "balanced" state for the structural finite element model analysis. The aircraft balance satisfies the assumption of quasi-steady flight loads. The term quasi-steady is used because there is a distinction made between steady maneuvering and non-steady maneuvering. Steady maneuvering is assumed to have zero rotational acceleration terms, meaning the aircraft is steady in its maneuver and its angular rates are not changing. It may however have some linear acceleration terms. For example, a steady 7g (Nz) pull-up maneuver is typical for fighters, where the aircraft is in an accelerated state from the Nz term, but has no rotational acceleration. Typically, there is an assumed associated pitch rate as a function of the load factor, which the Mil-Spec 8861b does outline.

Non-steady maneuvers are those where the pitch acceleration terms are non-zero. For example, a dynamic pitch maneuver where the pilot would, in a quick fashion pull back on the longitudinal stick to impose an immediate nose up reaction and then later return the stuck for an abrupt nose down reaction. These maneuvers are considered to be non-steady in their abruptness, however for the purpose of static loads analysis, a single time point will be considered. This single time point is typically, but not always, assumed to be point at the highest rotational acceleration. That is, the most positive and most negative acceleration for the initiation and termination points along the maneuver. The analysis of a single time slice creates a quasi-steady assumption since the aircraft inertial forces and moments are assumed to be balanced. This is the assumption on which the static aeroelastic database will be derived.

Stability and Control Derivatives

As described above the stability and control derivatives are necessary to predict aircraft trim. These can be generated by running Sol 144 in NASTRAN with TRIM decks defined. The NASTRAN User Guide describes the trim input shown in Figure 10.

1	2	3	4	5	6	7	8	9	10
TRIM	ID	MACH	Q	LABEL1	UX1	LABEL2	UX2	AEQR	
	LABEL3	UX3	-etc.-						

Figure 10 NASTRAN Trim Input Bulk Data Entries

The input deck used to generate the “baseline” derivatives is shown in Figure 11.

The trim input deck was developed in a separate bulk data file altogether from the model, and was connected to the model by use of an "INCLUDE" statement which indicates to NASTRAN that external files exist that should be included in the solution sequence identified in the bulk data file submitted to the NASTRAN executable.

The use of an "INCLUDE" statement reduces the file sizes for multiple runs of common tasks, and allows the user to quickly "swap" out properties, components, etc. In this case, it was useful to be able to "swap" out trim solutions, so that a file for the model did not have to be developed each time, and reduced the manual efforts to a minimum. The ECHO request was used, so that the combined model would be read back out into the .f06 and .pch files.

```
$ BASELINE 1g TRIM AT MACH = 0.1 CASE ID 1
TRIM      100000    0.114.81351   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.2 CASE ID 2
TRIM      100001    0.259.25406   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.3 CASE ID 3
TRIM      100002    0.3133.3216   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.4 CASE ID 4
TRIM      100003    0.4237.0162   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.5 CASE ID 5
TRIM      100004    0.5370.3379   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.6 CASE ID 6
TRIM      100005    0.6533.2865   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.7 CASE ID 7
TRIM      100006    0.7725.8622   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.8 CASE ID 8
TRIM      100007    0.8948.0650   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
$ BASELINE 1g TRIM AT MACH = 0.9 CASE ID 9
TRIM      100008    0.91199.894   URDD2      0.    URDD3      1.      1.0+
                     URDD4      0.    URDD5      0.    URDD6      0.    ROLL      0.+
                     PITCH      0.      YAW      0.
```

Figure 11 Baseline Trim Input

The trim is varied in Mach at 0.1 increments from Mach = 0.1 to Mach = 0.9. The choice in the increment of Mach is based on the assumption of piece-wise linearity between breakpoints. In other words, there is negligibly small variation between increments of 0.1 Mach. This is a requirement because the database will interpolate across the next higher and the next lower Mach values to predict what the aerodynamic derivative is. The dynamic pressure is the only other variation in the TRIM decks, such that the altitude stays constant. Each of the corresponding Mach and qbar, or dynamic pressure in pounds-per-square foot, values corresponds to an altitude of 0. Again, it is assumed that the variation of altitude has no effect on the derivatives. Typically, this is a valid assumption and should only be affected by external energy of the system such as propulsive effects. For example, propeller wash is not simply scalable by dynamic pressure, but must be a function of dynamic pressure and altitude.

The selection of breakpoints at delta Mach = 0.1 was based on engineering judgement and validated through the results presented. If the predicted database results had significant error against the NASTRAN run results, this would indicate that poor breakpoints in Mach were chosen and the assumption of piece-wise linearity failed. In other words, a smaller Mach would have needed to be chosen. In such a case where the predicted HM was matched to many decimals places, perhaps too small of an increment was chosen. Convergence studies could be conducted to find the optimum delta Mach (or any variable) for the database.

The resulting output file is known as an ".f06" file. An example of the output is shown in Figure 12.

TRIM VARIABLE	COEFFICIENT	RIGID		ELASTIC	
		UNSPLINED	SPLINED	RESTRAINED	UNRESTRAINED
REF. COEFF.	CX	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00
	CY	-2.399320E-17	-2.399320E-17	-1.317206E-17	-2.399327E-17
	CZ	-2.541705E-16	-2.541705E-16	-2.384257E-16	-2.545114E-16
	CMX	7.437081E-17	7.437081E-17	7.409287E-17	7.436927E-17
	CMY	2.679838E-17	2.679838E-17	2.065606E-17	2.693055E-17
	CMZ	3.539705E-18	3.539705E-18	6.321998E-18	3.539381E-18
ANGLEA	CX	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00
	CY	-3.455164E-11	-3.455163E-11	2.438077E-11	7.070667E-12
	CZ	3.771357E+00	3.771357E+00	3.779107E+00	3.783832E+00
	CMX	8.105204E-13	8.107368E-13	6.586142E-11	5.226622E-11
	CMY	-1.119716E+00	-1.119716E+00	-1.122483E+00	-1.124549E+00
	CMZ	-2.095265E-11	-2.095265E-11	-1.433255E-12	-6.519285E-12
SIDES	CX	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00
	CY	-6.639979E-01	-6.639979E-01	-6.635990E-01	-6.640441E-01
	CZ	2.826981E-11	2.826981E-11	6.739469E-11	6.025614E-12
	CMX	1.187594E-01	1.187594E-01	1.188155E-01	1.187413E-01
	CMY	-5.728647E-12	-5.728627E-12	-2.336344E-11	3.992909E-12
	CMZ	-1.321387E-01	-1.321387E-01	-1.320268E-01	-1.321558E-01
ROLL	CX	0.000000E+00	0.000000E+00	0.000000E+00	0.000000E+00
	CY	1.580472E-01	1.580472E-01	1.581682E-01	1.581366E-01
	CZ	2.279074E-11	2.279099E-11	-2.106135E-10	-2.606459E-11
	CMX	-3.757508E-01	-3.757508E-01	-3.764519E-01	-3.759733E-01
	CMY	-2.074796E-11	-2.074792E-11	8.041501E-11	-2.574917E-12
	CMZ	-4.546690E-05	-4.546690E-05	-1.023702E-04	-4.804922E-05

Figure 12 .f06 Aerodynamic Derivative Ouptut

In order to calculate the estimated trim for a given Mach number, single variable interpolation across Mach is accomplished via a method of multivariable interpolation. Multivariable interpolation is necessary due to the fact that the tables are 5x5 (trim variables) and 5x3 (set variables). The method used is a MATLAB function, '*interpn'* which can be used to interpolate across multi-dimensional arrays. Because the input breakpoints are dummy arrays, 1 through 5 and 1 through 3, are the same as the query arrays, the interpolation is one-dimensional across multiple two dimensional tables. The interpolation method used is ‘linear’. The dummy arrays could in theory be any value so long as the query arrays are the same as the breakpoint arrays.

A hand calculation was performed to verify that indeed the interpolation scheme and the database produced what was expected.

Table 1 Stability Derivative Interpolation Check

Test Interpolate, Mach = 0.25								
	ANGLEA	**SIDES**	**ELEV**	**AILERON**	**RUDDER**	**ROLL**	**PITCH**	**YAW**
CY	0	0.1194595	0	-0.2243	-0.02414	-0.37865	0	0.065861
CZ	3.8221785	0	0.117279	0	0	0	7.126874	0
MX	-1.143025	0	0.2168985	0	0	0	-4.22647	0
MY	0	-0.668085	0	0.066483	0.207653	0.157521	0	-0.55717
MZ	0	-0.13301	0	-0.00175	0.088882	-0.00057	0	-0.1785
INTERPOLATED VALUES FROM MATLAB CODE								
	ANGLEA	**SIDES**	**ELEV**	**AILERON**	**RUDDER**	**ROLL**	**PITCH**	**YAW**
CY	0	0.1194595	0	-0.2243	-0.02414	-0.37865	0	0.065861
CZ	3.8221785	0	0.117279	0	0	0	7.126874	0
MX	-1.143025	0	0.2168985	0	0	0	-4.22647	0
MY	0	-0.668085	0	0.066483	0.207653	0.157521	0	-0.55717
MZ	0	-0.13301	0	-0.00175	0.088882	-0.00057	0	-0.1785

Now that the stability derivatives are created, a trim estimation can be generated for a given aircraft maneuvering state. In order to verify that the trim estimation works properly, a simple check is performed against the existing NASTRAN Aeroelastic Trim solutions from the database generation. Using the methods prescribed, the same trim solution that was generated from the actual NASTRAN run should be able to be generated from the database.

A summary of the stability and control derivatives are plotted in Appendix C
Flexible Stability Derivatives to compare the derivatives in each of the six degrees of freedom (only five are actually used in the TRIM analysis). These figures are useful in

determining the effectiveness or contribution from each variable on the aircraft trim. For example, one can compare the effects on aircraft rolling moment between the baseline mean flow state, angle of attack, sideslip angle, roll rate, pitch rate, yaw rate, Ny, Nz, Pdot, Qdot, Rdot, elevator control surface deflection, aileron control surface deflection and rudder control surface deflection. These plots should be used to validate modeling from a qualitative perspective. Intuitively, it should be expected that variables such as aileron deflection, roll rate, yaw rate, rudder deflection largest “effectiveness”. In fact, it can be seen that roll rate has the most significant effectiveness, followed by aileron deflection, sideslip angle and yaw rate.

Appendix E

Unit Hinge Moment Database Per Mach and Variable shows individual degrees of freedom with respect to each of the stability and control derivatives. Additionally, both flexible and rigid aircraft derivatives are shown, to compare the resulting variation of variable effectiveness affected by the aircraft flexibility, or the static aeroelastic increments.

These static aeroelastic increments can be considered “flex to rigid” ratios. These are described as the ratio of the flexible values over the rigid values. These are assumed to follow a linear relationship, by the assumption of small aircraft deformations. Of course, if the structure deformed such that linear assumptions are no longer valid, more intricate methods, outside the scope of this paper, are necessary. However, for most practical aircraft applications, these assumptions of linearity are valid.

Notice, for this analysis it is assumed that the only non-linear variability is that each of the derivatives are non-linear in Mach. In other words, the assumption is that the stability and control derivatives are linearly independent functions of each other, and dependent only on Mach. This paper uses single baseline variable non-linearity for simplicity purposes, however, in many aircraft applications, these derivatives are dependent with respect to each other. For example, the aileron control surface effectiveness at a beta value of 10 degrees, will be in fact, different than that of of the aileron control surface effectiveness at a beta value of 0 degrees. In order to capture these coupled effects, multi-variable database interpolation methods are needed. These multi-variable interpolation methods are simply an extension of the presented method, whereas instead of simply interpolating across a single variable independently, each of the derivatives are interpolated across each of the other variables as well. It should also be noted that if this multi-variable interpolation is required for the aircraft analysis, the order of interpolation does matter. The ordering and methodology for interpolation should be studied, so that the methods of interpolation do not skew inteded results from what the “true” database values are.

Now that the aircraft stability and control derivatives are defined, the database can be generated. As stated, a single variable, Mach, is interpolated across. Visually, this is expressed in Figure 13.

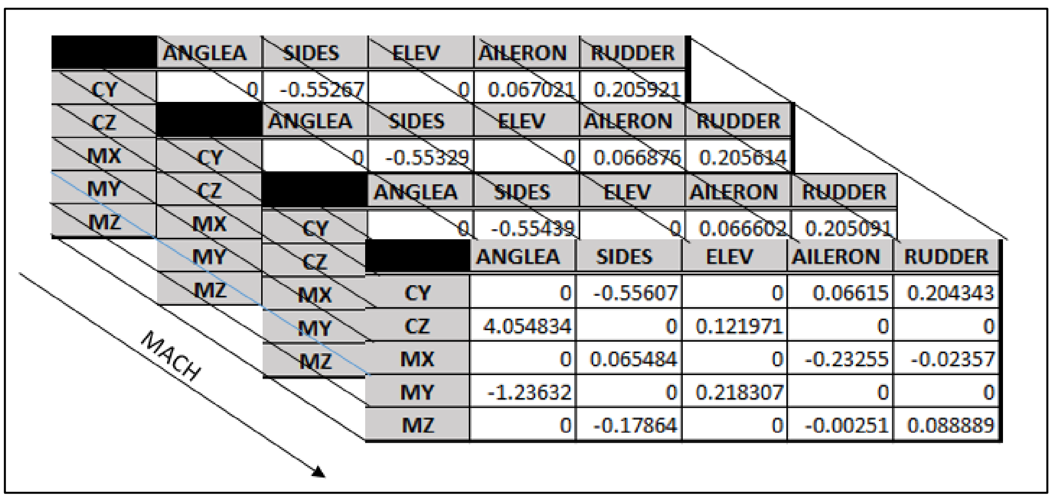


Figure 13 Derivative Database Representation

Given that the aircraft trim state can be predicted for any load factor, Mach, Altitude and rotational accelerations, the use of the aircraft hinge moment derivatives can be used to develop predicted aircraft hinge moments. A hinge moment database is developed in a similar fashion that the stability and control database is developed.

The aircraft derivatives will give a trim solution, but still do not give insight into aircraft loading. To develop this database, another matrix of static aeroelastic runs is required. These runs are used to develop aircraft loading increments as a function of the aircraft degrees of freedom, i.e. Y, Z, L, M and N.

Incremental Load Sets

In order to predict loading at any aircraft state, incremental aircraft loading will need to be computed at various states. For the example presented in this paper, the following matrix in Table 2 was generated.

Table 2 Incremental State Matrix

	Mach	Ny	Nz	$\dot{P}$ [deg/s²]	$\dot{Q}$ [deg/s²]	$\dot{R}$ [deg/s²]	P [deg/s]	Q [deg/s]	R [deg/s]
Start Value	0.1	-2	-4	-100	-100	-50	-100	-100	-50
Increment	0.1	1	1	50	50	25	50	50	25
End Value	0.9	2	9	100	100	50	100	100	50

The values chosen for this paper are somewhat arbitrary and the fact that multiple values (greater than two) were chosen is unnecessary here but would be necessary in the event of non-linearity. The only necessary set of multiple values in this example is in Mach, which as was stated previously, is assumed to have an effect on the aerodynamic derivatives.

In this exercise, the aircraft control surface hinge-moments are virtually instrumented with “monitor points”. Monitor points are groups of elements and nodes in the finite element model that reference a coordinate system and whose output values in the NASTRAN solution are tabulated.This is an easy way to get output load data from each run, without having to use some form of external load calculation.

An example of the output of the aerodynamic monitor point integrated loads is shown in Figure 14. These loads are parsed and read into a database (both flexible and rigid solutions were run). The values for hinge moments output in the NASTRAN .f06 file are in units of [lb-ft^2]. The hinge moments per condition are divided by the dynamic pressure of that condition. The purpose of this is such that a condition at any altitude at a given Mach can be computed. Otherwise, there would need to be a matrix presented above developed for each altitude desired – or at the minimum a maximum and minimum altitude which can be interpolated on. Both of these methods would be unnecessary since scaling by dynamic pressure to vary altitude at a given Mach is just as accurate.

```
                 A E R O D Y N A M I C   M O N I T O R   P O I N T   I N T E G R A T E D   L O A D S
              CONFIGURATION = AEROSG2D     XY-SYMMETRY = ASYMMETRIC     XZ-SYMMETRY = ASYMMETRIC
                                MACH = 1.000000E-01               Q = 1.481351E+01

CONTROLLER STATE:
SIDES    =   1.6553E+01        URDD2    =  -2.0000E+00        AILERON  =   6.1863E+00        RUDDER   =   2.4946E+01

MONITOR POINT NAME = AEROSG2D         COMPONENT =                   CLASS = COEFFICIENT
LABEL = Full Vehicle Integrated Loads
CID =   100000          X =  0.00000E+00          Y =  0.00000E+00          Z =  0.00000E+00

  AXIS      RIGID AIR       ELASTIC REST.
  ----    -------------    -------------
   CX     0.000000E+00     0.000000E+00
   CY    -3.218030E+04    -3.220000E+04
   CZ     1.898355E-07    -5.194247E-12
   CMX   -3.499258E+02    -1.600002E-02
   CMY   -5.209991E-07     5.496267E-13
   CMZ    1.902498E+02     1.040007E-01

MONITOR POINT NAME = AILERON          COMPONENT = 3                 CLASS = HINGE MOMENT
LABEL = AILERON  - Control Surface Hinge Moment
CID =      110          X =  0.00000E+00          Y =  0.00000E+00          Z =  0.00000E+00

  AXIS      RIGID AIR       ELASTIC REST.
  ----    -------------    -------------
   CMY   -3.374169E+03    -3.373870E+03

MONITOR POINT NAME = AILERON          COMPONENT = 4                 CLASS = HINGE MOMENT
LABEL = AILERON  - Control Surface Hinge Moment (Second Axis)
CID =      210          X =  0.00000E+00          Y =  0.00000E+00          Z =  0.00000E+00

  AXIS      RIGID AIR       ELASTIC REST.
  ----    -------------    -------------
   CMY   -3.374169E+03    -3.373870E+03

MONITOR POINT NAME = ELEV             COMPONENT = 1                 CLASS = HINGE MOMENT
LABEL = ELEV     - Control Surface Hinge Moment
CID =       90          X =  0.00000E+00          Y =  0.00000E+00          Z =  0.00000E+00

  AXIS      RIGID AIR       ELASTIC REST.
  ----    -------------    -------------
   CMY    5.556094E+03     5.556119E+03

MONITOR POINT NAME = ELEV             COMPONENT = 2                 CLASS = HINGE MOMENT
LABEL = ELEV     - Control Surface Hinge Moment (Second Axis)
CID =       90          X =  0.00000E+00          Y =  0.00000E+00          Z =  0.00000E+00

  AXIS      RIGID AIR       ELASTIC REST.
  ----    -------------    -------------
   CMY   -5.556094E+03    -5.556119E+03
```

Figure 14 Example Monitor Point Loads

An example plot of the hinge moment vs. aircraft pitch rate can be seen in Figure 15. Because of the linear assumptions made by use of the doublet-lattice solution, and as previously stated, only two values of pitch rate are necessary to predict the aircraft hinge moment based on that pitch rate. However, any non-linearities due to other aircraft effects would determine how many breakpoints in the database are necessary. Having excessive break points does not affect the solution and should not affect runtime.

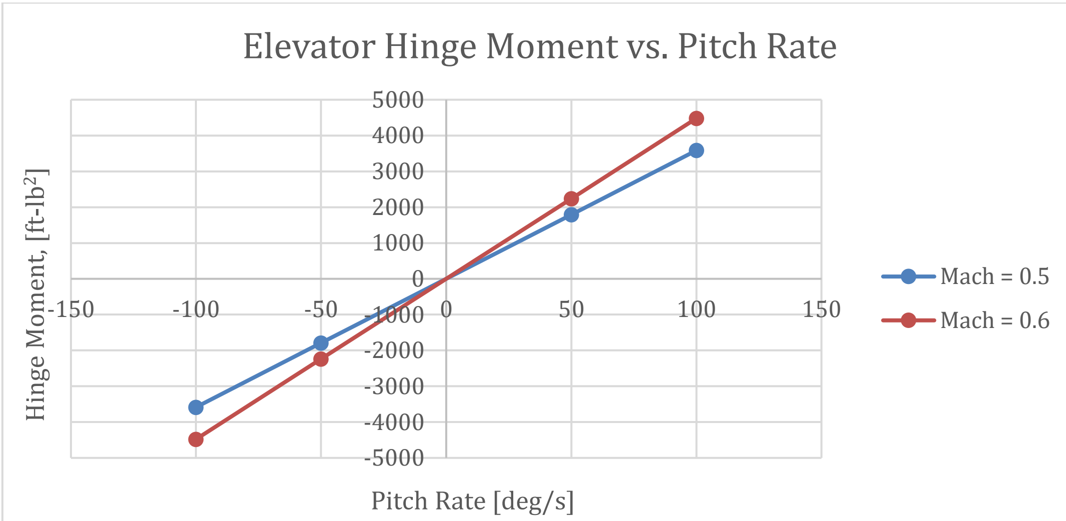


Figure 15 Elevator Hinge Moment vs. Pitch Rate, Mach 0.5, 0.6

Database Predicted Loads Results

Control Surface Hinge Moments

In order to validate the control surface hinge moment loads, test cases were run with varying degrees of freedom in the solution. These cases are a combination of "realistic" restricted degrees of freedom, such as 2 degree of freedom dynamic pitching conditions, and higher degree of freedom conditions, where Ny, Nz, Pdot, Qdot and Rdot are included degrees of freedom with non-zero values. These conditions are used to validate that the prediction methods are not limited to reduced degrees of freedom and can be used for all degrees of freedom of interest.

To prove the theory presented, a set of nominal Mil-Spec maneuvers and those which are "unordinary" maneuvers were run to show that the solution is valid where all degrees of freedom and trim variables are used.

The first set of conditions run were for standard "dynamic pitching maneuvers". See section "Military Specifications" for more information regarding this and other type of standard Mil-Spec maneuvers. There are two subsets of the dynamic pitching maneuvers presented in Appendix F
Proof of Concept Test Matrices. The abrupt pitch initiation assumes that the pilot is at steady flight, and in this case at elevated or negative-g load factors, and an abrupt maneuver to the longitudinal stick is generated such that no pitch rate occurs by aircraft response and only aircraft acceleration acts on the vehicle in the form of Nz and Qdot. The abrupt pitch termination assumes that the pilot has already initiated a pitching maneuver and the aircraft has responded with some pitch rate. The pilot then puts longitudinal stick in such that the aircraft accelerates in the opposite direction from the pitch rate developed.

The unit control surface hinge moments are outlined in Appendix E
Unit Hinge Moment Database Per Mach and Variable. All hinge moments reference the RHS aileron or RHS elevator hinge moments.

Suppose the condition Ny = 0.45 [g] , Nz = 4.3 [g] , Pdot = 13 [deg/s^2], Qdot = 49 [deg/s^s], Rdot = 4 [deg/s^2], P = -24 [deg/s], Q = 3 [deg/s] and R = 2 [deg/s] at Mach = 0.3. The resulting unit hinge moment coefficients are presented in Table 3.

The coefficients are then multiplied through by the values of the condition, wehre URDD2 and URDD3 are in units of g's (non-dimensional) and the rotational acceleration and rotational rate terms are in units of deg/s^s and deg/s, respectively.

The theory for this echoes that which was developed for the static aeroelastic trim, in Static Aeroelastic Trim Theory.

Table 3 Unit Hinge Moments at Mach = 0.3

Mach	RATE VAR	Aileron Hinge Moment/(qbar* URDD2)	Elevator Hinge Moment/(qbar* URDD2)	Rudder Hinge Moment/(qbar* URDD2)
0.3	URDD2	12.839	-20.7369	31.7138
0.3	URDD3	0.961	57.4937	0
0.3	PDOT	0.5027	-0.0659	0.1212
0.3	QDOT	-0.0161	4.5065	0
0.3	RDOT	-0.5489	1.7987	-5.9859
0.3	P	0.064	-0.0192	-0.0046
0.3	Q	0.0065	0.1533	0
0.3	R	0.0045	-0.0165	-0.0296

For example, the total aileron hinge moment is calculated by Equation 18.

