

Cost Benefits of Aerodynamic Data Generation Techniques for Aircraft Stability and Control Analysis using the J2 Universal Tool-Kit

John Jeffery and Paul Docksey
J2 Aircraft Dynamics Ltd.

Copyright © 2008 SAE International

ABSTRACT

In modern aircraft development, effective stability and control analysis running parallel to the aircraft design is essential to the success of the manufacturer. Numerous aircraft manufacturers have had to spend large amounts of resources and time over the years as their in flight tests show the aircraft to be an unstable design. Even worse case scenarios have resulted in the loss of passengers and crew as aircraft have not responded safely to a situation.

In order to complete stability and control analysis on an aircraft model, the aircraft's aerodynamic data is necessary. This paper investigates a series of methods currently available, in the generation of aerodynamic data and how that data relates to actual aircraft stability and control. Furthermore, the integration of the aerodynamic data will be demonstrated within the J2 Universal software. Firstly to show how to use the aerodynamic data within a state of the art stability and control design tool, and secondly to show how each method of data generation can affect the accuracy of the stability and control analysis. A summary of methods concludes this paper in the hopes of informing design engineers as to what methods would be most appropriate given a projects particular design phase. This paper aims to encourage a design engineer to maximize their knowledge of their aircraft at every stage of the design process. The methods of generating Aerodynamic data that are investigated include Wind Tunnel testing, Computational Fluid Dynamics, Boundary Element Theory, Strip Theory, DATCOM+ and more traditional geometric methods.

NOMENCLATURE

b = Span
 $\bar{c}$ = Mean Aerodynamic Chord Length
 C_D = Drag Coefficient
 C_{D_0} = Drag Coefficient at 0° incidence
 C_L = Lift Coefficient
 C_{L_α} = Lift Curve gradient
 $C_{L_{\dot{q}}}$ = Lift coefficient derivative due to non-dimensional pitch rate.
 C_M = Pitch Coefficient

D = Drag Force
 $I_{X,Y,Z}$ = Moment of Inertia about X,Y,Z axis respectively
 $J_{XY,XZ,YZ}$ = Product of Inertia about XY,XZ,YZ axis respectively
L = Lift Force
 L_A = Aerodynamic Rolling Moment
 L_R = Resultant Rolling Moment
m = Mass
M = Pitch Moment
 M_A = Aerodynamic Pitch Moment
 M_R = Resultant Pitch Moment
 N_A = Aerodynamic Yawing Moment
 N_R = Resultant Yawing Moment
P = Roll Rate
q = Dynamic Pressure
 $\dot{q}$ = Non-dimensional pitch rate
Q = Pitch Rate
R = Yaw Rate
S = Wing Reference Area
 S_s = Strip Reference area
TAS = True Airspeed
U = Forward Velocity
V = Lateral Velocity
W = Vertical Velocity
 x_h = Non-dimensional longitudinal distance from the centre of gravity to the horizontal tail aerodynamic centre
 X_A = Aerodynamic X Force
 X_R = Resultant X Force
 y_w = Non-dimensional lateral distance from the centre of gravity to the wings aerodynamic centre
 Y_A = Aerodynamic Y Force
 Y_R = Resultant Y Force
 Z_A = Aerodynamic Z Force
 Z_R = Resultant Z Force
 α = Airfoil Angle of Attack
 α_w = Wing Local Angle of Attack
 α_{AF} = Total Airframe Angle of Attack
 β = Aerofoil Side Slip Angle
 γ = Angle of climb
 ϕ = Bank Angle
 ρ = Air Density
 θ = Pitch Angle

INTRODUCTION

When considering the investigation of aircraft stability and control in the design lifecycle. It can be split into 2 areas.

- i. Estimation of modes of motion frequency and damping using linear coefficient derivatives.
- ii. Detailed investigation of flight dynamics and aircraft behaviour through simulation and response modelling

When considering simulation and response modelling, it is possible to extend investigation to utilize non-linear derivatives, and perform complete FAA certification procedures. Experience has found that the first approach is used in the early stages of design whilst the second approach is left until much later on in the lifecycle when the costs of making modifications to the design can be significantly increased. This has often been due to the level of data that is required as entry into the available tools in order to start to perform the analysis. The expense of generating the data to provide "sufficient" information is to investigate behaviour is often seen as the primary driver in leaving the analysis until later on such that fewer options are being investigated.

The intention of this paper is to look at what can be discovered when performing response modelling using differing levels of aerodynamic data generation that provide different levels of fidelity. The paper will document the different methods of aerodynamic data generation, and provide an indication as to the level of time and costs that was required to generate the data, and then using those different techniques on a single aircraft project to see what characteristics can be spotted. In order to do the simulation and analysis, the J2 Universal Tool-Kit will be used that can take data from any source and very quickly enable the user to create a model and then "fly" it across the complete flight envelope. To ensure consistency, the basic aircraft structure will be built using J2 Builder and will contain the structure of the aircraft, and any mass & inertia information necessary to calculate complete Mass, inertias and centre of gravity. Several "delta" models will then be created into which the aerodynamic data will be added from the variety of methods. This ensures consistency of structure and mass information across all the models under comparison and allows for changes to be made on the initial model and be instantly reflected in all "deltas" thus avoiding the potential for discontinuities as the analysis progresses.

Using each of these delta models a consistent set of analyses will be performed, using J2 Freedom, to investigate the stability and control, and behaviour of the aircraft. These analyses will include.

- i. Longitudinal static stability such as angle of attack to trim and elevator to trim across a range of airspeeds and altitudes.
- ii. Longitudinal dynamic stability including investigations into time histories across the complete flight envelope

- iii. Lateral dynamic stability including investigations into time histories across the complete flight envelope
- iv. A selection of FAA certification manoeuvres to view key characteristics.

A comparison of the flight results for each analysis phase, and comments relating to how they compare to the wind tunnel (baseline) data will be presented. The results will then be contrasted with the costs in generating the different data types, and recommendations as to how each method can be used throughout the design lifecycle to enable engineers to optimise designs through using the different techniques to consider different configurations and scenarios throughout the project. This will show the most cost effective method of generating an aircraft's aerodynamic data for a given phase of the design process as well as highlight how early the stability and control of an aircraft can actually be estimated within a certain degree of accuracy.

AERODYNAMIC DATA GENERATION

The topic of aerodynamics is rudimentary to understanding the stability and control of an object in flight. It is a specific branch of fluid and gas dynamics that focuses on the specific medium of air and the how it interacts with a moving object within the medium. Studying the flow of air around a moving object allows for the objects aerodynamic forces and moments to be calculated. These forces and moments determine the flight derivatives that power the flight dynamic equations. Knowing the results of these equations allows for an aerospace engineer to understand how an aircraft will respond to a given flight condition or situation. Reference [1] contains a description of basic aerodynamic theory that will be