Equation 18 Total Hinge Moment Calculation

$$\begin{Bmatrix} \boldsymbol{Aileron_{HM}} \\ \boldsymbol{Elevator_{HM}} \\ \boldsymbol{Rudder_{HM}} \end{Bmatrix} = \mathbf{133.3216} \left(\begin{bmatrix} \mathbf{12.839} & \mathbf{-20.7369} & \mathbf{31.7138} \\ \mathbf{0.961} & \mathbf{57.4937} & \mathbf{0} \\ \mathbf{0.5027} & \mathbf{-0.0659} & \mathbf{0.1212} \\ \mathbf{-0.0161} & \mathbf{4.5065} & \mathbf{0} \\ \mathbf{-0.5489} & \mathbf{1.7987} & \mathbf{-5.9859} \\ \mathbf{0.064} & \mathbf{-0.0192} & \mathbf{-0.0046} \\ \mathbf{0.0065} & \mathbf{0.1533} & \mathbf{0} \\ \mathbf{0.0045} & \mathbf{-0.0165} & \mathbf{-0.0296} \end{bmatrix}^{\boldsymbol{T}} \begin{Bmatrix} \mathbf{0.45} \\ \mathbf{4.3} \\ \mathbf{13} \\ \mathbf{49} \\ \mathbf{4} \\ \mathbf{-24} \\ \mathbf{3} \\ \mathbf{2} \end{Bmatrix} \right)$$

The result gives hinge moments for aileron, elevator and rudder as 1593 [ft-lb], 62,119 [ft-lb] and -1072 [ft-lb], respectively. These results can be checked qualitatively against the maneuver representation. It is an elevated Nz maneuver with a dominant pitch acceleration term, therefore, the highest hinge moments are expected to be the elevator hinge moment term.

Utilizing the Database

Now that the aircraft database has been generated, the analyst has the first set of tools available to generate design aircraft load cases (at least for control surface component cases). The goal is to develop full aircraft, flexible and balanced loads to deliver to the stress team.

Conceptual and Preliminary Design Usage

Suppose the aircraft is in the conceptual design phase, and it is up to the flight loads analysts to come up with trends of aircraft hinge moment for a particular concept. At this stage in the design phase, a rudimentary aircraft shape, number of control surface and control surface sizing and generic propulsion data is known, however there is no aerodynamic data (high order CFD or wind tunnel). The method presented thus far provides aerodynamic data. The analyst can generate a simplified stiffness and mass model with doublet-lattice aerodynamics to produce aircraft loading. It is assumed that there are some general aircraft requirements but even these may not be set in stone yet, or they are being developed based on these studies.

Neill and Whiting state, "In the design of aircraft, it is important to have an accurate simulation of both the structural characteristic and the aerodynamic characteristics of the vehicle to produce accurate trimmed loads. While MSC/NASTRAN has long had a static aeroelastic analysis capability, it utilizes the embedded unsteady aerodynamic methods (e.g., Doublet-Lattice) at zero reduced frequency." They go on to state that "…these methods are very useful for aeroelastic analyses at the conceptual and preliminary design stage". [3]

Indeed, these methods are very powerful tools for conceptual and preliminary design. An example of how this method can be implemented is developing initial criteria for conceptual and preliminary design sizing. In order for the structural design and analysis teams to come up with conceptual or parametric sizing, an initial set of loads need to be balanced aircraft load set must be developed. The question arises as to what load cases should be delivered.

The details of these decisions depend on the nature of the aircraft development, and usually specify particular wing stations of interest, fuselage stations, control surface hinge moments, elevator stations, and rudder stations. In this paper, only the control surfaces had monitor point loads, however, it is just as easy for any component of the vehicle to have the same methodology applied to it. The same process would be followed, where aircraft loads are normalized to the angular rate, angular acceleration or load factor associated to that incremental run and in turn a database of coefficients which can be multiplied by the dynamic pressure and variable value can be used to quickly identify critical component loads.

In this example, suppose one of the components to critique is the elevator hinge-moment to set preliminary actuator sizing requirements. A survey could be run, much like the one presented in Appendix F
Proof of Concept Test Matrices, however expanded for incremental pitch rates, associated with some pitch accelerations at normal vertical load factors. Such survey increments are presented in Table 4.

Table 4 Abrupt Pitch Initial Sizing Survey

	Ny	Nz	Pdot [deg/s^2]	Qdot [deg/s^2]	Rdot [deg/s^2]	P [deg/s]	Q [deg/s]	R [deg/s]
Min Value	0	-2	0	-200	0	0	-100	0
Increment	0	1	0	25	0	0	25	0
Max Value	0	4	0	200	0	0	100	0

An expanded test matrix results in a combination of these values to 9639 combinations of cases over increments of Mach from 0.1 to 0.9, following suit with the database break points, though, any Mach value between the bounds of the database is valid. The resulting array of elevator control surface hinge moments are sorted from most positive to most negative (the sign of the hinge moment is important, especially for actuator sizing, since depending on the control actuation device, it will be actuator extend upper/lower or actuator extend and retract loads). All results of the survey are presented in Figure 16 Dynamic Pitch Initial Actuator Sizing Survey - Elevator Hinge Moment

The tabulated results of the critical conditions are provided in Table 5 for maximum positive hinge moments and in Table 6 for maximum negative hinge moments.

Table 5 Maximum Positive Elevator Hinge Moment Results

Case ID	Mach	Ny	Nz	Pdot [deg/s^2]	Qdot [deg/s^2]	Rdot [deg/s^2]	P [deg/s]	Q [deg/s]	R [deg/s]	Elevator HM [ft lb]
9639	0.9	0	4	0	200	0	0	100	0	164,402
9638	0.9	0	4	0	200	0	0	75	0	162,195
8568	0.8	0	4	0	200	0	0	100	0	161,029
9637	0.9	0	4	0	200	0	0	50	0	159,988
8567	0.8	0	4	0	200	0	0	75	0	159,312

Table 6 Maximum Negative Elevator Hinge Moment Results

Case ID	Mach	Ny	Nz	Pdot [deg/s^2]	Qdot [deg/s^2]	Rdot [deg/s^2]	P [deg/s]	Q [deg/s]	R [deg/s]	Elevator HM [ft lb]
8569	0.9	0	-2	0	-200	0	0	-100	0	-147,868
8570	0.9	0	-2	0	-200	0	0	-75	0	-145,662
7498	0.8	0	-2	0	-200	0	0	-100	0	-144,896
8571	0.9	0	-2	0	-200	0	0	-50	0	-143,455
7499	0.8	0	-2	0	-200	0	0	-75	0	-143,179

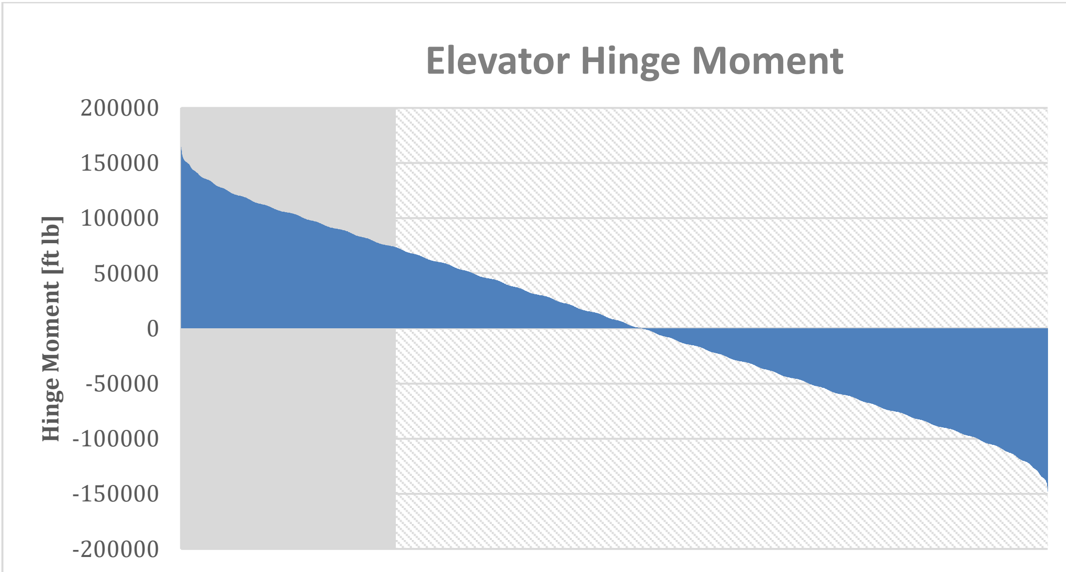


Figure 16 Dynamic Pitch Initial Actuator Sizing Survey - Elevator Hinge Moment

The trend in Figure 16 Dynamic Pitch Initial Actuator Sizing Survey - Elevator Hinge Moment that the results are not simply linear, however the tabulated results do show, that the combination of the max positive pitch acceleration, with the max positive pitch rate at the maximum positive load factor does cause the most positive hinge moment. A positive hinge moment, as can be seen in the tabulated results in Appendix F
Proof of Concept Test Matrices, is from a trailing edge up deflection causing aircraft nose up moment.

A figure is presented to show the trends of Mach on the elevator hinge moments. Because the same conditions were run at each Mach, a direct case by case comparison can be done. Presented in Figure 17 is a summary of the same data presented in Figure 16, except that the data is sorted and colored by Mach. A trend of the peaks, per Mach, can be seen with

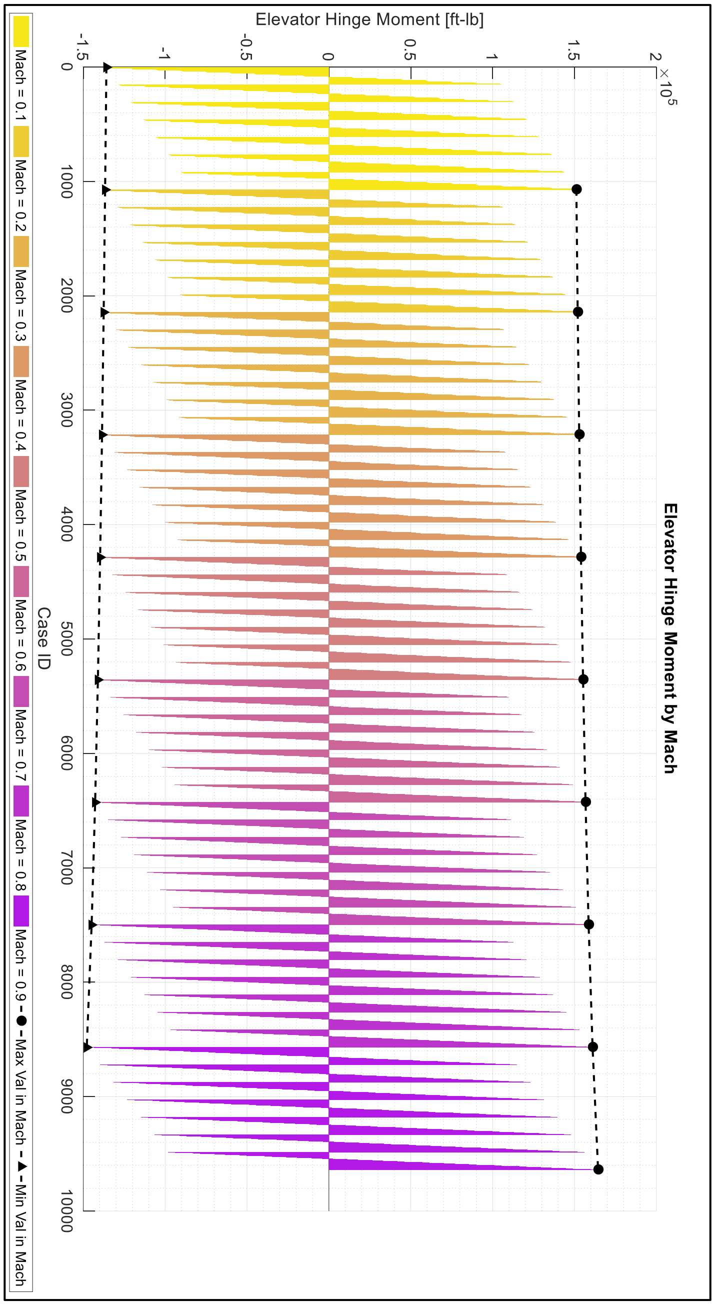


Figure 17 Dynamic Pitch Initial Actuator Sizing Survey - Elevator Hinge Moment Sorted by Mach

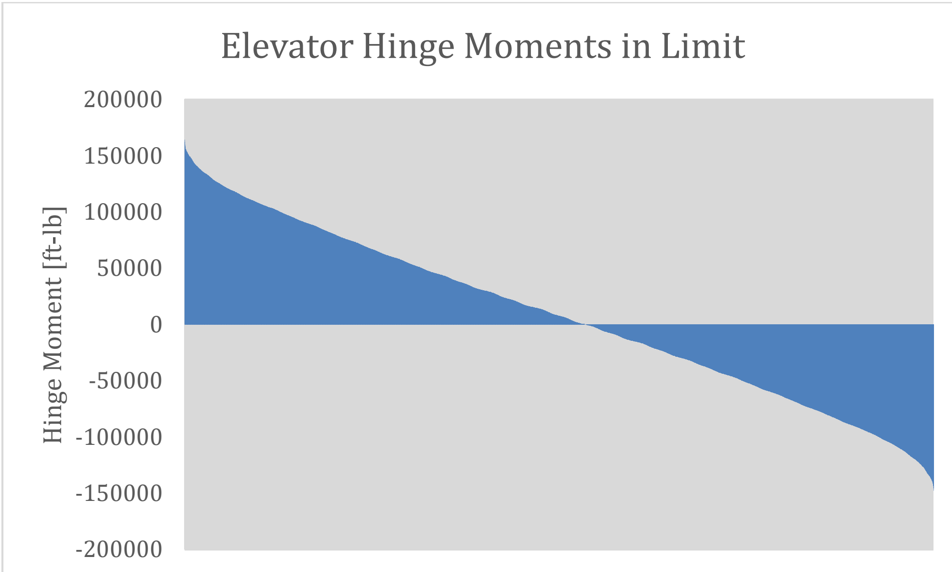
the dashed black lines, where the dashed black line with circles trends the maximum peak positive elevator hinge moments and the dashed black line with triangles trends the maximum peak negative hinge moments. Of course, again, piecewise linearity between the variations of Mach is the only non-linear assumed term in this survey.

There is, however, one important piece of information that has been neglected in the assumptions of this survey, and this the fact that the aircraft control authority limits have been neglected. The control derivatives can be extrapolated to solve any aircraft state that is input, however, the question of whether the results are valid needs to be analyzed. Therefore, the trim estimator must be used to predict control surface deflections.

Since the aircraft control derivatives were developed using doublet-lattice methods, there is some inherent factor that should be maintained due to the primitive nature in the solution methods. Therefore, when the down-select of critical hinge moments limited by aircraft control authority is taken into account, some margin should be assumed for limits of the control surface.

Suppose the aircraft was limited to 45 degrees trailing edge up and 45 degrees trailing edge down. Additionally, since alpha is assumed to have no stall limits, stall in alpha is assumed at +20 degrees and -15 degrees.

Table 7 Elevator Hinge Moment in Alpha and Surface Deflection Limit



The resulting maximum positive and maximum negative hinge moments obtained from the survey satisfy the “realistic” stall and surface travel limits. There were a reduction of about 20% of the cases, of which occurred at the lowest qbar and highest accelerations. This validates logical reasoning because the aircraft capability is limited at lower airspeeds.

This type of survey can be useful in predicted aircraft limits during preliminary and conceptual design phases. Perhaps the vehicle is maneuverable limited by airframe structural design limits. As the aircraft requirements are being developed, there may be a maximum allowable design load for some portion of the aircraft, which shall not be exceeded. If this is the case, it would be up to the flight controls team to rectify this requirement by setting aircraft control limits on the pilot, or reducing aircraft capability from a controls standpoint.

Of course, this is dependent on the type of aircraft and its mission purpose. Perhaps for a Cirrus-SR22 weight optimization may be key, while maneuverability can be sacrificed. The highest weighted objective to some aircraft design is minimum weight and a lower weighted objective is aircraft maneuverability.

In the case of fighter design, such as the example, aircraft maneuverability may be the highest objective requirement to meet the customer's expectations and therefore the airframe structure is a fallout of that. Typically, though, as aircraft weight increases, aircraft maneuverability tends to decrease and thus performance becomes a function of itself.

In order to reduce the opportunity of getting caught in a circular requirement loop of maneuverability and structural rigidity, clear requirements should be set early in the developmental phases of the design, so that both teams can work together to come up with trade space studies weighing out impacts of aircraft maneuvering capability and structural strength and weight as a fallout, or vice-versa.

This type of survey, as mentioned, can be expanded to any component on the vehicle. It is typical that the stress team will want to analyze the vehicle for the critical load conditions. Suppose that one of the critical locations surveyed was the wing root, which stress wants to analyze on the full vehicle for critical internal loads. That condition can be generated by running a survey with monitor point loads integrated at that location to find a critical trim associated to that load.

While it is of interest to the structural analysis team what the wing root bending and shear are, for them to analyze that loading on the FEM, they will need distributed net loads.

These nets loads are the distribution of aerodynamic loads and inertial loads and select points on their FEM. While the methods to convert aeroelastic model loads to the FEM model (assuming they are different), the process to develop critical aeroelastic model loads is discussed.

In running the database generation, the aerodynamic pressure associated with the flexible solution is captured into a similar database. This database is an array of pressure coefficients at that condition. In the same method for creating derivatives of total monitor point loads at each of the flight conditions, incremented on aircraft state values, the same can be done for the pressure coefficients. If there is an nx1 array where n is the number of grid points on the aeroelastic model, that array can be divided by the increment, which can then be later post-multiplied through to get the incremental aerodynamic pressure for that condition.

The inertial forces at discreet points where lumped masses represent the aircraft structure and subsystems, can be calculated using the same equations presented in Static Aeroelastic Trim Theory, using Euler rigid body mechanics.

The combination of these can be used to develop “distributed point loads” on the aeroelastic model. The method for transferring the distributed point loads on the aeroelastic model to the finite element model for internal loads assessment is very complex and there are many methods to accomplish this. One method is presented by Samareh. [9]

Detail Design and Flight Test

The process for generating a critical loads survey should vary only slightly between the various design phases. In the previous section, detailed discussion the development of critical elevator hinge moments were laid out. The assumptions were that no wind tunnel or CFD data existed and that the loads analyst only had a basic structural beam model representing aircraft rigidity and a basic doublet-lattice model for aircraft aerodynamics.

During the detail design phase, it can be assumed that high order CFD, such as Euler or Navier-Stokes solutions are used along with wind tunnel test data. With this assumption, the use of the doublet-lattice solution is unnecessary, however the methods of developing derivatives can still be consistent with the previously described methodology.

There is a method of interfacing external, high order aerodynamics into NASTRAN for aeroelastic analysis presented by Whiting and Neill. [3] The method describes replacing the doublet-lattice solution with that of the high order CFD. With this, there are many assumptions which can be eliminated and details which can be added.

With the use of high order CFD and wind tunnel data, accurate total aircraft coefficients can be captured. The coefficients should be consistent across disciplines, so that the flight controls team is working to the same data the loads team is analyzing to. In the conceptual and preliminary design phases there may be discrepancies, as expected. There may, however still be discrepancies even at the later detail design phases of design. The flight controls team may have simulations with rigid aircraft assumptions, while the loads team will have aeroelastic effects in their analysis. This should be noted if time history data is used in calculating time accurate aircraft loads.

Modeling fidelity can increase with the use of CFD by capturing wing-body interaction and complex flow such as effects due to inlet and exhaust effects. The doublet-lattice method is restricted to neglect effects due to propulsion, unless a direct matrix input, or DMI is used to supersede the doublet-lattice pressures. Of course, this is not possible with CFD solutions defining those inputs.

With the usage of high order CFD and wind tunnel data, non-linear effects can be captured. To utilize this data a multi-dimensional non-linear database would need to be generated.

The format of the database is consistent with that of the single-dimension non-linear database which was generated in the example of this paper. In the sample presented, the non-linear independent variable was Mach with dependent variables of alpha, beta, roll rate, pitch rate, yaw rate, elevator deflection, aileron deflection and rudder deflection. The variation is the extension to n-dimensions, where n is the number of independent variables. The database becomes a multi-dimensional array of size m x n1 x n2 x n3 x … nk where k is the number of dependent variables. The same method of interpolation within MATLAB can be utilized.

Additionally, increased fidelity on time accurate aircraft simulation, or six degree of freedom simulation becomes available as the aircraft design matures. Utilizing the flight control's simulation increases the reliability of the data, due to modeling of the intended aircraft flight control system involving surface scheduling, accurate gains and other parameters that a basic simulation would not capture.

In the flight test phase, it is critical to be able to produce and assess loads quickly and accurately. Having a well correlated database to the instrumented flight test vehicle is critical during the envelope expansion phase of flight test. There is limited literature on instrumented loads flight testing. The extent of the discussion on flight test methods will be limited in this paper as well. The methods for which a flight test operation is conducted can vary greatly, from vehicle to vehicle, however, the intent is that the loads team has predicted critical loading on the vehicle such that the loads experienced during flight test do not exceed those which were used to design the vehicle.

In an effort to rapidly certify the aircraft through different flight speeds and altitudes, having a derivative database which is well correlated to flight test data allows the analysts to predict loads and "next" test points. If the analysts can accurately predict loads during the envelope expansion phase, certain points may be able to be skipped, showing them as confidently benign maneuver points, and show the aircraft “good” at the predicted critical points. Good correlation also gives confidence through the rest of the envelope not tested, since it is not reasonable to fly at every single point in the envelope, performing every maneuver. [11]

Additionally, with the usage of the derivative database, the loads analysts can predict loads real-time to assist in “knock-it-off” calls. These calls are to immediate stop the maneuvering and have the pilot resume to level and safe flight attitudes and airspeeds.

There are many methods for correcting the database to correlate to flight test data points, the details of which are up the analyst to prove, however, it is very efficient to have methods of reducing manual efforts. With this, as the flight test is conducted, the database can be automatically corrected. As the database is corrected for previous maneuvers, this allows the analyst to predict loads for the next maneuver. With good correlation, the program has confidence to be more aggressive in removing test points in the program and thus saving money and time for the flight test program.

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Appendix A
Tabulated Rigid Stability Derivatives at Incremental Mach Values

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.10 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	0.118759	0	-0.22182	-0.02385	-0.37575	0	0.065414
CZ	3.771357	0	0.11746	0	0	0	7.022341	0
MX	-1.11971	0	0.215084	0	0	0	-4.15394	0
MY	0	-0.66399	0	0.066982	0.206269	0.158047	0	-0.55318
MZ	0	-0.13213	0	-0.00149	0.08808	-9.547	0	-0.17729

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.20 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	0.119137	0	-0.22315	-0.024	-0.37731	0	0.065654
CZ	3.798565	0	0.11737	0	0	0	7.078246	0
MX	-1.13214	0	0.216056	0	0	0	-4.1926	0
MY	0	-0.6662	0	0.066719	0.207018	0.157766	0	-0.55533
MZ	0	-0.13261	0	-0.00162	0.088512	-0.00033	0	-0.17794

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.30 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	0.119782	0	-0.22545	-0.02427	-0.37998	0	0.066067
CZ	3.845792	0	0.117188	0	0	0	7.175501	0
MX	-1.15391	0	0.217741	0	0	0	-4.26033	0
MY	0	-0.66997	0	0.066247	0.208288	0.157275	0	-0.55901
MZ	0	-0.13341	0	-0.00187	0.089251	-0.00081	0	-0.17906

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.40 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	0.120718	0	-0.22885	-0.02467	-0.38388	0	0.066674
CZ	3.915868	0	0.11688	0	0	0	7.320131	0
MX	-1.18621	0	0.220211	0	0	0	-4.36123	0
MY	0	-0.67547	0	0.065511	0.210129	0.156532	0	-0.5644
MZ	0	-0.13456	0	-0.00223	0.090334	-0.00153	0	-0.18068

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.50 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	0.121984	0	-0.23358	-0.02523		-0.3892	0	0.067512
CZ	4.013866	0	0.116356	0	0		0	7.523119	0
MX	-1.23178	0	0.22363	0	0		0	-4.50409	0
MY	0	-0.68297	0	0.064397	0.212574		0.155464	0	-0.57174
MZ	0	-0.13609	0	-0.00274	0.091803		-0.00252	0	-0.18286

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.60 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	0.123641	0	-0.24002	-0.026		-0.39623	0	0.06864
CZ	4.148857	0	0.115476	0	0		0	7.804097	0
MX	-1.29533	0	0.228248	0	0		0	-4.7042	0
MY	0	-0.69291	0	0.062744	0.215705		0.153994	0	-0.58148
MZ	0	-0.13806	0	-0.00345	0.09374		-0.00382	0	-0.18572

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.70 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	0.125789	0	-0.24883	-0.02707		-0.40545	0	0.070168
CZ	4.335835	0	0.113862	0	0		0	8.195831	0
MX	-1.38441	0	0.234562	0	0		0	-4.98743	0
MY	0	-0.70602	0	0.060226	0.219609		0.151984	0	-0.59433
MZ	0	-0.14051	0	-0.00444	0.096266		-0.00553	0	-0.1894

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.80 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	0.128614	0	-0.26125	-0.02856		-0.41771	0	0.072313
CZ	4.606368	0	0.11076	0	0		0	8.766963	0
MX	-1.51529	0	0.243522	0	0		0	-5.40943	0
MY	0	-0.72378	0	0.056176	0.224394		0.14929	0	-0.6117
MZ	0	-0.14357	0	-0.00585	0.099587		-0.00776	0	-0.19419

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.90 (RIGID)

	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	0.132558	0	-0.28011	-0.03075		-0.43467	0	0.075649
CZ	5.040129	0	0.103787	0	0		0	9.689857	0
MX	-1.73046	0	0.257668	0	0		0	-6.11739	0
MY	0	-0.74937	0	0.049118	0.229748		0.14582	0	-0.63644
MZ	0	-0.14739	0	-0.00796	0.103989		-0.01069	0	-0.20063

Appendix B
Tabulated Flexible Stability Derivatives at Incremental Mach Values

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.10 (FLEX)									
	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	-0.55267	0	0.067021	0.205921		-0.66359	0.158168	0
CZ	3.779107	0	0.117743	0	0		0	0	7.02638
MX	0	0.065341	0	-0.22204	-0.02378		0.118815	-0.37645	0
MY	-1.12248	0	0.21498	0	0		0	0	-4.15513
MZ	0	-0.17716	0	-0.0015	0.087993		-0.13202	-0.0001	0

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.20 (FLEX)									
	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	-0.55329	0	0.066876	0.205614		-0.6646	0.158254	0
CZ	3.830249	0	0.118527	0	0		0	0	7.094713
MX	0	0.065362	0	-0.22403	-0.02374		0.119368	-0.38016	0
MY	-1.14347	0	0.215628	0	0		0	0	-4.19747
MZ	0	-0.17744	0	-0.00169	0.088161		-0.13216	-0.00056	0

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.30 (FLEX)									
	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	-0.55439	0	0.066602	0.205091		-0.66635	0.158389	0
CZ	3.919795	0	0.119894	0	0		0	0	7.213746
MX	0	0.065405	0	-0.22746	-0.02366		0.120329	-0.38656	0
MY	-1.18047	0	0.216735	0	0		0	0	-4.27167
MZ	0	-0.17793	0	-0.00202	0.088452		-0.13239	-0.00135	0

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.40 (FLEX)									
	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	-0.55607	0	0.06615	0.204343		-0.66898	0.158556	0
CZ	4.054834	0	0.121971	0	0		0	0	7.391332
MX	0	0.065484	0	-0.23255	-0.02357		0.121761	-0.39603	0
MY	-1.23632	0	0.218307	0	0		0	0	-4.38243
MZ	0	-0.17864	0	-0.00251	0.088889		-0.13273	-0.00254	0

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.50 (FLEX)									
	ANGLEA	SIDES	ELEV	AILERON	RUDDER		ROLL	PITCH	YAW
CY	0	-0.55852	0	0.065407	0.203312		-0.67273	0.158708	0
CZ	4.247888	0	0.124951	0	0		0	0	7.641419
MX	0	0.065626	0	-0.2396	-0.02346		0.123773	-0.4092	0
MY	-1.31671	0	0.220393	0	0		0	0	-4.53948
MZ	0	-0.17964	0	-0.00321	0.089492		-0.13318	-0.0042	0

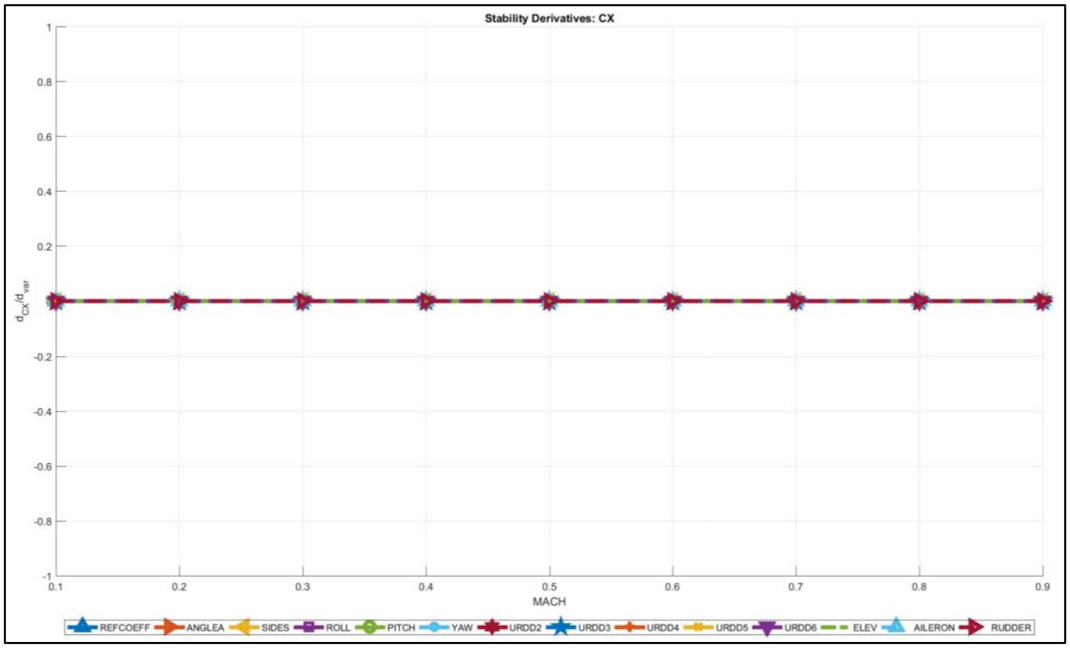
TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.60 (FLEX)								
	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	-0.56204	0	0.064218	0.201933	-0.67796	0.158812	0
CZ	4.521177	0	0.129197	0	0	0	0	7.988261
MX	0	0.065883	0	-0.24916	-0.02335	0.12655	-0.42704	0
MY	-1.43158	0	0.223031	0	0	0	0	-4.75956
MZ	0	-0.18098	0	-0.00417	0.090309	-0.13379	-0.00647	0

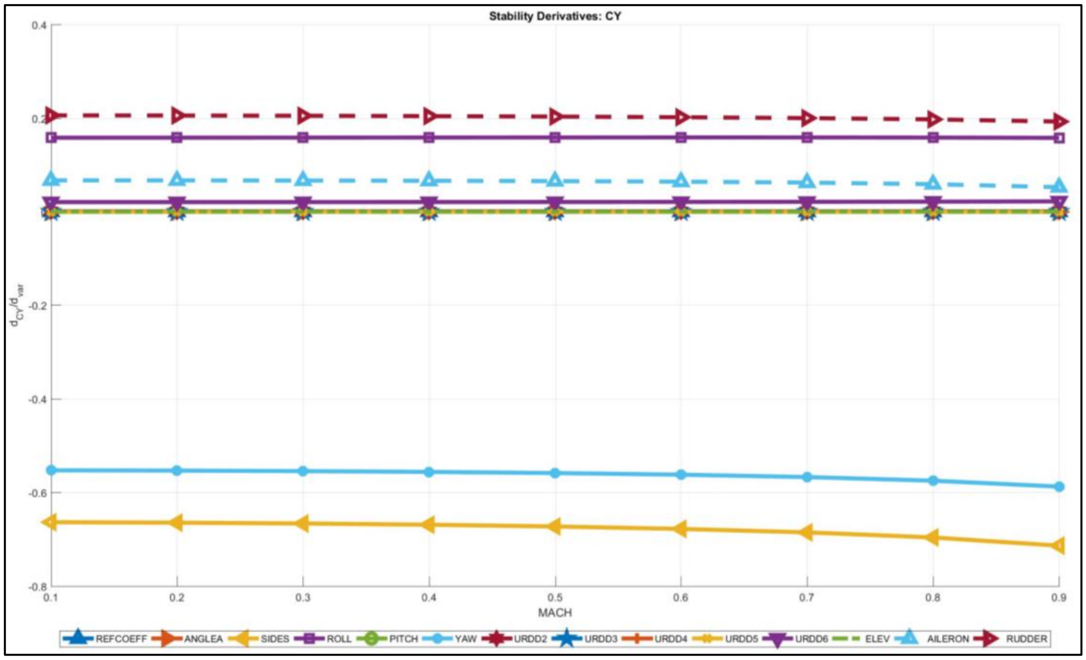
TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.70 (FLEX)								
	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	-0.56714	0	0.062255	0.200053	-0.68533	0.15877	0
CZ	4.914487	0	0.13528	0	0	0	0	8.472071
MX	0	0.066358	0	-0.26211	-0.02326	0.13042	-0.45121	0
MY	-1.59847	0	0.22632	0	0	0	0	-5.07076
MZ	0	-0.1828	0	-0.00552	0.091407	-0.13457	-0.00956	0

TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.80 (FLEX)								
	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	-0.57488	0	0.058836	0.19735	-0.69623	0.15847	0
CZ	5.512249	0	0.144476	0	0	0	0	9.172571
MX	0	0.067299	0	-0.28004	-0.02324	0.136032	-0.48469	0
MY	-1.85513	0	0.230341	0	0	0	0	-5.53153
MZ	0	-0.1853	0	-0.00743	0.092901	-0.13557	-0.01386	0

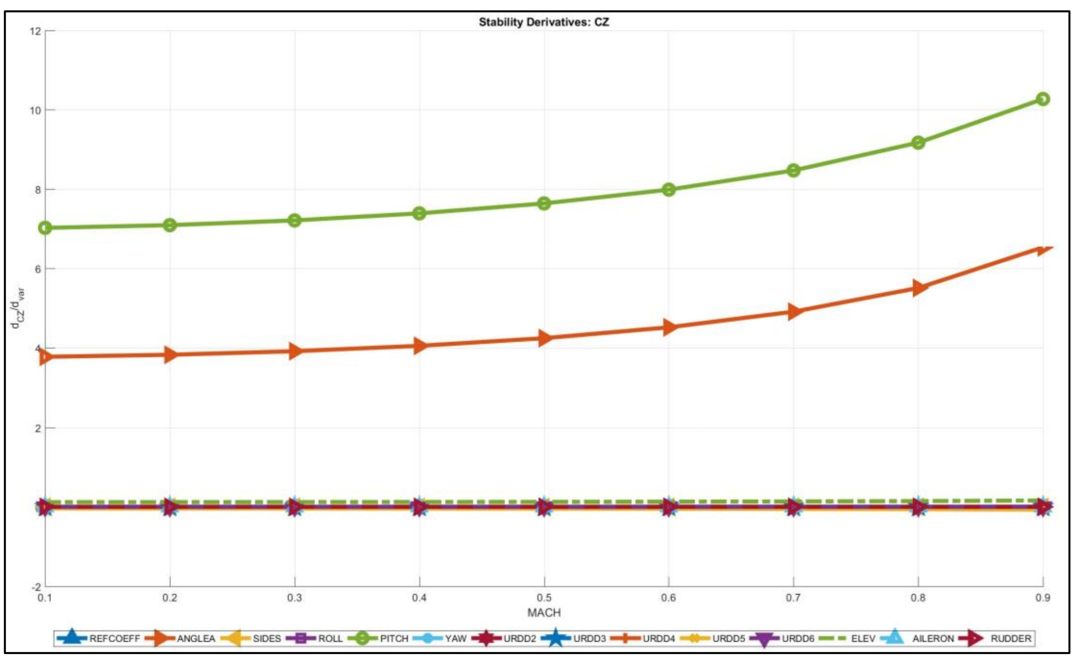
TABULATED STABILITY DERIVATIVE DATA FOR MACH 0.90(FLEX)								
	ANGLEA	SIDES	ELEV	AILERON	RUDDER	ROLL	PITCH	YAW
CY	0	-0.58751	0	0.052418	0.192658	-0.71377	0.157835	0
CZ	6.537208	0	0.15995	0	0	0	0	10.27152
MX	0	0.069446	0	-0.30627	-0.02329	0.144997	-0.53358	0
MY	-2.30318	0	0.23522	0	0	0	0	-6.28797
MZ	0	-0.18893	0	-0.01025	0.094911	-0.13687	-0.02007	0

Appendix C
Flexible Stability Derivatives

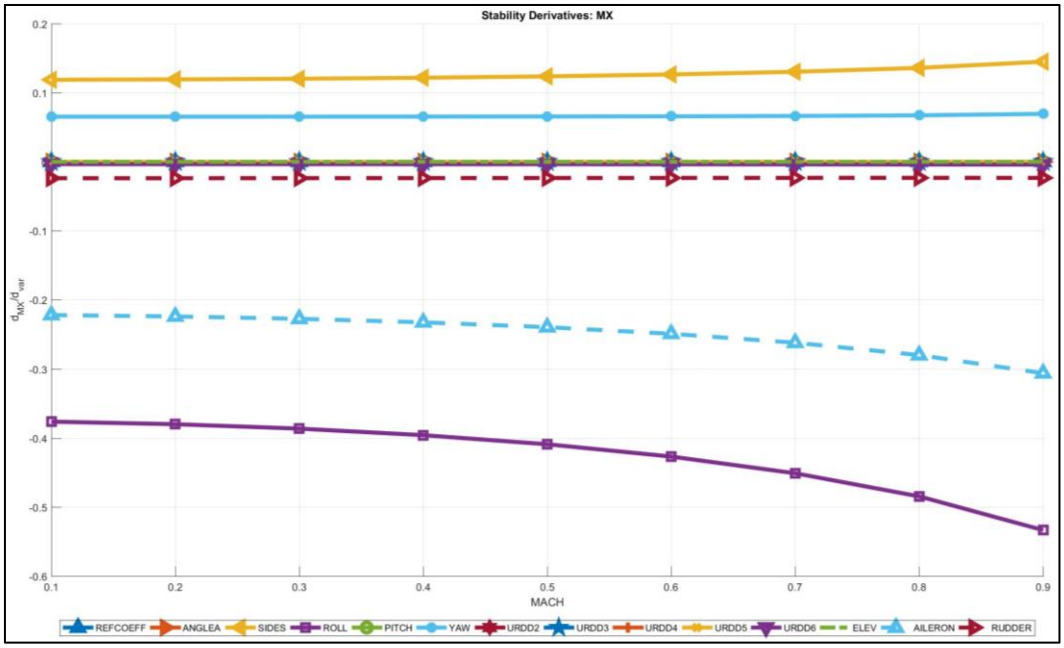




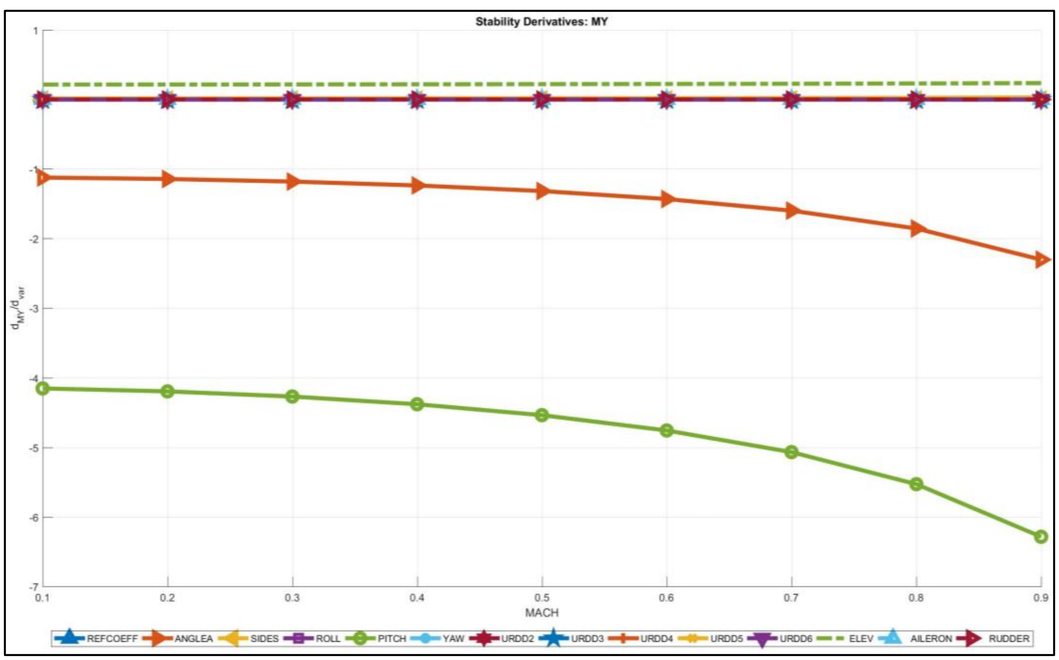
Stability Derivatives: CY
d_{CY}/d_{var}
MACH
REFCOEFF ANGLEA SIDES ROLL PITCH YAW URDD2 URDD3 URDD4 URDD5 URDD6 ELEV AILERON RUDDER



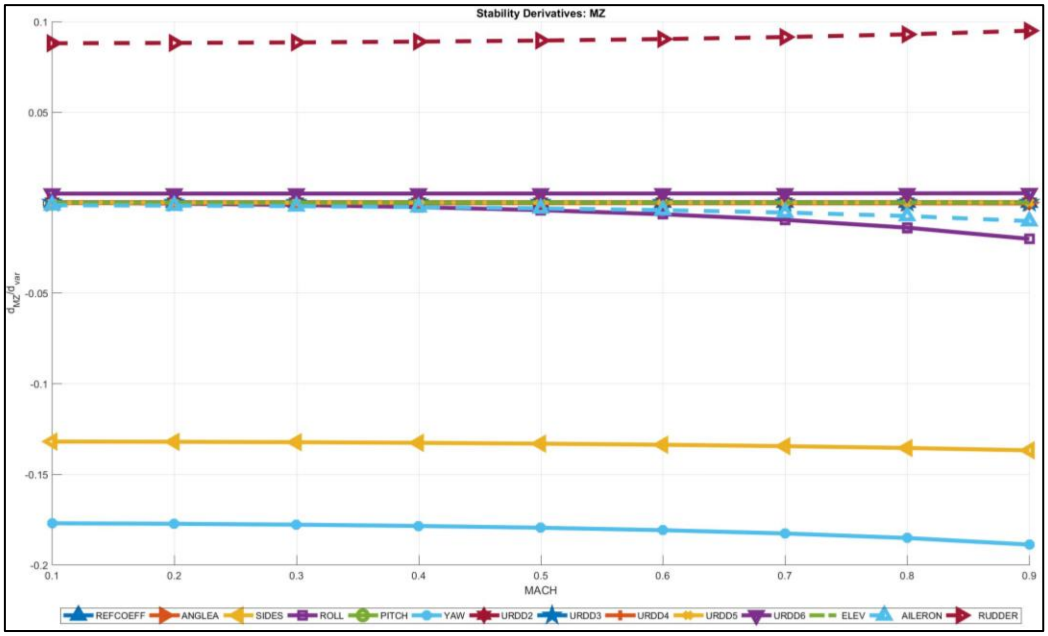
Stability Derivatives: CZ
d_{CZ}/d_{var}
MACH
REFCOEFF ANGLEA SIDES ROLL PITCH YAW URDD2 URDD3 URDD4 URDD5 URDD6 ELEV AILERON RUDDER



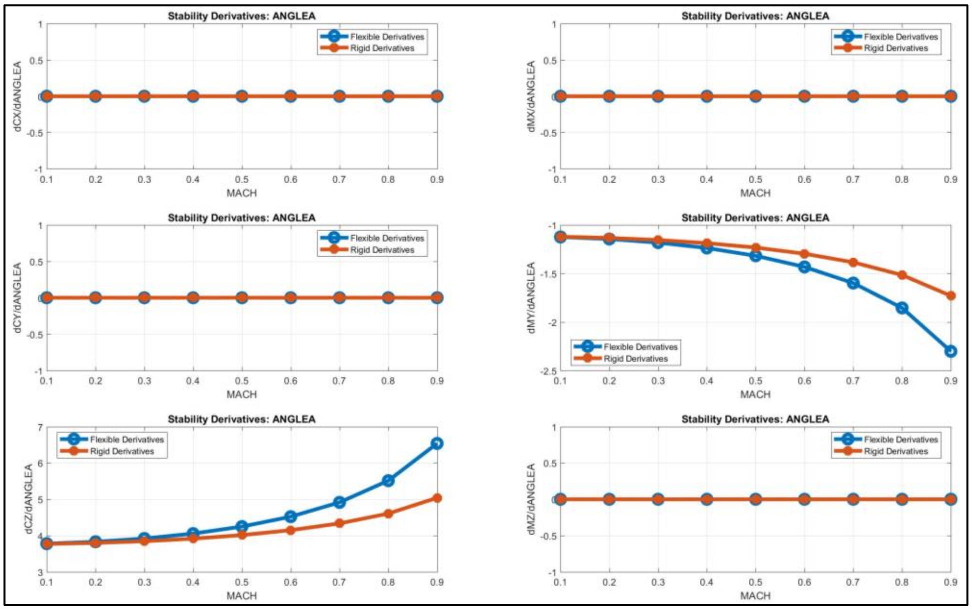
Stability Derivatives: MX
MACH
REFCOEFF
ANGLEA
SIDES
ROLL
PITCH
YAW
URDD2
URDD3
URDD4
URDD5
URDD6
ELEV
AILERON
RUDDER

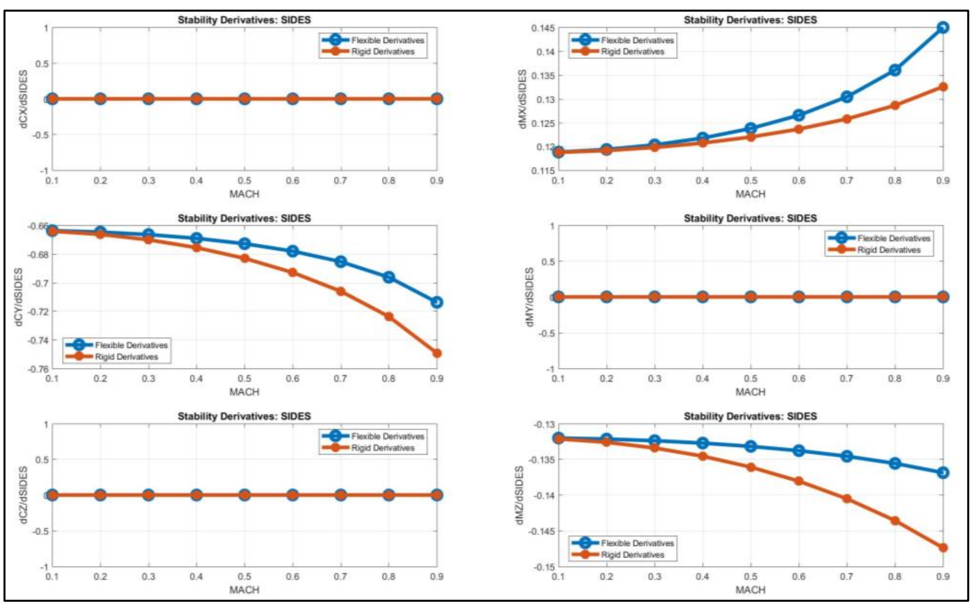


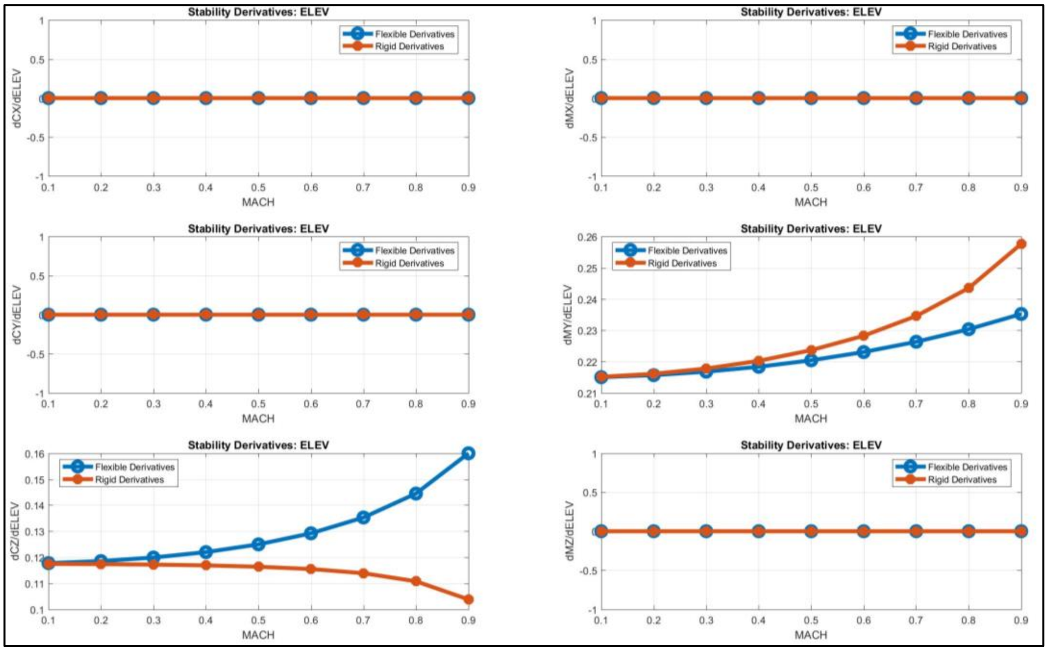
Stability Derivatives: MY
MACH
REFCOEFF
ANGLEA
SIDES
ROLL
PITCH
YAW
URDD2
URDD3
URDD4
URDD5
URDD6
ELEV
AILERON
RUDDER

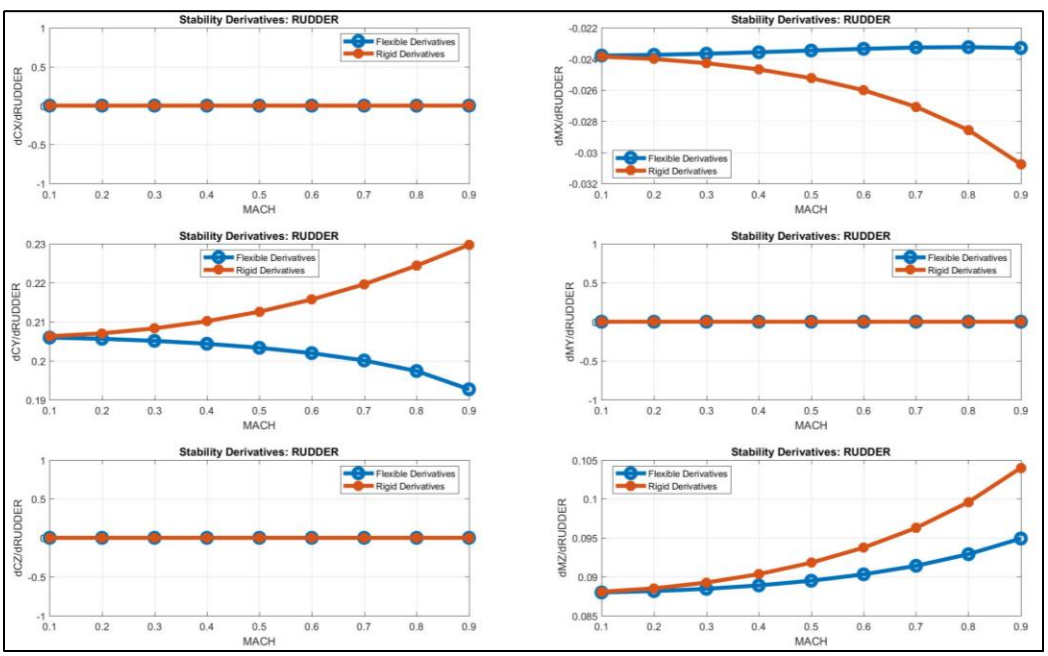

Stability Derivatives: MZ
0.1
0.05
0
-0.05
-0.1
-0.15
-0.2
d_{MZ}/d_{var}
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
MACH
REFCOEFF
ANGLEA
SIDES
ROLL
PITCH
YAW
URDD2
URDD3
URDD4
URDD5
URDD6
ELEV
AILERON
RUDDER

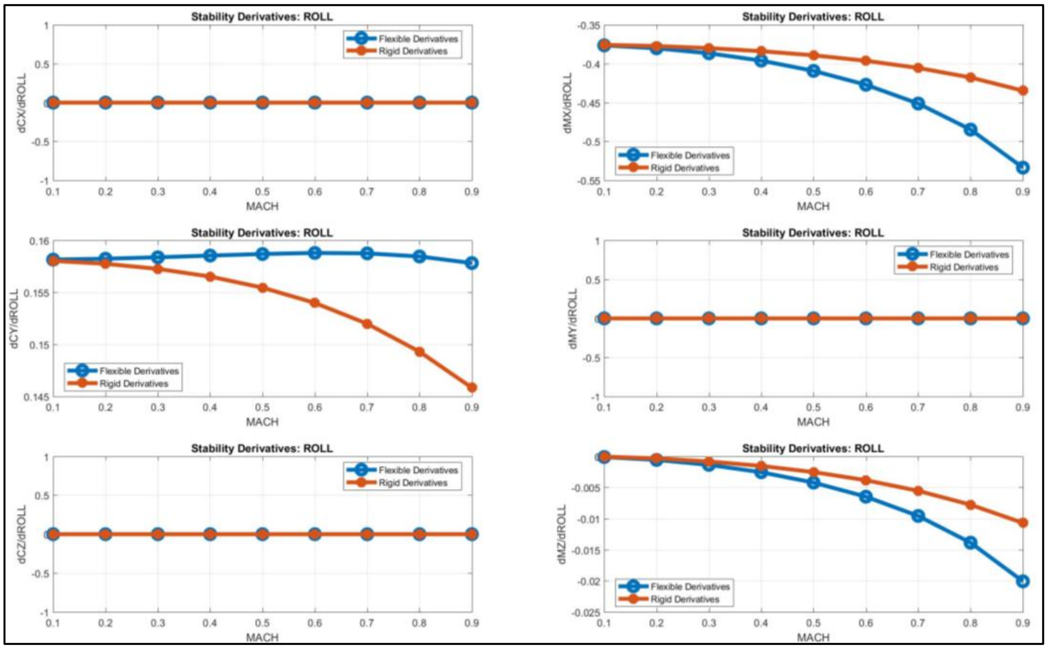
Appendix D
Flexible vs. Rigid Stability Derivatives

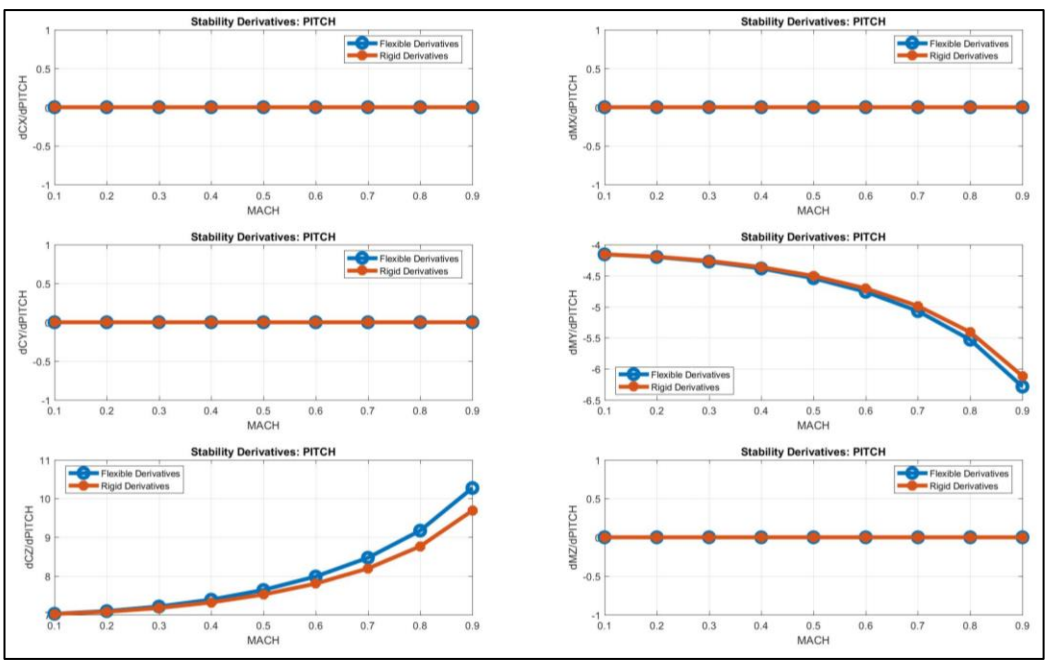


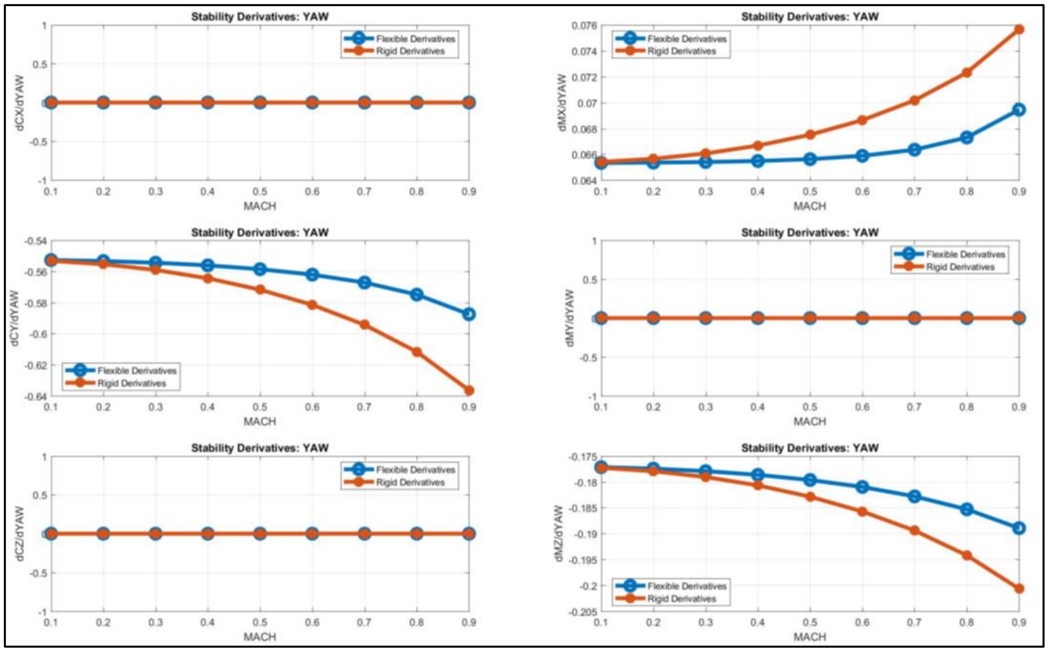



Stability Derivatives: ELEV
dCX/dELEV
dMX/dELEV
dCY/dELEV
dMY/dELEV
dCZ/dELEV
dMZ/dELEV
MACH
Flexible Derivatives
Rigid Derivatives

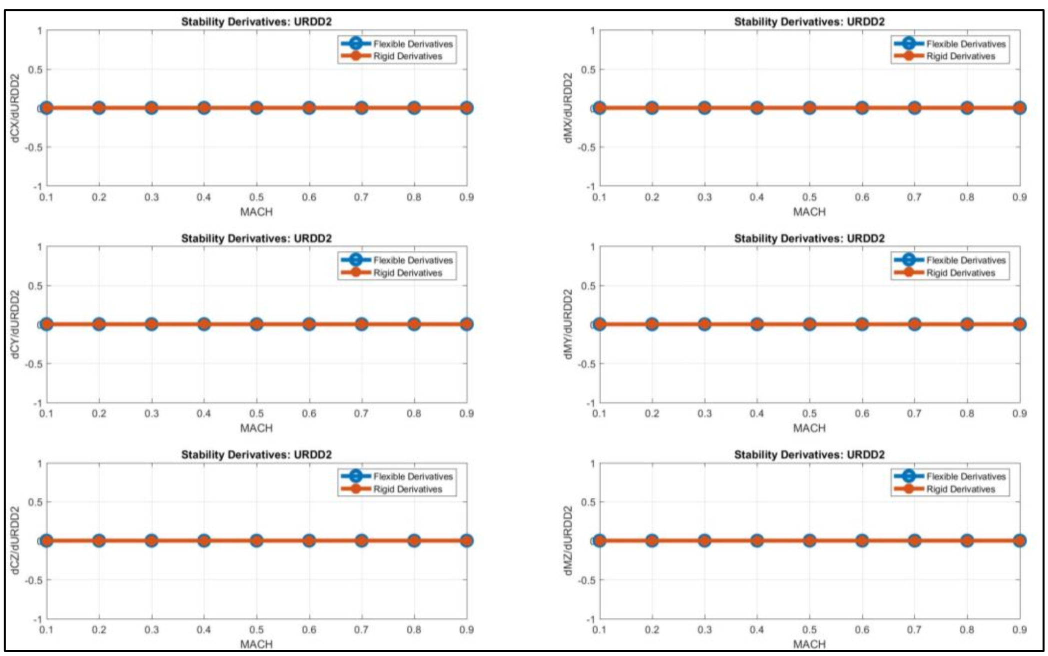

Stability Derivatives: RUDDER
dCX/dRUDDER
dMX/dRUDDER
dCY/dRUDDER
dMY/dRUDDER
dCZ/dRUDDER
dMZ/dRUDDER
MACH
Flexible Derivatives
Rigid Derivatives


Stability Derivatives: ROLL
dCX/dROLL
dMX/dROLL
dCY/dROLL
dMY/dROLL
dCZ/dROLL
dMZ/dROLL
MACH
Flexible Derivatives
Rigid Derivatives

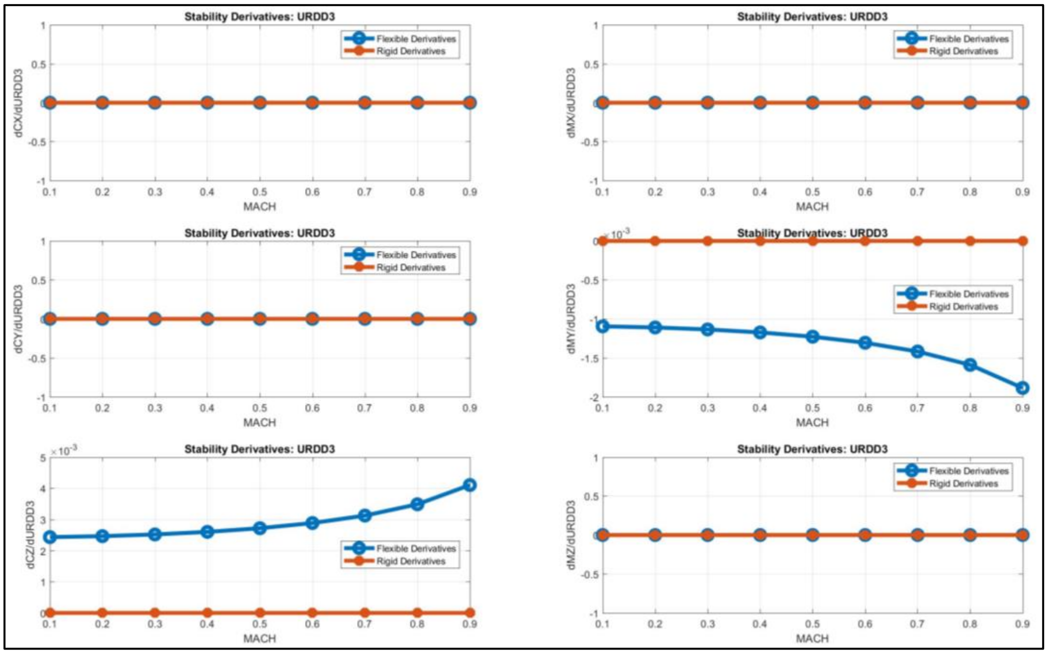

Stability Derivatives: PITCH
dCX/dPITCH
dMX/dPITCH
dCY/dPITCH
dMY/dPITCH
dCZ/dPITCH
dMZ/dPITCH
MACH
Flexible Derivatives
Rigid Derivatives

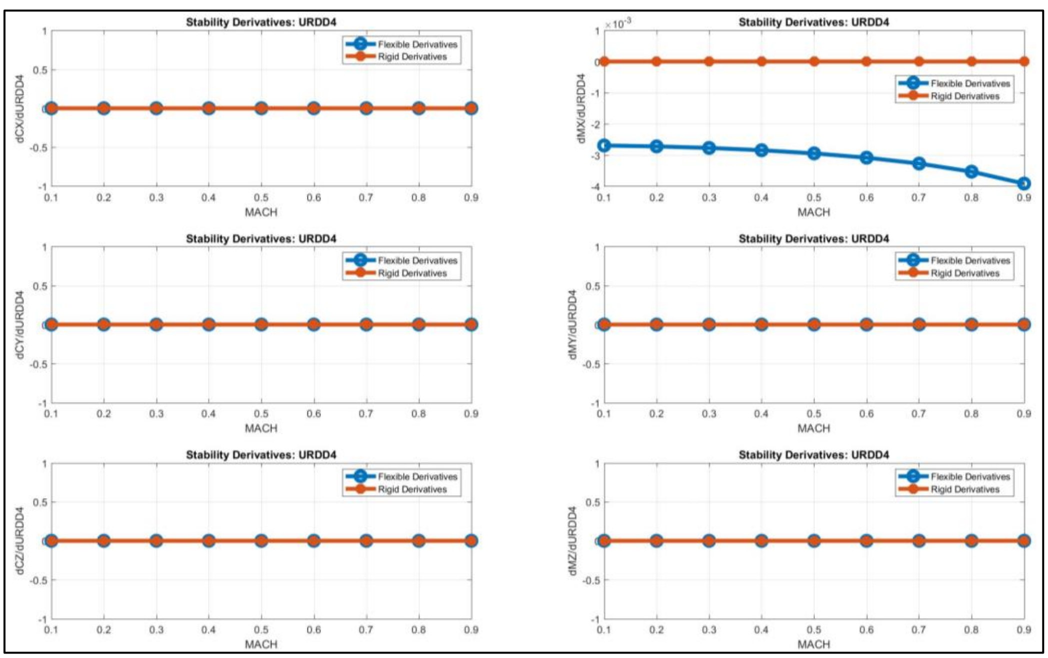


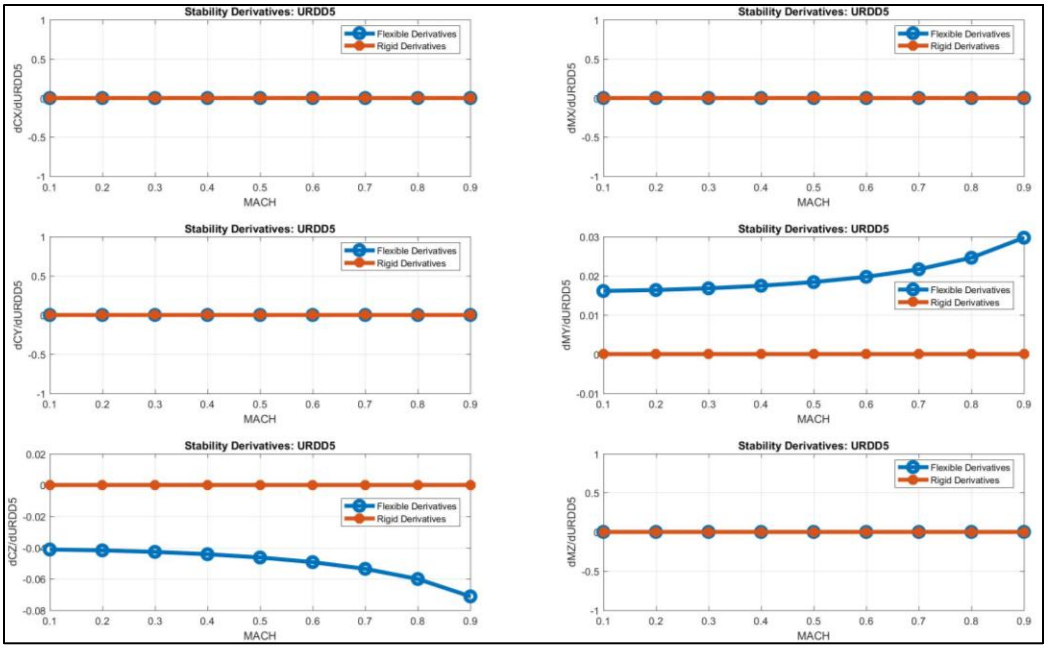
Stability Derivatives: YAW
dCX/dYAW
dMX/dYAW
dCY/dYAW
dMY/dYAW
dCZ/dYAW
dMZ/dYAW
MACH
Flexible Derivatives
Rigid Derivatives

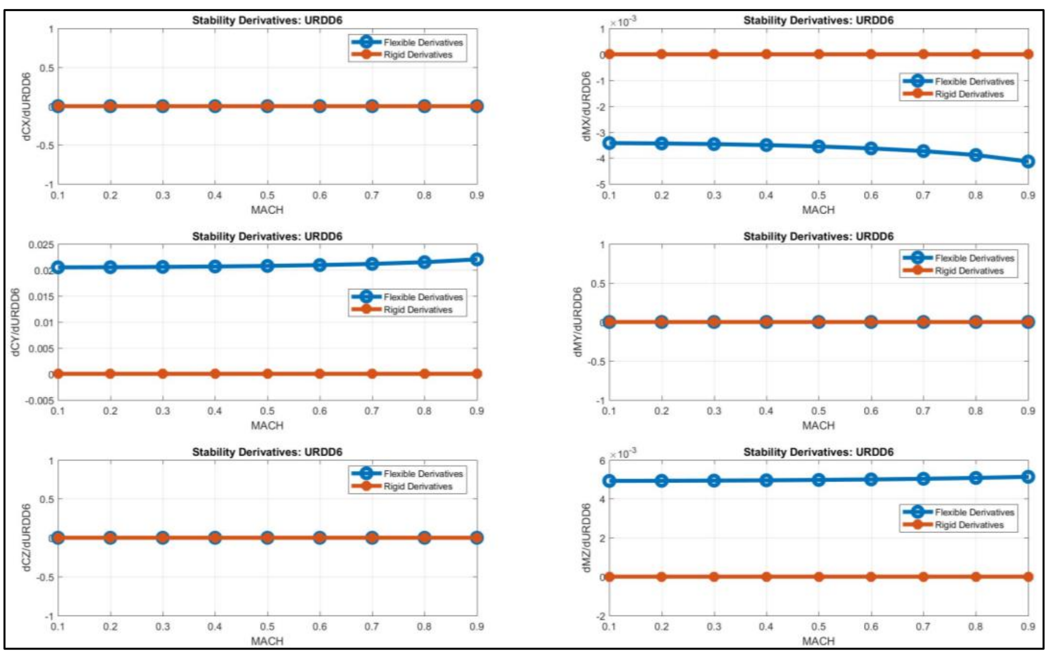


Stability Derivatives: URDD2
dCX/dURDD2
dMX/dURDD2
dCY/dURDD2
dMY/dURDD2
dCZ/dURDD2
dMZ/dURDD2
MACH
Flexible Derivatives
Rigid Derivatives


Stability Derivatives: URDD3
Flexible Derivatives
Rigid Derivatives
dCX/dURDD3
dCY/dURDD3
dCZ/dURDD3
dMX/dURDD3
dMY/dURDD3
dMZ/dURDD3
MACH


Stability Derivatives: URDD4
Flexible Derivatives
Rigid Derivatives
dCX/dURDD4
dCY/dURDD4
dCZ/dURDD4
dMX/dURDD4
dMY/dURDD4
dMZ/dURDD4
MACH


Stability Derivatives: URDD5
Flexible Derivatives
Rigid Derivatives
dCX/dURDD5
dCY/dURDD5
dCZ/dURDD5
dMX/dURDD5
dMY/dURDD5
dMZ/dURDD5
MACH


Stability Derivatives: URDD6
Flexible Derivatives
Rigid Derivatives
dCX/dURDD6
dCY/dURDD6
dCZ/dURDD6
dMX/dURDD6
dMY/dURDD6
dMZ/dURDD6
MACH

Appendix E
Unit Hinge Moment Database Per Mach and Variable

Mach	RATE VAR	Aileron Hinge Moment/(qbar* URDD2)	Elevator Hinge Moment/(qbar* URDD2)	Rudder Hinge Moment/(qbar* URDD2)
0.1	**URDD2**	113.8882	-187.5346	280.9917
0.2	**URDD2**	28.6236	-46.8023	70.6559
0.3	**URDD2**	12.8390	-20.7369	31.7138
0.4	**URDD2**	7.3203	-11.6080	18.0951
0.5	**URDD2**	4.7734	-7.3751	11.8064
0.6	**URDD2**	3.4001	-5.0662	8.4088
0.7	**URDD2**	2.5867	-3.6610	6.3841
0.8	**URDD2**	2.0818	-2.7296	5.1045
0.9	**URDD2**	1.7805	-2.0599	4.2906
Mach	**RATE VAR**	**Aileron Hinge Moment/(qbar* URDD3)**	**Elevator Hinge Moment/(qbar* URDD3)**	**Rudder Hinge Moment/(qbar* URDD3)**
0.1	**URDD3**	8.9319	514.5834	0.0000
0.2	**URDD3**	2.2069	128.9061	0.0000
0.3	**URDD3**	0.9610	57.4937	0.0000
0.4	**URDD3**	0.5241	32.5052	0.0000
0.5	**URDD3**	0.3209	20.9497	0.0000
0.6	**URDD3**	0.2091	14.6867	0.0000
0.7	**URDD3**	0.1398	10.9262	0.0000
0.8	**URDD3**	0.0915	8.5083	0.0000
0.9	**URDD3**	0.0514	6.8894	0.0000
Mach	**RATE VAR**	**Aileron Hinge Moment/(qbar* PDOT)**	**Elevator Hinge Moment/(qbar* PDOT)**	**Rudder Hinge Moment/(qbar* PDOT)**
0.1	**PDOT**	4.4725	-0.5966	1.0783
0.2	**PDOT**	1.1229	-0.1489	0.2707
0.3	**PDOT**	0.5027	-0.0659	0.1212
0.4	**PDOT**	0.2859	-0.0369	0.0689
0.5	**PDOT**	0.1858	-0.0233	0.0447
0.6	**PDOT**	0.1317	-0.0159	0.0316
0.7	**PDOT**	0.0996	-0.0114	0.0237
0.8	**PDOT**	0.0795	-0.0082	0.0186
0.9	**PDOT**	0.0670	-0.0058	0.0151

Mach	RATE VAR	Aileron Hinge Moment/(qbar* QDOT)	Elevator Hinge Moment/(qbar* QDOT)	Rudder Hinge Moment/(qbar* QDOT)
0.1	**QDOT**	-0.1444	40.4923	0.0000
0.2	**QDOT**	-0.0361	10.1293	0.0000
0.3	**QDOT**	-0.0161	4.5065	0.0000
0.4	**QDOT**	-0.0091	2.5388	0.0000
0.5	**QDOT**	-0.0060	1.6282	0.0000
0.6	**QDOT**	-0.0043	1.1338	0.0000
0.7	**QDOT**	-0.0034	0.8359	0.0000
0.8	**QDOT**	-0.0030	0.6429	0.0000
0.9	**QDOT**	-0.0032	0.5105	0.0000
Mach	**RATE VAR**	**Aileron Hinge Moment/(qbar* RDOT)**	**Elevator Hinge Moment/(qbar* RDOT)**	**Rudder Hinge Moment(/qbar* RDOT)**
0.1	**RDOT**	-4.8396	16.3457	-53.2556
0.2	**RDOT**	-1.2192	4.0721	-13.3706
0.3	**RDOT**	-0.5489	1.7987	-5.9859
0.4	**RDOT**	-0.3145	1.0024	-3.4029
0.5	**RDOT**	-0.2061	0.6329	-2.2097
0.6	**RDOT**	-0.1474	0.4311	-1.5645
0.7	**RDOT**	-0.1124	0.3079	-1.1795
0.8	**RDOT**	-0.0903	0.2259	-0.9354
0.9	**RDOT**	-0.0764	0.1662	-0.7791
Mach	**RATE VAR**	**Aileron Hinge Moment/(qbar* P)**	**Elevator Hinge Moment/(qbar* P)**	**Rudder Hinge Moment/(qbar* P)**
0.1	**P**	0.1861	-0.0574	-0.0127
0.2	**P**	0.0941	-0.0287	-0.0066
0.3	**P**	0.0640	-0.0192	-0.0046
0.4	**P**	0.0494	-0.0144	-0.0037
0.5	**P**	0.0411	-0.0116	-0.0032
0.6	**P**	0.0361	-0.0097	-0.0030
0.7	**P**	0.0331	-0.0083	-0.0030
0.8	**P**	0.0319	-0.0072	-0.0031
0.9	**P**	0.0325	-0.0063	-0.0036

Mach	RATE VAR	Aileron Hinge Moment/(qbar* Q)	Elevator Hinge Moment/(qbar* Q)	Rudder Hinge Moment/(qbar* Q)
0.1	Q	0.0187	0.4498	0.0000
0.2	Q	0.0095	0.2267	0.0000
0.3	Q	0.0065	0.1533	0.0000
0.4	Q	0.0051	0.1174	0.0000
0.5	Q	0.0043	0.0968	0.0000
0.6	Q	0.0039	0.0840	0.0000
0.7	Q	0.0037	0.0762	0.0000
0.8	Q	0.0037	0.0725	0.0000
0.9	Q	0.0040	0.0736	0.0000
Mach	RATE VAR	Aileron Hinge Moment/(qbar* R)	Elevator Hinge Moment/(qbar* R)	Rudder Hinge Moment/(qbar* R)
0.1	R	0.0130	-0.0485	-0.0871
0.2	R	0.0066	-0.0244	-0.0438
0.3	R	0.0045	-0.0165	-0.0296
0.4	R	0.0035	-0.0126	-0.0226
0.5	R	0.0030	-0.0104	-0.0185
0.6	R	0.0027	-0.0090	-0.0160
0.7	R	0.0025	-0.0080	-0.0143
0.8	R	0.0025	-0.0074	-0.0133
0.9	R	0.0027	-0.0070	-0.0129

Appendix F
Proof of Concept Test Matrices

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
1	Abrupt Pitch Initiation	0.15	0	0	1	0	35.8	0	0	0	0
2	Abrupt Pitch Initiation	0.23	0	0	1	0	35.8	0	0	0	0
3	Abrupt Pitch Initiation	0.38	0	0	1	0	35.8	0	0	0	0
4	Abrupt Pitch Initiation	0.51	0	0	1	0	35.8	0	0	0	0
5	Abrupt Pitch Initiation	0.60	0	0	1	0	35.8	0	0	0	0
6	Abrupt Pitch Initiation	0.64	0	0	1	0	35.8	0	0	0	0
7	Abrupt Pitch Initiation	0.89	0	0	1	0	35.8	0	0	0	0
8	Abrupt Pitch Initiation	0.15	0	0	2	0	35.8	0	0	0	0
9	Abrupt Pitch Initiation	0.23	0	0	2	0	35.8	0	0	0	0
10	Abrupt Pitch Initiation	0.38	0	0	2	0	35.8	0	0	0	0
11	Abrupt Pitch Initiation	0.51	0	0	2	0	35.8	0	0	0	0
12	Abrupt Pitch Initiation	0.60	0	0	2	0	35.8	0	0	0	0
13	Abrupt Pitch Initiation	0.64	0	0	2	0	35.8	0	0	0	0
14	Abrupt Pitch Initiation	0.89	0	0	2	0	35.8	0	0	0	0
15	Abrupt Pitch Initiation	0.15	0	0	1	0	-35.8	0	0	0	0
16	Abrupt Pitch Initiation	0.23	0	0	1	0	-35.8	0	0	0	0
17	Abrupt Pitch Initiation	0.38	0	0	1	0	-35.8	0	0	0	0
18	Abrupt Pitch Initiation	0.51	0	0	1	0	-35.8	0	0	0	0
19	Abrupt Pitch Initiation	0.60	0	0	1	0	-35.8	0	0	0	0
20	Abrupt Pitch Initiation	0.64	0	0	1	0	-35.8	0	0	0	0
21	Abrupt Pitch Initiation	0.89	0	0	1	0	-35.8	0	0	0	0
22	Abrupt Pitch Initiation	0.15	0	0	2	0	-35.8	0	0	0	0
23	Abrupt Pitch Initiation	0.23	0	0	2	0	-35.8	0	0	0	0
24	Abrupt Pitch Initiation	0.38	0	0	2	0	-35.8	0	0	0	0
25	Abrupt Pitch Initiation	0.51	0	0	2	0	-35.8	0	0	0	0
26	Abrupt Pitch Initiation	0.60	0	0	2	0	-35.8	0	0	0	0
27	Abrupt Pitch Initiation	0.64	0	0	2	0	-35.8	0	0	0	0
28	Abrupt Pitch Initiation	0.89	0	0	2	0	-35.8	0	0	0	0
29	Abrupt Pitch Initiation	0.15	0	0	-1	0	35.8	0	0	0	0
30	Abrupt Pitch Initiation	0.23	0	0	-1	0	35.8	0	0	0	0
31	Abrupt Pitch Initiation	0.38	0	0	-1	0	35.8	0	0	0	0
32	Abrupt Pitch Initiation	0.51	0	0	-1	0	35.8	0	0	0	0
33	Abrupt Pitch Initiation	0.60	0	0	-1	0	35.8	0	0	0	0
34	Abrupt Pitch Initiation	0.64	0	0	-1	0	35.8	0	0	0	0
35	Abrupt Pitch Initiation	0.89	0	0	-1	0	35.8	0	0	0	0
36	Abrupt Pitch Initiation	0.15	0	0	-2	0	35.8	0	0	0	0
37	Abrupt Pitch Initiation	0.23	0	0	-2	0	35.8	0	0	0	0
38	Abrupt Pitch Initiation	0.38	0	0	-2	0	35.8	0	0	0	0
39	Abrupt Pitch Initiation	0.51	0	0	-2	0	35.8	0	0	0	0
40	Abrupt Pitch Initiation	0.60	0	0	-2	0	35.8	0	0	0	0
41	Abrupt Pitch Initiation	0.64	0	0	-2	0	35.8	0	0	0	0
42	Abrupt Pitch Initiation	0.89	0	0	-2	0	35.8	0	0	0	0
43	Abrupt Pitch Initiation	0.15	0	0	-1	0	-35.8	0	0	0	0
44	Abrupt Pitch Initiation	0.23	0	0	-1	0	-35.8	0	0	0	0
45	Abrupt Pitch Initiation	0.38	0	0	-1	0	-35.8	0	0	0	0
46	Abrupt Pitch Initiation	0.51	0	0	-1	0	-35.8	0	0	0	0
47	Abrupt Pitch Initiation	0.60	0	0	-1	0	-35.8	0	0	0	0
48	Abrupt Pitch Initiation	0.64	0	0	-1	0	-35.8	0	0	0	0
49	Abrupt Pitch Initiation	0.89	0	0	-1	0	-35.8	0	0	0	0
50	Abrupt Pitch Initiation	0.15	0	0	-2	0	-35.8	0	0	0	0
51	Abrupt Pitch Initiation	0.23	0	0	-2	0	-35.8	0	0	0	0
52	Abrupt Pitch Initiation	0.38	0	0	-2	0	-35.8	0	0	0	0
53	Abrupt Pitch Initiation	0.51	0	0	-2	0	-35.8	0	0	0	0
54	Abrupt Pitch Initiation	0.60	0	0	-2	0	-35.8	0	0	0	0
55	Abrupt Pitch Initiation	0.64	0	0	-2	0	-35.8	0	0	0	0
56	Abrupt Pitch Initiation	0.89	0	0	-2	0	-35.8	0	0	0	0

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
1	Abrupt Pitch Initiation	55	29111	0	55	29082	0	0	-29	0	-0.17%	0.10%	0.73%
2	Abrupt Pitch Initiation	53	29138	0	54	29139	0	0	-1	0	-0.25%	0.00%	0.51%
3	Abrupt Pitch Initiation	48	29228	0	48	29224	0	0	4	0	-0.38%	0.01%	0.57%
4	Abrupt Pitch Initiation	39	29359	0	39	29358	0	0	1	0	-0.44%	0.00%	0.35%
5	Abrupt Pitch Initiation	29	29479	0	29	29480	0	0	-1	0	0.00%	0.00%	-0.16%
6	Abrupt Pitch Initiation	23	29549	0	23	29545	0	-1	4	0	-4.07%	0.01%	1.01%
7	Abrupt Pitch Initiation	-70	30165	0	-68	30161	0	-3	3	0	4.15%	0.01%	1.16%
8	Abrupt Pitch Initiation	186	36742	0	187	36704	0	0	37	0	-0.06%	0.10%	1.71%
9	Abrupt Pitch Initiation	184	36783	0	184	36783	0	0	0	0	-0.14%	0.00%	1.15%
10	Abrupt Pitch Initiation	173	36923	0	173	36917	0	0	6	0	-0.17%	0.02%	0.95%
11	Abrupt Pitch Initiation	157	37125	0	157	37123	0	0	2	0	-0.17%	0.01%	0.52%
12	Abrupt Pitch Initiation	141	37311	0	141	37312	0	0	-1	0	0.00%	0.00%	-0.16%
13	Abrupt Pitch Initiation	130	37420	0	131	37414	0	-1	7	0	-1.04%	0.02%	1.21%
14	Abrupt Pitch Initiation	-6	38411	0	-2	38404	0	-4	7	0	149.46%	0.02%	1.17%
15	Abrupt Pitch Initiation	208	-13850	0	208	-13837	0	0	-13	0	0.03%	0.09%	-1.94%
16	Abrupt Pitch Initiation	207	-13848	0	207	-13849	0	0	1	0	-0.05%	-0.01%	-2.10%
17	Abrupt Pitch Initiation	203	-13839	0	203	-13838	0	0	-1	0	-0.03%	0.01%	-43.37%
18	Abrupt Pitch Initiation	198	-13827	0	198	-13828	0	0	0	0	0.00%	0.00%	1.54%
19	Abrupt Pitch Initiation	194	-13814	0	194	-13814	0	0	0	0	0.00%	0.00%	-0.20%
20	Abrupt Pitch Initiation	192	-13805	0	192	-13807	0	0	2	0	0.07%	-0.01%	1.91%
21	Abrupt Pitch Initiation	199	-13672	0	198	-13676	0	1	5	0	0.53%	-0.04%	1.17%
22	Abrupt Pitch Initiation	340	-6220	0	340	-6215	0	0	-4	0	0.02%	0.07%	-3.82%
23	Abrupt Pitch Initiation	337	-6203	0	337	-6205	0	0	2	0	-0.07%	-0.03%	-5.53%
24	Abrupt Pitch Initiation	328	-6145	0	328	-6146	0	0	1	0	-0.05%	-0.01%	4.28%
25	Abrupt Pitch Initiation	316	-6062	0	316	-6062	0	0	1	0	-0.03%	-0.01%	1.05%
26	Abrupt Pitch Initiation	305	-5982	0	305	-5982	0	0	0	0	0.00%	0.00%	-0.19%
27	Abrupt Pitch Initiation	300	-5934	0	300	-5939	0	0	5	0	-0.09%	-0.09%	1.66%
28	Abrupt Pitch Initiation	263	-5425	0	263	-5434	0	0	9	0	0.07%	-0.17%	1.17%
29	Abrupt Pitch Initiation	-208	13850	0	-208	13838	0	0	12	0	0.03%	0.09%	-1.95%
30	Abrupt Pitch Initiation	-207	13848	0	-207	13850	0	0	-2	0	-0.05%	-0.01%	-2.10%
31	Abrupt Pitch Initiation	-203	13839	0	-203	13839	0	0	1	0	-0.03%	0.00%	-43.40%
32	Abrupt Pitch Initiation	-198	13827	0	-198	13828	0	0	-1	0	0.00%	0.00%	1.54%
33	Abrupt Pitch Initiation	-194	13814	0	-194	13815	0	0	0	0	0.00%	0.00%	-0.20%
34	Abrupt Pitch Initiation	-192	13805	0	-192	13808	0	0	-2	0	0.07%	-0.02%	1.91%
35	Abrupt Pitch Initiation	-199	13672	0	-198	13677	0	-1	-5	0	0.53%	-0.04%	1.17%
36	Abrupt Pitch Initiation	-340	6220	0	-340	6216	0	0	4	0	0.02%	0.06%	-3.83%
37	Abrupt Pitch Initiation	-337	6203	0	-337	6205	0	0	-2	0	-0.07%	-0.04%	-5.53%
38	Abrupt Pitch Initiation	-328	6145	0	-328	6146	0	0	-1	0	-0.05%	-0.02%	4.28%
39	Abrupt Pitch Initiation	-316	6062	0	-316	6063	0	0	-1	0	-0.03%	-0.02%	1.05%
40	Abrupt Pitch Initiation	-305	5982	0	-305	5982	0	0	0	0	0.00%	0.00%	-0.19%
41	Abrupt Pitch Initiation	-300	5934	0	-300	5939	0	0	-6	0	-0.09%	-0.09%	1.66%
42	Abrupt Pitch Initiation	-263	5425	0	-263	5434	0	0	-9	0	0.07%	-0.17%	1.17%
43	Abrupt Pitch Initiation	-55	-29111	0	-55	-29082	0	0	-29	0	-0.18%	0.10%	0.73%
44	Abrupt Pitch Initiation	-53	-29138	0	-54	-29138	0	0	0	0	-0.25%	0.00%	0.52%
45	Abrupt Pitch Initiation	-48	-29228	0	-48	-29224	0	0	-5	0	-0.38%	0.02%	0.57%
46	Abrupt Pitch Initiation	-39	-29359	0	-39	-29358	0	0	-2	0	-0.44%	0.01%	0.35%
47	Abrupt Pitch Initiation	-29	-29479	0	-29	-29479	0	0	0	0	-0.01%	0.00%	-0.16%
48	Abrupt Pitch Initiation	-23	-29549	0	-23	-29545	0	1	-4	0	-4.07%	0.01%	1.01%
49	Abrupt Pitch Initiation	70	-30165	0	68	-30161	0	3	-4	0	4.15%	0.01%	1.16%
50	Abrupt Pitch Initiation	-186	-36742	0	-187	-36704	0	0	-38	0	-0.06%	0.10%	1.71%
51	Abrupt Pitch Initiation	-184	-36783	0	-184	-36783	0	0	0	0	-0.14%	0.00%	1.15%
52	Abrupt Pitch Initiation	-173	-36923	0	-173	-36916	0	0	-7	0	-0.17%	0.02%	0.95%
53	Abrupt Pitch Initiation	-157	-37125	0	-157	-37123	0	0	-2	0	-0.17%	0.01%	0.52%
54	Abrupt Pitch Initiation	-141	-37311	0	-141	-37312	0	0	1	0	0.00%	0.00%	-0.16%
55	Abrupt Pitch Initiation	-130	-37420	0	-131	-37413	0	1	-7	0	-1.04%	0.02%	1.21%
56	Abrupt Pitch Initiation	6	-38411	0	2	-38403	0	4	-8	0	149.74%	0.02%	1.17%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
113	**Abrupt Pitch Termination**	0.15	0	0	1.0	0	-35.8	0	0	16.826	0
114	**Abrupt Pitch Termination**	0.23	0	0	1.0	0	-35.8	0	0	16.826	0
115	**Abrupt Pitch Termination**	0.38	0	0	1.0	0	-35.8	0	0	16.826	0
116	**Abrupt Pitch Termination**	0.51	0	0	1.0	0	-35.8	0	0	16.826	0
117	**Abrupt Pitch Termination**	0.60	0	0	1.0	0	-35.8	0	0	16.826	0
118	**Abrupt Pitch Termination**	0.64	0	0	1.0	0	-35.8	0	0	16.826	0
119	**Abrupt Pitch Termination**	0.89	0	0	1.0	0	-35.8	0	0	16.826	0
120	**Abrupt Pitch Termination**	0.15	0	0	2.0	0	-35.8	0	0	16.826	0
121	**Abrupt Pitch Termination**	0.23	0	0	2.0	0	-35.8	0	0	16.826	0
122	**Abrupt Pitch Termination**	0.38	0	0	2.0	0	-35.8	0	0	16.826	0
123	**Abrupt Pitch Termination**	0.51	0	0	2.0	0	-35.8	0	0	16.826	0
124	**Abrupt Pitch Termination**	0.60	0	0	2.0	0	-35.8	0	0	16.826	0
125	**Abrupt Pitch Termination**	0.64	0	0	2.0	0	-35.8	0	0	16.826	0
126	**Abrupt Pitch Termination**	0.89	0	0	2.0	0	-35.8	0	0	16.826	0
127	**Abrupt Pitch Termination**	0.15	0	0	1.0	0	35.8	0	0	-16.826	0
128	**Abrupt Pitch Termination**	0.23	0	0	1.0	0	35.8	0	0	-16.826	0
129	**Abrupt Pitch Termination**	0.38	0	0	1.0	0	35.8	0	0	-16.826	0
130	**Abrupt Pitch Termination**	0.51	0	0	1.0	0	35.8	0	0	-16.826	0
131	**Abrupt Pitch Termination**	0.60	0	0	1.0	0	35.8	0	0	-16.826	0
132	**Abrupt Pitch Termination**	0.64	0	0	1.0	0	35.8	0	0	-16.826	0
133	**Abrupt Pitch Termination**	0.89	0	0	1.0	0	35.8	0	0	-16.826	0
134	**Abrupt Pitch Termination**	0.15	0	0	2.0	0	35.8	0	0	-16.826	0
135	**Abrupt Pitch Termination**	0.23	0	0	2.0	0	35.8	0	0	-16.826	0
136	**Abrupt Pitch Termination**	0.38	0	0	2.0	0	35.8	0	0	-16.826	0
137	**Abrupt Pitch Termination**	0.51	0	0	2.0	0	35.8	0	0	-16.826	0
138	**Abrupt Pitch Termination**	0.60	0	0	2.0	0	35.8	0	0	-16.826	0
139	**Abrupt Pitch Termination**	0.64	0	0	2.0	0	35.8	0	0	-16.826	0
140	**Abrupt Pitch Termination**	0.89	0	0	2.0	0	35.8	0	0	-16.826	0
141	**Abrupt Pitch Termination**	0.15	0	0	-1.0	0	-35.8	0	0	16.826	0
142	**Abrupt Pitch Termination**	0.23	0	0	-1.0	0	-35.8	0	0	16.826	0
143	**Abrupt Pitch Termination**	0.38	0	0	-1.0	0	-35.8	0	0	16.826	0
144	**Abrupt Pitch Termination**	0.51	0	0	-1.0	0	-35.8	0	0	16.826	0
145	**Abrupt Pitch Termination**	0.60	0	0	-1.0	0	-35.8	0	0	16.826	0
146	**Abrupt Pitch Termination**	0.64	0	0	-1.0	0	-35.8	0	0	16.826	0
147	**Abrupt Pitch Termination**	0.89	0	0	-1.0	0	-35.8	0	0	16.826	0
148	**Abrupt Pitch Termination**	0.15	0	0	-2.0	0	-35.8	0	0	16.826	0
149	**Abrupt Pitch Termination**	0.23	0	0	-2.0	0	-35.8	0	0	16.826	0
150	**Abrupt Pitch Termination**	0.38	0	0	-2.0	0	-35.8	0	0	16.826	0
151	**Abrupt Pitch Termination**	0.51	0	0	-2.0	0	-35.8	0	0	16.826	0
152	**Abrupt Pitch Termination**	0.60	0	0	-2.0	0	-35.8	0	0	16.826	0
153	**Abrupt Pitch Termination**	0.64	0	0	-2.0	0	-35.8	0	0	16.826	0
154	**Abrupt Pitch Termination**	0.89	0	0	-2.0	0	-35.8	0	0	16.826	0
155	**Abrupt Pitch Termination**	0.15	0	0	-1.0	0	35.8	0	0	-16.826	0
156	**Abrupt Pitch Termination**	0.23	0	0	-1.0	0	35.8	0	0	-16.826	0
157	**Abrupt Pitch Termination**	0.38	0	0	-1.0	0	35.8	0	0	-16.826	0
158	**Abrupt Pitch Termination**	0.51	0	0	-1.0	0	35.8	0	0	-16.826	0
159	**Abrupt Pitch Termination**	0.60	0	0	-1.0	0	35.8	0	0	-16.826	0
160	**Abrupt Pitch Termination**	0.64	0	0	-1.0	0	35.8	0	0	-16.826	0
161	**Abrupt Pitch Termination**	0.89	0	0	-1.0	0	35.8	0	0	-16.826	0
162	**Abrupt Pitch Termination**	0.15	0	0	-2.0	0	35.8	0	0	-16.826	0
163	**Abrupt Pitch Termination**	0.23	0	0	-2.0	0	35.8	0	0	-16.826	0
164	**Abrupt Pitch Termination**	0.38	0	0	-2.0	0	35.8	0	0	-16.826	0
165	**Abrupt Pitch Termination**	0.51	0	0	-2.0	0	35.8	0	0	-16.826	0
166	**Abrupt Pitch Termination**	0.60	0	0	-2.0	0	35.8	0	0	-16.826	0
167	**Abrupt Pitch Termination**	0.64	0	0	-2.0	0	35.8	0	0	-16.826	0
168	**Abrupt Pitch Termination**	0.89	0	0	-2.0	0	35.8	0	0	-16.826	0

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
113	Abrupt Pitch Termination	215	-13681	0	215	-13669	0	0	12	0	0.05%	0.09%	-1.86%
114	Abrupt Pitch Termination	218	-13592	0	218	-13594	0	0	-2	0	-0.03%	-0.01%	-1.95%
115	Abrupt Pitch Termination	222	-13402	0	222	-13402	0	0	0	0	0.01%	0.00%	-21.16%
116	Abrupt Pitch Termination	225	-13209	0	225	-13210	0	0	-1	0	0.03%	-0.01%	1.48%
117	Abrupt Pitch Termination	229	-13061	0	229	-13061	0	0	0	0	0.00%	0.00%	-0.23%
118	Abrupt Pitch Termination	232	-12981	0	231	-12987	0	0	-6	0	0.19%	-0.04%	1.67%
119	Abrupt Pitch Termination	276	-12219	0	275	-12232	0	-2	-13	0	0.58%	-0.11%	1.02%
120	Abrupt Pitch Termination	347	-6051	0	347	-6047	0	0	4	0	0.03%	0.07%	-3.73%
121	Abrupt Pitch Termination	348	-5947	0	348	-5950	0	0	-2	0	-0.05%	-0.04%	-5.26%
122	Abrupt Pitch Termination	347	-5708	0	347	-5709	0	0	-2	0	-0.03%	-0.03%	4.24%
123	Abrupt Pitch Termination	344	-5444	0	344	-5445	0	0	-2	0	-0.01%	-0.03%	1.01%
124	Abrupt Pitch Termination	340	-5229	0	340	-5228	0	0	0	0	0.00%	0.01%	-0.20%
125	Abrupt Pitch Termination	339	-5109	0	339	-5118	0	0	-9	0	0.01%	-0.17%	1.54%
126	Abrupt Pitch Termination	341	-3973	0	340	-3990	0	-1	-17	0	0.21%	-0.44%	1.09%
127	Abrupt Pitch Termination	48	28942	0	48	28914	0	0	-28	0	-0.29%	0.10%	0.79%
128	Abrupt Pitch Termination	43	28882	0	43	28884	0	0	1	0	-0.43%	-0.01%	0.59%
129	Abrupt Pitch Termination	29	28791	0	29	28788	0	0	-3	0	-1.00%	0.01%	0.67%
130	Abrupt Pitch Termination	11	28741	0	11	28743	0	0	2	0	-2.98%	-0.01%	0.45%
131	Abrupt Pitch Termination	-6	28725	0	-6	28726	0	0	1	0	0.19%	0.00%	-0.14%
132	Abrupt Pitch Termination	-17	28724	0	-15	28726	0	1	2	0	8.77%	-0.01%	1.27%
133	Abrupt Pitch Termination	-148	28712	0	-144	28724	0	4	12	0	2.56%	-0.04%	1.39%
134	Abrupt Pitch Termination	179	36573	0	180	36536	0	0	-37	0	-0.09%	0.10%	1.77%
135	Abrupt Pitch Termination	173	36527	0	173	36528	0	0	1	0	-0.18%	0.00%	1.21%
136	Abrupt Pitch Termination	154	36486	0	154	36481	0	0	-5	0	-0.26%	0.01%	1.02%
137	Abrupt Pitch Termination	129	36507	0	129	36508	0	0	1	0	-0.33%	0.00%	0.57%
138	Abrupt Pitch Termination	106	36557	0	106	36559	0	0	1	0	-0.01%	0.00%	-0.15%
139	Abrupt Pitch Termination	91	36596	0	93	36595	0	2	-1	0	-1.88%	0.00%	1.34%
140	Abrupt Pitch Termination	-84	36959	0	-79	36967	0	5	8	0	5.77%	-0.02%	1.28%
141	Abrupt Pitch Termination	-48	-28942	0	-48	-28913	0	0	29	0	-0.28%	0.10%	0.79%
142	Abrupt Pitch Termination	-43	-28882	0	-43	-28883	0	0	-1	0	-0.41%	0.00%	0.59%
143	Abrupt Pitch Termination	-29	-28791	0	-29	-28787	0	0	4	0	-0.87%	0.01%	0.67%
144	Abrupt Pitch Termination	-11	-28741	0	-11	-28740	0	0	1	0	-2.23%	0.00%	0.43%
145	Abrupt Pitch Termination	6	-28725	0	6	-28726	0	0	0	0	-0.05%	0.00%	-0.14%
146	Abrupt Pitch Termination	17	-28724	0	15	-28724	0	-1	0	0	8.24%	0.00%	1.25%
147	Abrupt Pitch Termination	148	-28712	0	145	-28717	0	-3	-5	0	2.30%	-0.02%	1.32%
148	Abrupt Pitch Termination	-179	-36573	0	-180	-36535	0	0	37	0	-0.08%	0.10%	1.77%
149	Abrupt Pitch Termination	-173	-36527	0	-173	-36527	0	0	0	0	-0.17%	0.00%	1.21%
150	Abrupt Pitch Termination	-154	-36486	0	-154	-36480	0	0	6	0	-0.24%	0.02%	1.01%
151	Abrupt Pitch Termination	-129	-36507	0	-129	-36505	0	0	1	0	-0.26%	0.00%	0.56%
152	Abrupt Pitch Termination	-106	-36557	0	-106	-36558	0	0	0	0	0.00%	0.00%	-0.16%
153	Abrupt Pitch Termination	-91	-36596	0	-93	-36593	0	-2	3	0	-1.81%	0.01%	1.34%
154	Abrupt Pitch Termination	84	-36959	0	80	-36960	0	-4	-1	0	5.28%	0.00%	1.24%
155	Abrupt Pitch Termination	-215	13681	0	-215	13669	0	0	-12	0	0.05%	0.09%	-1.86%
156	Abrupt Pitch Termination	-218	13592	0	-218	13595	0	0	2	0	-0.03%	-0.02%	-1.95%
157	Abrupt Pitch Termination	-222	13402	0	-222	13403	0	0	1	0	0.02%	-0.01%	-21.11%
158	Abrupt Pitch Termination	-225	13209	0	-225	13213	0	0	3	0	0.07%	-0.02%	1.45%
159	Abrupt Pitch Termination	-229	13061	0	-229	13062	0	0	1	0	0.00%	-0.01%	-0.23%
160	Abrupt Pitch Termination	-232	12981	0	-231	12989	0	1	8	0	0.23%	-0.06%	1.65%
161	Abrupt Pitch Termination	-276	12219	0	-275	12240	0	2	21	0	0.71%	-0.17%	0.94%
162	Abrupt Pitch Termination	-347	6051	0	-347	6047	0	0	-3	0	0.03%	0.06%	-3.73%
163	Abrupt Pitch Termination	-348	5947	0	-348	5950	0	0	3	0	-0.05%	-0.05%	-5.26%
164	Abrupt Pitch Termination	-347	5708	0	-347	5711	0	0	3	0	-0.02%	-0.05%	4.23%
165	Abrupt Pitch Termination	-344	5444	0	-344	5448	0	0	4	0	0.02%	-0.07%	1.00%
166	Abrupt Pitch Termination	-340	5229	0	-340	5229	0	0	0	0	0.00%	-0.01%	-0.20%
167	Abrupt Pitch Termination	-339	5109	0	-339	5120	0	0	11	0	0.03%	-0.21%	1.53%
168	Abrupt Pitch Termination	-341	3973	0	-340	3997	0	1	25	0	0.32%	-0.62%	1.05%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
57	Abrupt Roll Initiation	0.15	0	0	1.0	28.998	0	0	0	0	0
58	Abrupt Roll Initiation	0.225	0	0	1.0	28.998	0	0	0	0	0
59	Abrupt Roll Initiation	0.375	0	0	1.0	28.998	0	0	0	0	0
60	Abrupt Roll Initiation	0.51	0	0	1.0	28.998	0	0	0	0	0
61	Abrupt Roll Initiation	0.6	0	0	1.0	28.998	0	0	0	0	0
62	Abrupt Roll Initiation	0.64	0	0	1.0	28.998	0	0	0	0	0
63	Abrupt Roll Initiation	0.89	0	0	1.0	28.998	0	0	0	0	0
64	Abrupt Roll Initiation	0.15	0	0	2.0	28.998	0	0	0	0	0
65	Abrupt Roll Initiation	0.225	0	0	2.0	28.998	0	0	0	0	0
66	Abrupt Roll Initiation	0.375	0	0	2.0	28.998	0	0	0	0	0
67	Abrupt Roll Initiation	0.51	0	0	2.0	28.998	0	0	0	0	0
68	Abrupt Roll Initiation	0.6	0	0	2.0	28.998	0	0	0	0	0
69	Abrupt Roll Initiation	0.64	0	0	2.0	28.998	0	0	0	0	0
70	Abrupt Roll Initiation	0.89	0	0	2.0	28.998	0	0	0	0	0
71	Abrupt Roll Initiation	0.15	0	0	1.0	-28.998	0	0	0	0	0
72	Abrupt Roll Initiation	0.225	0	0	1.0	-28.998	0	0	0	0	0
73	Abrupt Roll Initiation	0.375	0	0	1.0	-28.998	0	0	0	0	0
74	Abrupt Roll Initiation	0.51	0	0	1.0	-28.998	0	0	0	0	0
75	Abrupt Roll Initiation	0.6	0	0	1.0	-28.998	0	0	0	0	0
76	Abrupt Roll Initiation	0.64	0	0	1.0	-28.998	0	0	0	0	0
77	Abrupt Roll Initiation	0.89	0	0	1.0	-28.998	0	0	0	0	0
78	Abrupt Roll Initiation	0.15	0	0	2.0	-28.998	0	0	0	0	0
79	Abrupt Roll Initiation	0.225	0	0	2.0	-28.998	0	0	0	0	0
80	Abrupt Roll Initiation	0.375	0	0	2.0	-28.998	0	0	0	0	0
81	Abrupt Roll Initiation	0.51	0	0	2.0	-28.998	0	0	0	0	0
82	Abrupt Roll Initiation	0.6	0	0	2.0	-28.998	0	0	0	0	0
83	Abrupt Roll Initiation	0.64	0	0	2.0	-28.998	0	0	0	0	0
84	Abrupt Roll Initiation	0.89	0	0	2.0	-28.998	0	0	0	0	0
85	Abrupt Roll Initiation	0.15	0	0	-1.0	28.998	0	0	0	0	0
86	Abrupt Roll Initiation	0.225	0	0	-1.0	28.998	0	0	0	0	0
87	Abrupt Roll Initiation	0.375	0	0	-1.0	28.998	0	0	0	0	0
88	Abrupt Roll Initiation	0.51	0	0	-1.0	28.998	0	0	0	0	0
89	Abrupt Roll Initiation	0.6	0	0	-1.0	28.998	0	0	0	0	0
90	Abrupt Roll Initiation	0.64	0	0	-1.0	28.998	0	0	0	0	0
91	Abrupt Roll Initiation	0.89	0	0	-1.0	28.998	0	0	0	0	0
92	Abrupt Roll Initiation	0.15	0	0	-2.0	28.998	0	0	0	0	0
93	Abrupt Roll Initiation	0.225	0	0	-2.0	28.998	0	0	0	0	0
94	Abrupt Roll Initiation	0.375	0	0	-2.0	28.998	0	0	0	0	0
95	Abrupt Roll Initiation	0.51	0	0	-2.0	28.998	0	0	0	0	0
96	Abrupt Roll Initiation	0.6	0	0	-2.0	28.998	0	0	0	0	0
97	Abrupt Roll Initiation	0.64	0	0	-2.0	28.998	0	0	0	0	0
98	Abrupt Roll Initiation	0.89	0	0	-2.0	28.998	0	0	0	0	0
99	Abrupt Roll Initiation	0.15	0	0	-1.0	-28.998	0	0	0	0	0
100	Abrupt Roll Initiation	0.225	0	0	-1.0	-28.998	0	0	0	0	0
101	Abrupt Roll Initiation	0.375	0	0	-1.0	-28.998	0	0	0	0	0
102	Abrupt Roll Initiation	0.51	0	0	-1.0	-28.998	0	0	0	0	0
103	Abrupt Roll Initiation	0.6	0	0	-1.0	-28.998	0	0	0	0	0
104	Abrupt Roll Initiation	0.64	0	0	-1.0	-28.998	0	0	0	0	0
105	Abrupt Roll Initiation	0.89	0	0	-1.0	-28.998	0	0	0	0	0
106	Abrupt Roll Initiation	0.15	0	0	-2.0	-28.998	0	0	0	0	0
107	Abrupt Roll Initiation	0.225	0	0	-2.0	-28.998	0	0	0	0	0
108	Abrupt Roll Initiation	0.375	0	0	-2.0	-28.998	0	0	0	0	0
109	Abrupt Roll Initiation	0.51	0	0	-2.0	-28.998	0	0	0	0	0
110	Abrupt Roll Initiation	0.6	0	0	-2.0	-28.998	0	0	0	0	0
111	Abrupt Roll Initiation	0.64	0	0	-2.0	-28.998	0	0	0	0	0
112	Abrupt Roll Initiation	0.89	0	0	-2.0	-28.998	0	0	0	0	0

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
57	Abrupt Roll Initiation	2057	7374	464	2054	7366	464	-2	-8	-1	0.12%	0.11%	0.13%
58	Abrupt Roll Initiation	2063	7389	466	2063	7389	466	0	-1	0	0.01%	0.01%	0.02%
59	Abrupt Roll Initiation	2085	7441	472	2084	7439	472	-1	-2	0	0.04%	0.03%	0.04%
60	Abrupt Roll Initiation	2117	7516	481	2117	7515	481	-1	-1	0	0.02%	0.01%	0.02%
61	Abrupt Roll Initiation	2149	7586	488	2149	7586	488	0	0	0	0.00%	0.00%	0.00%
62	Abrupt Roll Initiation	2168	7628	493	2166	7624	492	-2	-4	0	0.09%	0.05%	0.04%
63	Abrupt Roll Initiation	2380	8043	524	2377	8038	524	-3	-5	0	0.14%	0.06%	0.00%
64	Abrupt Roll Initiation	2188	15005	464	2186	14989	464	-2	-16	-1	0.11%	0.11%	0.13%
65	Abrupt Roll Initiation	2193	15034	466	2193	15033	466	0	-1	0	0.01%	0.01%	0.02%
66	Abrupt Roll Initiation	2210	15135	472	2209	15132	472	-1	-4	0	0.03%	0.02%	0.04%
67	Abrupt Roll Initiation	2235	15281	481	2235	15280	481	0	-2	0	0.02%	0.01%	0.02%
68	Abrupt Roll Initiation	2260	15418	488	2260	15418	488	0	0	0	0.00%	0.00%	0.00%
69	Abrupt Roll Initiation	2276	15500	493	2274	15493	492	-2	-7	0	0.07%	0.04%	0.04%
70	Abrupt Roll Initiation	2444	16290	524	2442	16280	524	-2	-9	0	0.10%	0.06%	0.00%
71	Abrupt Roll Initiation	-1794	7887	-464	-1791	7878	-464	2	-8	1	0.14%	0.11%	0.13%
72	Abrupt Roll Initiation	-1803	7900	-466	-1802	7900	-466	1	0	0	0.03%	0.01%	0.02%
73	Abrupt Roll Initiation	-1834	7948	-472	-1833	7946	-472	1	-2	0	0.06%	0.02%	0.04%
74	Abrupt Roll Initiation	-1881	8016	-481	-1880	8015	-481	1	-1	0	0.04%	0.01%	0.02%
75	Abrupt Roll Initiation	-1925	8079	-488	-1925	8079	-488	0	0	0	0.00%	0.00%	0.00%
76	Abrupt Roll Initiation	-1953	8115	-493	-1950	8113	-492	3	-3	0	0.15%	0.03%	0.04%
77	Abrupt Roll Initiation	-2252	8450	-524	-2247	8447	-524	5	-3	0	0.23%	0.04%	0.00%
78	Abrupt Roll Initiation	-1662	15517	-464	-1660	15500	-464	3	-17	1	0.15%	0.11%	0.13%
79	Abrupt Roll Initiation	-1673	15545	-466	-1672	15544	-466	1	-1	0	0.04%	0.01%	0.02%
80	Abrupt Roll Initiation	-1709	15643	-472	-1708	15639	-472	1	-4	0	0.07%	0.02%	0.04%
81	Abrupt Roll Initiation	-1763	15782	-481	-1762	15780	-481	1	-2	0	0.05%	0.01%	0.02%
82	Abrupt Roll Initiation	-1814	15911	-488	-1814	15911	-488	0	0	0	0.00%	0.00%	0.00%
83	Abrupt Roll Initiation	-1846	15987	-493	-1843	15981	-492	3	-6	0	0.18%	0.04%	0.04%
84	Abrupt Roll Initiation	-2188	16696	-524	-2182	16689	-524	6	-7	0	0.27%	0.04%	0.00%
85	Abrupt Roll Initiation	1794	-7887	464	1791	-7878	464	-2	8	-1	0.14%	0.11%	0.13%
86	Abrupt Roll Initiation	1803	-7900	466	1802	-7900	466	-1	0	0	0.03%	0.01%	0.02%
87	Abrupt Roll Initiation	1834	-7948	472	1833	-7946	472	-1	2	0	0.05%	0.02%	0.04%
88	Abrupt Roll Initiation	1881	-8016	481	1880	-8015	481	-1	1	0	0.04%	0.01%	0.02%
89	Abrupt Roll Initiation	1925	-8079	488	1926	-8079	488	0	0	0	0.00%	0.00%	0.00%
90	Abrupt Roll Initiation	1953	-8115	493	1951	-8113	492	-3	3	0	0.14%	0.03%	0.04%
91	Abrupt Roll Initiation	2252	-8450	524	2247	-8447	524	-5	3	0	0.23%	0.04%	0.00%
92	Abrupt Roll Initiation	1662	-15517	464	1660	-15500	464	-2	17	-1	0.15%	0.11%	0.13%
93	Abrupt Roll Initiation	1673	-15545	466	1672	-15544	466	-1	1	0	0.04%	0.01%	0.02%
94	Abrupt Roll Initiation	1709	-15643	472	1708	-15639	472	-1	4	0	0.07%	0.02%	0.04%
95	Abrupt Roll Initiation	1763	-15782	481	1762	-15780	481	-1	2	0	0.04%	0.01%	0.02%
96	Abrupt Roll Initiation	1814	-15911	488	1814	-15911	488	0	0	0	0.00%	0.00%	0.00%
97	Abrupt Roll Initiation	1846	-15987	493	1843	-15981	492	-3	6	0	0.18%	0.04%	0.04%
98	Abrupt Roll Initiation	2188	-16696	524	2182	-16689	524	-6	7	0	0.27%	0.04%	0.00%
99	Abrupt Roll Initiation	-2057	-7374	-464	-2054	-7366	-464	2	8	1	0.12%	0.11%	0.13%
100	Abrupt Roll Initiation	-2063	-7389	-466	-2063	-7389	-466	0	1	0	0.02%	0.01%	0.02%
101	Abrupt Roll Initiation	-2085	-7441	-472	-2084	-7439	-472	1	2	0	0.04%	0.03%	0.04%
102	Abrupt Roll Initiation	-2117	-7516	-481	-2117	-7515	-481	1	1	0	0.03%	0.01%	0.02%
103	Abrupt Roll Initiation	-2149	-7586	-488	-2149	-7586	-488	0	0	0	0.00%	0.00%	0.00%
104	Abrupt Roll Initiation	-2168	-7628	-493	-2166	-7624	-492	2	4	0	0.09%	0.05%	0.04%
105	Abrupt Roll Initiation	-2380	-8043	-524	-2377	-8038	-524	3	5	0	0.14%	0.06%	0.00%
106	Abrupt Roll Initiation	-2188	-15005	-464	-2186	-14989	-464	2	16	1	0.11%	0.11%	0.13%
107	Abrupt Roll Initiation	-2193	-15034	-466	-2193	-15033	-466	0	1	0	0.01%	0.01%	0.02%
108	Abrupt Roll Initiation	-2210	-15135	-472	-2209	-15132	-472	1	4	0	0.03%	0.02%	0.04%
109	Abrupt Roll Initiation	-2235	-15281	-481	-2235	-15280	-481	0	2	0	0.02%	0.01%	0.02%
110	Abrupt Roll Initiation	-2260	-15418	-488	-2260	-15418	-488	0	0	0	0.00%	0.00%	0.00%
111	Abrupt Roll Initiation	-2276	-15500	-493	-2274	-15493	-492	2	7	0	0.07%	0.04%	0.04%
112	Abrupt Roll Initiation	-2444	-16290	-524	-2442	-16280	-524	2	9	0	0.10%	0.06%	0.00%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
169	Abrupt Roll Termination	0.15	0	0	1.0	-28.998	0	0	0	22.6184	0
170	Abrupt Roll Termination	0.225	0	0	1.0	-28.998	0	0	0	22.6184	0
171	Abrupt Roll Termination	0.375	0	0	1.0	-28.998	0	0	0	22.6184	0
172	Abrupt Roll Termination	0.51	0	0	1.0	-28.998	0	0	0	22.6184	0
173	Abrupt Roll Termination	0.6	0	0	1.0	-28.998	0	0	0	22.6184	0
174	Abrupt Roll Termination	0.64	0	0	1.0	-28.998	0	0	0	22.6184	0
175	Abrupt Roll Termination	0.89	0	0	1.0	-28.998	0	0	0	22.6184	0
176	Abrupt Roll Termination	0.15	0	0	2.0	-28.998	0	0	0	22.6184	0
177	Abrupt Roll Termination	0.225	0	0	2.0	-28.998	0	0	0	22.6184	0
178	Abrupt Roll Termination	0.375	0	0	2.0	-28.998	0	0	0	22.6184	0
179	Abrupt Roll Termination	0.51	0	0	2.0	-28.998	0	0	0	22.6184	0
180	Abrupt Roll Termination	0.6	0	0	2.0	-28.998	0	0	0	22.6184	0
181	Abrupt Roll Termination	0.64	0	0	2.0	-28.998	0	0	0	22.6184	0
182	Abrupt Roll Termination	0.89	0	0	2.0	-28.998	0	0	0	22.6184	0
183	Abrupt Roll Termination	0.15	0	0	1.0	28.998	0	0	0	-22.6184	0
184	Abrupt Roll Termination	0.225	0	0	1.0	28.998	0	0	0	-22.6184	0
185	Abrupt Roll Termination	0.375	0	0	1.0	28.998	0	0	0	-22.6184	0
186	Abrupt Roll Termination	0.51	0	0	1.0	28.998	0	0	0	-22.6184	0
187	Abrupt Roll Termination	0.6	0	0	1.0	28.998	0	0	0	-22.6184	0
188	Abrupt Roll Termination	0.64	0	0	1.0	28.998	0	0	0	-22.6184	0
189	Abrupt Roll Termination	0.89	0	0	1.0	28.998	0	0	0	-22.6184	0
190	Abrupt Roll Termination	0.15	0	0	2.0	28.998	0	0	0	-22.6184	0
191	Abrupt Roll Termination	0.225	0	0	2.0	28.998	0	0	0	-22.6184	0
192	Abrupt Roll Termination	0.375	0	0	2.0	28.998	0	0	0	-22.6184	0
193	Abrupt Roll Termination	0.51	0	0	2.0	28.998	0	0	0	-22.6184	0
194	Abrupt Roll Termination	0.6	0	0	2.0	28.998	0	0	0	-22.6184	0
195	Abrupt Roll Termination	0.64	0	0	2.0	28.998	0	0	0	-22.6184	0
196	Abrupt Roll Termination	0.89	0	0	2.0	28.998	0	0	0	-22.6184	0
197	Abrupt Roll Termination	0.15	0	0	-1.0	-28.998	0	0	0	22.6184	0
198	Abrupt Roll Termination	0.225	0	0	-1.0	-28.998	0	0	0	22.6184	0
199	Abrupt Roll Termination	0.375	0	0	-1.0	-28.998	0	0	0	22.6184	0
200	Abrupt Roll Termination	0.51	0	0	-1.0	-28.998	0	0	0	22.6184	0
201	Abrupt Roll Termination	0.6	0	0	-1.0	-28.998	0	0	0	22.6184	0
202	Abrupt Roll Termination	0.64	0	0	-1.0	-28.998	0	0	0	22.6184	0
203	Abrupt Roll Termination	0.89	0	0	-1.0	-28.998	0	0	0	22.6184	0
204	Abrupt Roll Termination	0.15	0	0	-2.0	-28.998	0	0	0	22.6184	0
205	Abrupt Roll Termination	0.225	0	0	-2.0	-28.998	0	0	0	22.6184	0
206	Abrupt Roll Termination	0.375	0	0	-2.0	-28.998	0	0	0	22.6184	0
207	Abrupt Roll Termination	0.51	0	0	-2.0	-28.998	0	0	0	22.6184	0
208	Abrupt Roll Termination	0.6	0	0	-2.0	-28.998	0	0	0	22.6184	0
209	Abrupt Roll Termination	0.64	0	0	-2.0	-28.998	0	0	0	22.6184	0
210	Abrupt Roll Termination	0.89	0	0	-2.0	-28.998	0	0	0	22.6184	0
211	Abrupt Roll Termination	0.15	0	0	-1.0	28.998	0	0	0	-22.6184	0
212	Abrupt Roll Termination	0.225	0	0	-1.0	28.998	0	0	0	-22.6184	0
213	Abrupt Roll Termination	0.375	0	0	-1.0	28.998	0	0	0	-22.6184	0
214	Abrupt Roll Termination	0.51	0	0	-1.0	28.998	0	0	0	-22.6184	0
215	Abrupt Roll Termination	0.6	0	0	-1.0	28.998	0	0	0	-22.6184	0
216	Abrupt Roll Termination	0.64	0	0	-1.0	28.998	0	0	0	-22.6184	0
217	Abrupt Roll Termination	0.89	0	0	-1.0	28.998	0	0	0	-22.6184	0
218	Abrupt Roll Termination	0.15	0	0	-2.0	28.998	0	0	0	-22.6184	0
219	Abrupt Roll Termination	0.225	0	0	-2.0	28.998	0	0	0	-22.6184	0
220	Abrupt Roll Termination	0.375	0	0	-2.0	28.998	0	0	0	-22.6184	0
221	Abrupt Roll Termination	0.51	0	0	-2.0	28.998	0	0	0	-22.6184	0
222	Abrupt Roll Termination	0.6	0	0	-2.0	28.998	0	0	0	-22.6184	0
223	Abrupt Roll Termination	0.64	0	0	-2.0	28.998	0	0	0	-22.6184	0
224	Abrupt Roll Termination	0.89	0	0	-2.0	28.998	0	0	0	-22.6184	0

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
169	Abrupt Roll Termination	-1784	8114	-464	-1783	8112	-464	1	-2	0	0.05%	0.02%	0.04%
170	Abrupt Roll Termination	-1788	8244	-466	-1788	8242	-466	1	-2	0	0.04%	0.02%	0.03%
171	Abrupt Roll Termination	-1809	8536	-472	-1808	8534	-472	1	-2	0	0.04%	0.03%	0.03%
172	Abrupt Roll Termination	-1843	8847	-481	-1843	8845	-481	1	-2	0	0.03%	0.02%	0.02%
173	Abrupt Roll Termination	-1878	9092	-488	-1878	9092	-488	0	0	0	0.00%	0.00%	0.00%
174	Abrupt Roll Termination	-1901	9224	-493	-1898	9215	-492	3	-8	0	0.14%	0.09%	0.05%
175	Abrupt Roll Termination	-2147	10402	-524	-2143	10388	-524	4	-14	0	0.21%	0.14%	0.00%
176	Abrupt Roll Termination	-1653	15744	-464	-1652	15741	-464	1	-3	0	0.06%	0.02%	0.04%
177	Abrupt Roll Termination	-1658	15889	-466	-1657	15886	-466	1	-3	0	0.05%	0.02%	0.03%
178	Abrupt Roll Termination	-1684	16230	-472	-1683	16227	-472	1	-4	0	0.05%	0.02%	0.03%
179	Abrupt Roll Termination	-1725	16613	-481	-1725	16610	-481	1	-3	0	0.04%	0.02%	0.02%
180	Abrupt Roll Termination	-1767	16924	-488	-1767	16924	-488	0	0	0	0.00%	0.00%	0.00%
181	Abrupt Roll Termination	-1793	17095	-493	-1790	17084	-492	3	-12	0	0.17%	0.07%	0.05%
182	Abrupt Roll Termination	-2083	18649	-524	-2078	18630	-524	5	-18	0	0.26%	0.10%	0.00%
183	Abrupt Roll Termination	2047	7147	464	2047	7146	464	-1	-1	0	0.03%	0.01%	0.04%
184	Abrupt Roll Termination	2049	7046	466	2048	7046	466	0	0	0	0.02%	0.01%	0.03%
185	Abrupt Roll Termination	2059	6853	472	2059	6853	472	0	0	0	0.02%	0.00%	0.03%
186	Abrupt Roll Termination	2080	6685	481	2079	6686	481	0	2	0	0.02%	-0.02%	0.02%
187	Abrupt Roll Termination	2101	6573	488	2101	6574	488	0	2	0	0.00%	-0.03%	0.00%
188	Abrupt Roll Termination	2116	6520	493	2114	6521	492	-2	2	0	0.08%	-0.03%	0.05%
189	Abrupt Roll Termination	2276	6091	524	2273	6102	524	-2	11	0	0.10%	-0.19%	0.00%
190	Abrupt Roll Termination	2179	14778	464	2178	14775	464	0	-2	0	0.02%	0.02%	0.04%
191	Abrupt Roll Termination	2179	14691	466	2178	14689	466	0	-2	0	0.01%	0.01%	0.03%
192	Abrupt Roll Termination	2184	14547	472	2184	14546	472	0	-1	0	0.02%	0.01%	0.03%
193	Abrupt Roll Termination	2198	14450	481	2198	14451	481	0	1	0	0.01%	-0.01%	0.02%
194	Abrupt Roll Termination	2213	14405	488	2213	14407	488	0	2	0	0.00%	-0.01%	0.00%
195	Abrupt Roll Termination	2223	14392	493	2222	14390	492	-1	-2	0	0.06%	0.01%	0.05%
196	Abrupt Roll Termination	2340	14337	524	2338	14344	524	-1	7	0	0.06%	-0.05%	0.00%
197	Abrupt Roll Termination	-2047	-7147	-464	-2047	-7146	-464	1	1	0	0.03%	0.01%	0.04%
198	Abrupt Roll Termination	-2049	-7046	-466	-2048	-7045	-466	0	1	0	0.02%	0.01%	0.03%
199	Abrupt Roll Termination	-2059	-6853	-472	-2059	-6853	-472	1	0	0	0.03%	0.00%	0.03%
200	Abrupt Roll Termination	-2080	-6685	-481	-2079	-6685	-481	0	0	0	0.02%	0.00%	0.02%
201	Abrupt Roll Termination	-2101	-6573	-488	-2101	-6572	-488	0	0	0	0.00%	0.01%	0.00%
202	Abrupt Roll Termination	-2116	-6520	-493	-2114	-6521	-492	2	-1	0	0.08%	-0.01%	0.05%
203	Abrupt Roll Termination	-2276	-6091	-524	-2273	-6096	-524	3	-5	0	0.12%	-0.09%	0.00%
204	Abrupt Roll Termination	-2179	-14778	-464	-2178	-14775	-464	0	2	0	0.02%	0.02%	0.04%
205	Abrupt Roll Termination	-2179	-14691	-466	-2178	-14689	-466	0	2	0	0.02%	0.01%	0.03%
206	Abrupt Roll Termination	-2184	-14547	-472	-2184	-14546	-472	0	2	0	0.02%	0.01%	0.03%
207	Abrupt Roll Termination	-2198	-14450	-481	-2197	-14450	-481	0	1	0	0.02%	0.00%	0.02%
208	Abrupt Roll Termination	-2213	-14405	-488	-2213	-14405	-488	0	0	0	0.00%	0.00%	0.00%
209	Abrupt Roll Termination	-2223	-14392	-493	-2222	-14389	-492	1	3	0	0.06%	0.02%	0.05%
210	Abrupt Roll Termination	-2340	-14337	-524	-2338	-14339	-524	2	-1	0	0.08%	-0.01%	0.00%
211	Abrupt Roll Termination	1784	-8114	464	1783	-8112	464	-1	2	0	0.05%	0.02%	0.04%
212	Abrupt Roll Termination	1788	-8244	466	1788	-8242	466	-1	2	0	0.04%	0.03%	0.03%
213	Abrupt Roll Termination	1809	-8536	472	1808	-8533	472	-1	3	0	0.04%	0.03%	0.03%
214	Abrupt Roll Termination	1843	-8847	481	1843	-8844	481	-1	3	0	0.03%	0.04%	0.02%
215	Abrupt Roll Termination	1878	-9092	488	1878	-9090	488	0	2	0	0.00%	0.02%	0.00%
216	Abrupt Roll Termination	1901	-9224	493	1898	-9215	492	-3	9	0	0.13%	0.10%	0.05%
217	Abrupt Roll Termination	2147	-10402	524	2143	-10382	524	-4	20	0	0.19%	0.19%	0.00%
218	Abrupt Roll Termination	1653	-15744	464	1652	-15741	464	-1	3	0	0.06%	0.02%	0.04%
219	Abrupt Roll Termination	1658	-15889	466	1657	-15886	466	-1	3	0	0.05%	0.02%	0.03%
220	Abrupt Roll Termination	1684	-16230	472	1683	-16226	472	-1	4	0	0.05%	0.03%	0.03%
221	Abrupt Roll Termination	1725	-16613	481	1725	-16609	481	-1	4	0	0.04%	0.02%	0.02%
222	Abrupt Roll Termination	1767	-16924	488	1767	-16922	488	0	2	0	0.00%	0.01%	0.00%
223	Abrupt Roll Termination	1793	-17095	493	1790	-17083	492	-3	12	0	0.16%	0.07%	0.05%
224	Abrupt Roll Termination	2083	-18649	524	2078	-18624	524	-5	24	0	0.24%	0.13%	0.00%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
225	**Abrupt Rudder Kick**	0.15	0	0	1.0	0	0	7.2	0	0	0
226	**Abrupt Rudder Kick**	0.225	0	0	1.0	0	0	7.2	0	0	0
227	**Abrupt Rudder Kick**	0.375	0	0	1.0	0	0	7.2	0	0	0
228	**Abrupt Rudder Kick**	0.51	0	0	1.0	0	0	7.2	0	0	0
229	**Abrupt Rudder Kick**	0.6	0	0	1.0	0	0	7.2	0	0	0
230	**Abrupt Rudder Kick**	0.64	0	0	1.0	0	0	7.2	0	0	0
231	**Abrupt Rudder Kick**	0.89	0	0	1.0	0	0	7.2	0	0	0
232	**Abrupt Rudder Kick**	0.15	0	1	1.0	0	0	7.2	0	0	0
233	**Abrupt Rudder Kick**	0.225	0	1	1.0	0	0	7.2	0	0	0
234	**Abrupt Rudder Kick**	0.375	0	1	1.0	0	0	7.2	0	0	0
235	**Abrupt Rudder Kick**	0.51	0	1	1.0	0	0	7.2	0	0	0
236	**Abrupt Rudder Kick**	0.6	0	1	1.0	0	0	7.2	0	0	0
237	**Abrupt Rudder Kick**	0.64	0	1	1.0	0	0	7.2	0	0	0
238	**Abrupt Rudder Kick**	0.89	0	1	1.0	0	0	7.2	0	0	0

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
225	Abrupt Rudder Kick	-387	9371	-5692	-386	9370	-5690	0	-1	2	0.12%	0.01%	0.04%
226	Abrupt Rudder Kick	-392	9380	-5715	-391	9379	-5713	0	-1	2	0.10%	0.01%	0.03%
227	Abrupt Rudder Kick	-409	9409	-5792	-409	9408	-5790	0	-1	2	0.10%	0.01%	0.04%
228	Abrupt Rudder Kick	-433	9450	-5904	-433	9450	-5902	0	0	1	0.06%	0.00%	0.03%
229	Abrupt Rudder Kick	-455	9487	-6007	-455	9487	-6007	0	0	0	0.00%	0.00%	0.00%
230	Abrupt Rudder Kick	-467	9508	-6070	-466	9507	-6064	1	-2	6	0.23%	0.02%	0.10%
231	Abrupt Rudder Kick	-591	9693	-6697	-589	9691	-6687	2	-2	9	0.34%	0.02%	0.14%
232	Abrupt Rudder Kick	1305	6595	-1518	1305	6594	-1518	0	-1	0	0.02%	0.02%	0.00%
233	Abrupt Rudder Kick	1308	6608	-1518	1308	6607	-1518	0	-1	0	0.02%	0.02%	0.00%
234	Abrupt Rudder Kick	1320	6654	-1518	1320	6653	-1518	0	-1	0	0.03%	0.02%	0.00%
235	Abrupt Rudder Kick	1339	6722	-1520	1339	6721	-1520	0	-1	0	0.03%	0.01%	0.01%
236	Abrupt Rudder Kick	1359	6786	-1523	1359	6786	-1523	0	0	0	0.00%	0.00%	0.00%
237	Abrupt Rudder Kick	1372	6824	-1526	1370	6821	-1525	-2	-4	1	0.12%	0.06%	0.04%
238	Abrupt Rudder Kick	1529	7210	-1579	1526	7205	-1578	-3	-5	2	0.20%	0.07%	0.11%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
239	**Abrupt Rudder Return**	0.15	0	0	1	0	0	-7.2	0	0	3.024
240	**Abrupt Rudder Return**	0.225	0	0	1	0	0	-7.2	0	0	3.024
241	**Abrupt Rudder Return**	0.375	0	0	1	0	0	-7.2	0	0	3.024
242	**Abrupt Rudder Return**	0.51	0	0	1	0	0	-7.2	0	0	3.024
243	**Abrupt Rudder Return**	0.6	0	0	1	0	0	-7.2	0	0	3.024
244	**Abrupt Rudder Return**	0.64	0	0	1	0	0	-7.2	0	0	3.024
245	**Abrupt Rudder Return**	0.89	0	0	1	0	0	-7.2	0	0	3.024
246	**Abrupt Rudder Return**	0.15	0	1	1	0	0	-7.2	0	0	3.024
247	**Abrupt Rudder Return**	0.225	0	1	1	0	0	-7.2	0	0	3.024
248	**Abrupt Rudder Return**	0.375	0	1	1	0	0	-7.2	0	0	3.024
249	**Abrupt Rudder Return**	0.51	0	1	1	0	0	-7.2	0	0	3.024
250	**Abrupt Rudder Return**	0.6	0	1	1	0	0	-7.2	0	0	3.024
251	**Abrupt Rudder Return**	0.64	0	1	1	0	0	-7.2	0	0	3.024
252	**Abrupt Rudder Return**	0.89	0	1	1	0	0	-7.2	0	0	3.024

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
239	Abrupt Rudder Return	651	5887	5686	650	5885	5684	0	-2	-2	0.03%	0.03%	0.04%
240	Abrupt Rudder Return	653	5905	5706	653	5904	5704	0	-2	-2	0.03%	0.03%	0.03%
241	Abrupt Rudder Return	662	5971	5777	662	5970	5775	0	-2	-2	0.02%	0.03%	0.04%
242	Abrupt Rudder Return	673	6070	5882	673	6068	5881	0	-1	-2	0.02%	0.02%	0.03%
243	Abrupt Rudder Return	682	6163	5982	682	6163	5982	0	0	0	0.00%	0.00%	0.00%
244	Abrupt Rudder Return	687	6219	6042	687	6214	6036	0	-5	-6	0.04%	0.09%	0.10%
245	Abrupt Rudder Return	729	6775	6651	729	6768	6641	0	-7	-9	0.04%	0.10%	0.14%
246	Abrupt Rudder Return	2342	3111	9861	2341	3109	9857	-1	-2	-4	0.04%	0.08%	0.04%
247	Abrupt Rudder Return	2353	3134	9903	2352	3132	9899	-1	-2	-4	0.04%	0.06%	0.04%
248	Abrupt Rudder Return	2391	3217	10050	2390	3214	10046	-1	-2	-4	0.04%	0.07%	0.04%
249	Abrupt Rudder Return	2445	3341	10266	2444	3340	10263	-1	-2	-3	0.03%	0.05%	0.03%
250	Abrupt Rudder Return	2495	3461	10466	2495	3461	10466	0	0	0	0.00%	0.00%	0.00%
251	Abrupt Rudder Return	2526	3535	10586	2523	3528	10575	-3	-8	-11	0.12%	0.22%	0.11%
252	Abrupt Rudder Return	2849	4292	11768	2844	4282	11751	-5	-10	-17	0.19%	0.23%	0.15%

Condition ID	Description	Mach	Altitude	Ny	Nz	Pdot [deg/s]	Qdot [deg/s]	Rdot [deg/s]	P [deg/s]	Q [deg/s]	R [deg/s]
253	Multi DOF Problem 1	0.55	0	-0.21	1.65	14.1	7.89	3.64	7.8	6.5	1.4
254	Multi DOF Problem 2	0.79	0	-0.21	1.65	14.1	7.89	3.64	-7.8	-6.5	-1.4
255	Multi DOF Problem 3	0.18	0	-0.21	-1.65	-14.1	-7.89	3.64	7.8	-6.5	-1.4

Condition ID	Description	Estimated Aileron HM [ft-lb]	Estimated Elevator HM [ft-lb]	Estimated Rudder HM [ft lb]	Actual Aileron HM [ft-lb]	Actual Elevator HM [ft-lb]	Actual Rudder HM [ft lb]	delta Aileron HM [ft-lb]	delta Elevator HM [ft-lb]	delta Rudder HM [ft-lb]	% ERROR Aileron HM	% ERROR Elevator HM	% ERROR RUDDER HM
253	Multi DOF Problem 1	643	19140	-3724	642	19135	-3721	0	-6	3	0.06%	0.03%	0.08%
254	Multi DOF Problem 2	202	18936	-3941	204	18935	-3939	2	0	2	-0.89%	0.00%	0.06%
255	Multi DOF Problem 3	-1720	-15836	-3985	-1720	-15833	-3984	0	2	1	0.01%	0.02%	0.03%

Appendix G
Baseline Matrix Generation

```matlab
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY

clear
clc

%                    NY          NZ          PDOT        QDOT        RDOT        P          Q          R
DOFMODES      = {  'URDD2'   ,  'URDD3'   ,  'URDD4'   ,  'URDD5'   ,  'URDD6'   ,  'ROLL'  ,  'PITCH'  ,  'YAW'   };
DOFVALS       = [     0      ,     1      ,     0      ,     0      ,     0      ,    0     ,    0      ,    0     ];

% Initialize the CASEID to be 1 and increment for each run.
CASEID = 1e5;
% Initialize the condition list
CONDITIONS = [];
% Set the MACH Array to Evaluate
MACH = 0.1:.1:.9;
% Get rho0
rho0 = stdatmo(0,0,'US');
%
TrimInput = struct('ID',   [],'DESCRIPTION',[],'MACH', [],'ALTITUDE',[],'QBAR', [],'KEAS',[],'KTAS', [],...
                    'URDD2',[],'URDD3',      [],'URDD4',[],'URDD5',   [],'URDD6',[],'ROLL',[],'PITCH',[],'YAW',[],'VARS',[]);
% Loop over each variable to create the trim input
cnt = 1;
for j=1:length(MACH)
    TrimInput(cnt).ID             = CASEID;
    TrimInput(cnt).MACH           = MACH(j);
    TrimInput(cnt).ALTITUDE       = 0;
    TrimInput(cnt).QBAR           = Getqbar(MACH(j),[],0);
    TrimInput(cnt).KEAS           = sqrt((2*TrimInput(cnt).QBAR)/rho0)*0.592484;
    TrimInput(cnt).KTAS           = GetTAS([],TrimInput(cnt).QBAR,0)*0.592484;
    for k=1:length(DOFMODES)
        TrimInput(cnt).VARS(k).DOF = DOFMODES{k};
        TrimInput(cnt).VARS(k).VAL = DOFVALS(k);
        TrimInput(cnt).(DOFMODES{k})    = DOFVALS(k);
    end
    TrimInput(cnt).DESCRIPTION = ['BASELINE 1g TRIM AT MACH = ' num2str(MACH(j)) ' CASE ID ' num2str(cnt)];
    CASEID = CASEID + 1;
    cnt = cnt + 1;
end


WriteTRIMInclude(TrimInput,'TrimInclude_BaseLine.bdf')

WriteFullModelBDF_Baseline('BaselineRun.bdf',TrimInput)
```

Appendix H
Expanded Matrix Generation

```
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY
clear
clc
%% This section defines the variable degrees of freedom and additional trim variables to include in the database buildup
% The min, incremental and max values are defined here. The code will loop
% over each of the DOFMODES, or the variables, and build a matrix array
% input for each.

%                    NY            NZ         PDOT [deg/s^2]   QDOT [deg/s^2]  RDOT [deg/s^2]    P [deg/s]      Q [deg/s]       R [deg/s]
DOFMODES       = {   'URDD2'    ,   'URDD3'   ,   'URDD4'     ,   'URDD5'     ,  'URDD6'     ,   'ROLL'     ,   'PITCH'     ,   'YAW'      };
MINVALARRAY    = [     -2       ,     -4      ,     -100      ,     -100      ,    -50       ,    -100      ,    -100       ,    -50       ];
INCREMENTARRAY = [      1       ,      1      ,       50      ,       25      ,     25       ,      50      ,      50       ,     25       ];
MAXVALARRAY    = [      2       ,      9      ,      100      ,      100      ,     50       ,     100      ,     100       ,     50       ];
DESCRIPTION    = {  'VAR NY'    ,  'VAR NZ'   ,  'VAR PDOT'   ,  'VAR QDOT'   , 'VAR RDOT'   ,  'VAR P'     ,  'VAR Q'      ,  'VAR R'     };

% Initialize the CASEID to be 1 and increment for each run.
CASEID = 1;
% Initialize the condition list
CONDITIONS = [];
% Set the MACH Array to Evaluate
MACH = 0.1:.1:.9;
% Get rho0
rho0 = stdatmo(0,0,'US');

TrimInput = struct('ID',[],'DESCRIPTION',[],'MACH',[],'ALTITUDE',[],'QBAR',[],'KEAS',[],'KTAS',[],'URDD2',[],'URDD3',[],'URDD4',[],'URDD5',[],'URDD6',[],'ROLL',[],'PITCH',[],'YAW',[],'VARS',[]);

% Loop over each variable to create the trim input
for DOF = 1:length(DOFMODES)
    % Get max/min/inc values for the DOF variable
    EVALARRAY =  MINVALARRAY(DOF) : INCREMENTARRAY(DOF) : MAXVALARRAY(DOF) ;
    VARMODE = DOFMODES{DOF};
    STDMODE = DOFMODES(~ismember(DOFMODES,{VARMODE,'URDD3'}));
    if ~strcmp(VARMODE,'URDD3')
        STDMODE = [STDMODE,'URDD3'];
        PHI = [0 0 0 0 0 0 0];
    else
        PHI = zeros(1,length(STDMODE));
    end
    fprintf('EVALUATING MODE: %-7s over range: [ %5.2f : %5.2f : %5.2f]\n', VARMODE , MINVALARRAY(DOF) , INCREMENTARRAY(DOF) , MAXVALARRAY(DOF) );

                for i=1:length(EVALARRAY)
                    for j=1:length(MACH)
                        TrimInput(CASEID).ID            = CASEID;
                        TrimInput(CASEID).MACH          = MACH(j);
                        TrimInput(CASEID).ALTITUDE      = 0;
                        TrimInput(CASEID).QBAR          = Getqbar(MACH(j),[],0);
                        TrimInput(CASEID).KEAS          = sqrt((2*TrimInput(CASEID).QBAR)/rho0)*0.592484;
                        TrimInput(CASEID).KTAS          = GetTAS([],TrimInput(CASEID).QBAR,0)*0.592484;

                        % Set the variable data
                        % Non-Dimensionalize the data as needed:
                        tmp = EVALARRAY(i);
                        switch VARMODE
                            case 'URDD4'
                                ROTVAL = NonDim('Pdot',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            case 'URDD5'
                                ROTVAL = NonDim('Qdot',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            case 'URDD6'
                                ROTVAL = NonDim('Rdot',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            case 'ROLL'
                                ROTVAL = NonDim('Roll',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            case 'PITCH'
                                ROTVAL = NonDim('Pitch',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            case 'YAW'
                                ROTVAL = NonDim('Yaw',EVALARRAY(i),TrimInput(CASEID).KTAS/0.592484);
                            otherwise
                                ROTVAL = EVALARRAY(i);
                        end

                        TrimInput(CASEID).VARS(ismember(DOFMODES,VARMODE)).DOF   = VARMODE;
                        TrimInput(CASEID).VARS(ismember(DOFMODES,VARMODE)).VAL   = ROTVAL;
                        TrimInput(CASEID).(DOFMODES{ismember(DOFMODES,VARMODE)})   = tmp;

                        for k=1:length(STDMODE)
                            TrimInput(CASEID).VARS(ismember(DOFMODES,STDMODE{k})).DOF = STDMODE{k};
                            TrimInput(CASEID).VARS(ismember(DOFMODES,STDMODE{k})).VAL = PHI(k);
                            TrimInput(CASEID).(DOFMODES{ismember(DOFMODES,STDMODE{k})}) = PHI(k);
                        end

                        TrimInput(CASEID).DESCRIPTION   = [DESCRIPTION{DOF} ' = ' num2str(tmp) '  MACH = ' num2str(MACH(j)) ' CASE ID ' num2str(CASEID)];


                        CASEID = CASEID+1;
                    end
                end

end

WriteTRIMInclude(TrimInput,'TrimInclude_FullMatrix.bdf')

WriteFullModelBDF_FullMatrix('FullMatrixRun.bdf',TrimInput)
```

Appendix I
Trim Estimator

```
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY
function TrimOut = TrimEstimator(DOF,PQRArray,Altitude,Mach)
clc
Ny = DOF(1);
Nz = DOF(2);
P  = deg2rad(PQRArray(1));
Q  = deg2rad(PQRArray(2));
R  = deg2rad(PQRArray(3));
Pdot = deg2rad(DOF(3));
Qdot = deg2rad(DOF(4));
Rdot = deg2rad(DOF(5));

qbar = Getqbar(Mach,[],Altitude);

TAS = GetTAS([],qbar,Altitude);


RateDesc = {'Roll','Pitch','Yaw'};
for i=1:3
    RateStates(i) = NonDim(RateDesc{i},rad2deg(PQRArray(i)),TAS);
end

S_REF = 400;
b_REF = 40;
c_REF = 10;

%% Inertia Data
I = [15668.3       0         456.3532;...
        0      56364.66         0    ;...
     456.3532      0         71777.36].*32.2;

 Weight = 500.4041*32.2;

 %CG = [17.27607, 0, 0.15528];

%% Calculate Inertia Loads
F_Inertia = [Ny,Nz]'*Weight;
omega = [P Q R]';
omegadot = [Pdot Qdot Rdot]';

M_Inertia = I*omegadot + cross(omega,(I*omega));

Inertia = [F_Inertia;M_Inertia];

[IntTVARDervs, IntSVARDervs] = GetInterpolatedDatabaseDerivatives(Mach);


NonDimVect = [     1/(qbar*S_REF);...
                   1/(qbar*S_REF);...
                   1/(qbar*S_REF*b_REF);...
                   1/(qbar*S_REF*c_REF);...
                   1/(qbar*S_REF*b_REF)];

TrimOut = IntTVARDervs{1}\(Inertia.*NonDimVect-IntSVARDervs{1}*RateStates');

TRIMVARS = {'ALPHA','SIDES','ELEV','AILERON','RUDDER'};
fprintf('TRIMVAR      [rad]         [deg]\n');
for i=1:length(TrimOut)
    fprintf('%s\t\t%-8s     %-8s\n',TRIMVARS{i},ToShort(TrimOut(i)),ToShort(rad2deg(TrimOut(i))));
end
```

Appendix J
Unit Loads Derivative Database Generator

```
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY
function MonitorPtLoads = MakeMonitorPointDatabase_Derivatives(f06Results,TrimInput)

f = fields(f06Results);
for i=1:length(f)
    CheckVars = [];
    QBAR = TrimInput(i).QBAR;
    CheckVars = [TrimInput(i).VARS.VAL]~=0;
    if ~any(CheckVars)
        VAL = 1;
        fprintf('No Variable to divide by...\n');
        VARNAME = '';
    else
    fprintf('Non-Dimensionalizing By %s\t VAL = %8.4f\n',TrimInput(i).VARS(CheckVars).DOF,TrimInput(i).(TrimInput(i).VARS(CheckVars).DOF));
    VARNAME = TrimInput(i).VARS(CheckVars).DOF;
    VAL = TrimInput(i).(VARNAME);

    end
    for j=1:length(f06Results.(f{i}).MonitorPtLoads)
        cmpnam = strrep(f06Results.(f{i}).MonitorPtLoads(j).COMP,' ','');
        cmpnam = strrep(cmpnam,'-','');
        cmpnam = strrep(cmpnam,'(','');
        cmpnam = strrep(cmpnam,')','');
    MonitorPtLoads.(cmpnam)(i,:) = f06Results.(f{i}).MonitorPtLoads(j).VALUE/(QBAR*VAL);
    MonitorPtLoads.DIMVAR{i} = VARNAME;
    end
end
```

Appendix K
Write Trim Include

```
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY
function WriteTRIMInclude(TrimInput,outfile)
%% Script to write input file for running multiple TRIM conditions
% Description:
% TrimInput is a structured array with fields:
%         ID: TRIM ID
%         MACH: MACH
%         Q: DYNAMIC PRESSURE
%         VARS.DOF: TRIM VARIABLE DEGREE OF FREEDOM
%                 i.e. URDD3, ANGLEA, ETC.
%         VARS.VAL: TRIM VARIABLE SET CONSTRAINT VALUE
fid = fopen(outfile,'w');
fprintf(fid,'$ FILE WRITTEN: %s\r\n',datestr(now));

for i=1:length(TrimInput)
    fprintf(fid,'$ %s\r\n',char(TrimInput(i).DESCRIPTION));
    fprintf(fid,'TRIM    %8i%8s%8s',TrimInput(i).ID,ToShort(TrimInput(i).MACH),ToShort(TrimInput(i).QBAR));
    cnt = 4;
    start = true;

    for j=1:length({TrimInput(i).VARS.DOF})
        if cnt==8 && start
            start = false;
            fprintf(fid,'     1.0+\r\n        ');
            cnt = 2;
        elseif cnt==10 && ~start
            fprintf(fid,'+\r\n        ');
            cnt = 2;

        end
            fprintf(fid,'%8s%8s',TrimInput(i).VARS(j).DOF,ToShort(TrimInput(i).VARS(j).VAL));
            cnt = cnt +2;

    end
    fprintf(fid,'\r\n');
end
fclose all;
fprintf('Trim Input Deck Successfully Written!\n\n');

fclose all;

fid = fopen('Subcase.bdf','w');
for i=1:length(TrimInput)
fprintf(fid,'SUBCASE %i\r\n   TRIM = %i\r\n',TrimInput(i).ID,TrimInput(i).ID);
end


end
```

Appendix L
Non-Dimensionalize Anglular Values

```
%% DEVELOPED BY R. HAUGHEY FOR THE PURPOSE OF DATA COLLECTION FOR MASTERS THESIS AT THE FLORIDA INSTITUTE OF TECHNOLOGY
% NOT FOR PUBLIC RELEASE - PROPERTY OF RYAN HAUGHEY AND THE FLORIDA
% INSTITUTE OF TECHNOLOGY
function [NonDimRate,Rate] = NonDim(Type,Rate,TASfts)

Type = upper(Type);

b_REF = 40;
c_REF = 10;

switch Type
    case 'ROLL'
        NonDimRate = (deg2rad(Rate)*b_REF)/(2*TASfts);
    case 'PITCH'
        NonDimRate = (deg2rad(Rate)*c_REF)/(2*TASfts);
    case 'YAW'
        NonDimRate = (deg2rad(Rate)*b_REF)/(2*TASfts);
    case 'PDOT'
        NonDimRate = (deg2rad(Rate)*b_REF)/(2*32.2);
    case 'QDOT'
        NonDimRate = (deg2rad(Rate)*c_REF)/(2*32.2);
    case 'RDOT'
        NonDimRate = (deg2rad(Rate)*b_REF)/(2*32.2);
end
```

Appendix M
HA-144F Model Data

```
INIT MASTER(S)
NASTRAN SYSTEM(442)=-1,SYSTEM(319)=1
ID EDS,Femap
SOL AESTAT
TIME 10
CEND
  TITLE = BASELINE MACH SOLUTION - THESIS MODEL
  SUBTITLE = BASELINE RUN
  LABEL = HA144F - NASTRAN MODEL
  ECHO = NONE
  DISPLACEMENT = ALL
  FORCE = ALL
  STRESS = ALL
  AEROF = All
  APRES = All
  SPC = 1
  MPC = 10
  ECHO = BOTH
  DISPLACEMENT = ALL
  FORCE = ALL
  STRESS = ALL
  AEROF = All
  APRES = All
  SPC = 1
  MPC = 10
OUTPUT(PLOT)
  PLOTTER = NASTRAN
  SET 1 = ALL
  FIND SCALE, ORIGIN 1,SET 1
  PLOT SET 1
  PLOT STATIC DEFORMATION 0, ORIGIN 1, SET 1, OUTLINE
BEGIN BULK
$
PARAM,PRGPST,YES
PARAM,AUTOSPC,YES
PARAM,GRDPNT,0
CORD2C         1       0      0.      0.      0.      0.      0.      1.+FEMAPC1
+FEMAPC1      1.      0.      1.
CORD2S         2       0      0.      0.      0.      0.      0.      1.+FEMAPC2
+FEMAPC2      1.      0.      1.
$ Femap with NX NASTRAN Coordinate System 3 : Rectangular Coordinate System
CORD2R         3       0 24.2265     10.      0.24.311529.829954.9817617+
+       25.09355 10.4981.0111873
$ Femap with NX NASTRAN Coordinate System 4 : Rectangular Coordinate System
CORD2R         4       0    12.5      0.      0.    12.5      0.      1.+
+           13.5      0.      0.
$ Femap with NX NASTRAN Coordinate System 5 : Rectangular Coordinate System
CORD2R         5       0     30.      0.      0.     30.      0.      1.+
+       30.86603.4999967      0.
$ Femap with NX NASTRAN Coordinate System 20 : Rectangular Coordinate System
CORD2R        20       0     30.      0.      0.     30.      0.      1.+
+       30.86603-.499997      0.
$ Femap with NX NASTRAN Coordinate System 30 : Rectangular Coordinate System
CORD2R        30       0 24.2265    -10.      0.24.31152-9.82995 .981761+
+       25.09355-10.4981.0111894
$ Femap with NX NASTRAN Coordinate System 90 : Rectangular Coordinate System
CORD2R        90     900      5.      0.      0.      5.      0.      1.+
+             6.      0.      0.
$ Femap with NX NASTRAN Coordinate System 100 : Rectangular Coordinate System
CORD2R       100       0     15.      0.      0.     15.      0.     -1.+
+            14.      0.      0.
$ Femap with NX NASTRAN Coordinate System 110 : Rectangular Coordinate System
CORD2R       110       0 26.7265     10.      0. 26.7265     10.     -1.+
+       27.59253    10.5      0.
$ Femap with NX NASTRAN Coordinate System 210 : Rectangular Coordinate System
CORD2R       210       0 26.7265    -10.      0. 26.7265    -10.      1.+
+       27.59253   -10.5      0.
$ Femap with NX NASTRAN Coordinate System 300 : Rectangular Coordinate System
CORD2R       300       0     30.      0.      0.     30.      1.      0.+
+       29.13397      0..4999964
$ Femap with NX NASTRAN Coordinate System 301 : Rectangular Coordinate System
CORD2R       301       0    32.5      0.      0.    32.5     -1.      0.+
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+       31.63397      0..4999964
$ Femap with NX NASTRAN Coordinate System 450 : Rectangular Coordinate System
CORD2R       450       0 24.2265     10.      0. 24.2265     11.      0.+
+        25.2265     10.      0.
$ Femap with NX NASTRAN Coordinate System 460 : Rectangular Coordinate System
CORD2R       460       0 24.2265    -10.      0. 24.2265     -9.      0.+
+        25.2265    -10.      0.
$ Femap with NX NASTRAN Coordinate System 900 : Rectangular Coordinate System
CORD2R       900       0     10.      0.     1.5     10.      0.     2.5+
+            11.      0.     1.5
$ Femap with NX NASTRAN Coordinate System 100000 : Rectangular Coordinate System
CORD2R    100000       017.27607      0.  .1552817.27607      0. 1.15528+
+       18.27607      0.  .15528
PARAM,AUNITS,.031081
AESTAT         1  ANGLEA
AESTAT         2   SIDES
AESTAT         3    ROLL
AESTAT         4   PITCH
AESTAT         5     YAW
AESTAT         6   URDD2
AESTAT         7   URDD3
AESTAT         8   URDD4
AESTAT         9   URDD5
AESTAT        10   URDD6
AEROS     100000  100000     10.     40.    400.       0       0
$ Femap with NX NASTRAN Aero Control Surface 505 : ELEV
AESURF       505    ELEV      90       1      90       2             LDW+
+
$ Femap with NX NASTRAN Aero Control Surface 517 : AILERON
AESURF       517 AILERON     110       3     210       4             LDW+
+
$ Femap with NX NASTRAN Aero Control Surface 518 : RUDDER
AESURF       518  RUDDER     301       5                             LDW+
+
AELIST         1    1000    1001    1002    1003    1004    1005    1006+
+           1007
AELIST         2    2000    2001    2002    2003    2004    2005    2006+
+           2007
AELIST         3    1119    1123    1127    1131
AELIST         4    2103    2107    2111    2115
AELIST         5    3103    3107    3111
$ Femap with NX NASTRAN Constraint Set 1 : NASTRAN SPC 1
SPC1           1       1      90
$ Femap with NX NASTRAN Constraint Set 2 : NASTRAN SUPORT
SUPORT        90   23456
$ Femap with NX NASTRAN Constraint Set 3 : NASTRAN OMIT
OMIT         110       4
OMIT         120       4
OMIT         210       4
OMIT         220       4
OMIT         310       4
$ Femap with NX NASTRAN Constraint Set 10 : NASTRAN MPC 10
MPC           10      89       1      1.     890       1     -1.
MPC           10      89       2      1.     890       2     -1.
MPC           10      89       3      1.     890       3     -1.
MPC           10      89       4      1.     890       4     -1.
MPC           10      89       5      1.     890       5     -1.
MPC           10      89       6      1.     890       6     -1.
MPC           10      91       1      1.     910       1     -1.
MPC           10      91       2      1.     910       2     -1.
MPC           10      91       3      1.     910       3     -1.
MPC           10      91       4      1.     910       4     -1.
MPC           10      91       5      1.     910       5     -1.
MPC           10      91       6      1.     910       6     -1.
$ Femap with NX NASTRAN Property 100 : BAR Property
PBAR         100       1      4. .347222      .3      1.      0.       +
+             1.      1.      1.     -1.     -1.      1.     -1.     -1.+
+                             0.
$ Femap with NX NASTRAN Property 101 : BAR Property
PBAR         101       1     1.5 .173611      2. .462963      0.       +
+             .5      3.      .5     -3.     -.5      3.     -.5     -3.+
+                             0.
$ Femap with NX NASTRAN Material 1 : ISOTROPIC Material
MAT1           1  1.44+9   5.4+8              0.      0.      0.
GRID          88     900      5.     -5.      0.       0
GRID          89     900      5.     -2.      0.       0
GRID          90       0     15.      0.      0.       0
GRID          91     900      5.      2.      0.       0
GRID          92     900      5.      5.      0.       0
GRID          97       0      0.      0.      0.       0
GRID          98       0     10.      0.      0.       0
GRID          99       0     20.      0.      0.       0
GRID         100       0     30.      0.      0.       0
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GRID         110       027.11325      5.      0.       0
GRID         111       024.61325      5.      0.       0
GRID         112       029.61325      5.      0.       0
GRID         115       0 24.2265     10.      0.       0
GRID         120       021.33975     15.      1.       0
GRID         121       018.83975     15.      1.       0
GRID         122       023.83975     15.      1.       0
GRID         150       0 19.2265     10.    -1.5       0
GRID         151       0 24.2265     10.    -1.5       0
GRID         210       027.11325     -5.      0.       0
GRID         211       024.61325     -5.      0.       0
GRID         212       029.61325     -5.      0.       0
GRID         215       0 24.2265    -10.      0.       0
GRID         220       021.33975    -15.      1.       0
GRID         221       018.83975    -15.      1.       0
GRID         222       023.83975    -15.      1.       0
GRID         250       0 19.2265    -10.    -1.5       0
GRID         251       0 24.2265    -10.    -1.5       0
GRID         310       032.88675      0.      5.       0
GRID         311       030.38675      0.      5.       0
GRID         312       035.38675      0.      5.       0
GRID         890       0     15.     -2.      0.       0
GRID         910       0     15.      2.      0.       0
CBAR          89     101      88      89      0.      0.      1.
CBAR          90     101     890      90      0.      0.      1.
CBAR          91     101      90     910      0.      0.      1.
CBAR          92     101      91      92      0.      0.      1.
CONM2         97      97       0  93.243      0.      0.      0.
CONM2         98      98       0  93.243      0.      0.      0.
CONM2         99      99       0  93.243      0.      0.      0.
CBAR         100     100      90      99      0.      0.      1.
CBAR         101     100      97      98      0.      0.      1.
CBAR         102     100      98      90      0.      0.      1.
CBAR         103     100      99     100      0.      0.      1.
CBAR         110     101     100     110      0.      0.      1.
RBE2         111     110  123456     111
RBE2         112     110  123456     112
CBAR         115     101     110     115      0.      0.      1.
CBAR         120     101     115     120      0.      0.      1.
RBE2         121     120  123456     121
RBE2         122     120  123456     122
CBAR         150     101     150     151      0.      0.      1.
CBAR         151     101     151     115      1.      0.      0.
CBAR         210     101     100     210      0.      0.      1.
RBE2         211     210  123456     211
RBE2         212     210  123456     212
CBAR         215     101     210     215      0.      0.      1.
CBAR         220     101     215     220      0.      0.      1.
RBE2         221     220  123456     221
RBE2         222     220  123456     222
CBAR         250     101     250     251      0.      0.      1.
CBAR         251     101     251     215      1.      0.      0.
CBAR         310     101     100     310      0.      0.      1.
RBE2         311     310  123456     311
RBE2         312     310  123456     312
CONM2        313     100       0  93.243      0.      0.      0.
CONM2        314     111       0 18.6486      0.      0.      0.
CONM2        315     112       0 12.4324      0.      0.      0.
CONM2        316     121       0 18.6486      0.      0.      0.
CONM2        317     122       0 12.4324      0.      0.      0.
CONM2        318     211       0 18.6486      0.      0.      0.
CONM2        319     212       0 12.4324      0.      0.      0.
CONM2        320     221       0 18.6486      0.      0.      0.
CONM2        321     222       0 12.4324      0.      0.      0.
CONM2        322     311       0 1.86486      0.      0.      0.
CONM2        323     312       0 1.24324      0.      0.      0.
$ Femap with NX NASTRAN Aero Property 1000 : Aero Property
PAERO1      1000    4000    4510    4610
$ Femap with NX NASTRAN Aero Property 4020 : Aero Property
PAERO2      4020      ZY     2.5      1.       1               2        +
+              1       8
AEFACT         1      0.   1.111   1.778      2.      2.      2.     2.5+
+            2.5     2.5
AEFACT         2     45.    135.    225.    315.
$ Femap with NX NASTRAN Aero Property 4520 : Aero Property
PAERO2      4520      ZY      .5      1.       3               4        +
+              1       4
AEFACT         3      0.      .5      .5      .5      0.
AEFACT         4     45.    135.    225.    315.
AEFACT         5      0.  .16667      1.
$ Femap with NX NASTRAN Aero Panel 1000 : Aero Panel
CAERO1      1000    1000     900               4       5               1+
+             0.      2.      0.     10.      0.      5.      0.     10.
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$ Femap with NX NASTRAN Aero Panel 1104 : Aero Panel
CAERO1      1104    1000       0       3       4                       1+
+       23.55662     2.5      0.     10. 19.2265     10.      0.     10.
$ Femap with NX NASTRAN Aero Panel 1116 : Aero Panel
CAERO1      1116    1000       0       4       4                       1+
+        19.2265     10.      0.     10.13.45299     20.      2.     10.
AEFACT         6      0.  .83333      1.
$ Femap with NX NASTRAN Aero Panel 2000 : Aero Panel
CAERO1      2000    1000     900               4       6               1+
+            0.     -5.      0.     10.      0.     -2.      0.     10.
$ Femap with NX NASTRAN Aero Panel 2100 : Aero Panel
CAERO1      2100    1000       0       4       4                       1+
+       13.45299    -20.      2.     10. 19.2265    -10.      0.     10.
$ Femap with NX NASTRAN Aero Panel 2116 : Aero Panel
CAERO1      2116    1000       0       3       4                       1+
+        19.2265    -10.      0.     10.23.55662    -2.5      0.     10.
$ Femap with NX NASTRAN Aero Panel 3100 : Aero Panel
CAERO1      3100    1000       0       3       4                       1+
+        30.7735      0.     10.     10.26.44334      0.     2.5     10.
$ Femap with NX NASTRAN Aero Panel 3500 : Aero Panel
CAERO1      3500    1000       0       1       2                       1+
+        21.7265     10.      0.      5. 21.7265     10.     -1.      5.
$ Femap with NX NASTRAN Aero Panel 3600 : Aero Panel
CAERO1      3600    1000       0       1       2                       1+
+        21.7265    -10.      0.      5. 21.7265    -10.     -1.      5.
$ Femap with NX NASTRAN Aero Panel 4000 : Aero Panel
CAERO2      4000    4020       0       8       8                       1+
+            -5.      0.      0.     40.
$ Femap with NX NASTRAN Aero Panel 4510 : Aero Panel
CAERO2      4510    4520       0       4       4                       1+
+        19.2265     10.    -1.5     10.
$ Femap with NX NASTRAN Aero Panel 4610 : Aero Panel
CAERO2      4610    4520       0       4       4                       1+
+        19.2265    -10.    -1.5     10.
SET1        1000      88      89      91      92
$ Femap with NX NASTRAN Aero Spline 1502 : Aero Spline
SPLINE2     1502    1000    1000    1007    1000      0.      1.      90+
+             0.      0.            BOTH
SET1        1101     100     111     112     115
$ Femap with NX NASTRAN Aero Spline 1602 : Aero Spline
SPLINE2     1602    1104    1104    1115    1101      0.      1.       5+
+            -1.     -1.            BOTH
SET1        1102     115     121     122
$ Femap with NX NASTRAN Aero Spline 1603 : Aero Spline
SPLINE2     1603    1116    1116    1131    1102      0.      1.       3+
+            -1.     -1.            BOTH
$ Femap with NX NASTRAN Aero Spline 2501 : Aero Spline
SPLINE2     2501    2000    2000    2007    1000      0.      1.      90+
+             0.      0.            BOTH
SET1        2102     215     221     222
$ Femap with NX NASTRAN Aero Spline 2601 : Aero Spline
SPLINE2     2601    2100    2100    2115    2102      0.      1.      30+
+            -1.     -1.            BOTH
SET1        2101     100     211     212     215
$ Femap with NX NASTRAN Aero Spline 2602 : Aero Spline
SPLINE2     2602    2116    2116    2127    2101      0.      1.      20+
+            -1.     -1.            BOTH
SET1        3100      99     100     311     312
$ Femap with NX NASTRAN Aero Spline 3100 : Aero Spline
SPLINE2     3100    3100    3100    3111    3100      0.      1.     300+
+            -1.     -1.            BOTH
SET1        4001      97      98      99     100
$ Femap with NX NASTRAN Aero Spline 4000 : Aero Spline
SPLINE2     4000    4000    4000    4007    4001      0.      1.       +
+             0.     -1.            BOTH
SET1        4521     115     150     151
$ Femap with NX NASTRAN Aero Spline 4520 : Aero Spline
SPLINE2     4520    3500    3500    3501    4521      0.      1.     450+
+            -1.     -1.            BOTH
SET1        4525     150     151
$ Femap with NX NASTRAN Aero Spline 4525 : Aero Spline
SPLINE2     4525    4510    4510    4513    4525      0.      1.       +
+             0.     -1.            BOTH
SET1        4621     215     250     251
$ Femap with NX NASTRAN Aero Spline 4620 : Aero Spline
SPLINE2     4620    3600    3600    3601    4621      0.      1.     460+
+            -1.     -1.            BOTH
SET1        4625     250     251
$ Femap with NX NASTRAN Aero Spline 4625 : Aero Spline
SPLINE2     4625    4610    4610    4613    4625      0.      1.       +
+             0.     -1.            BOTH
ENDDATA
```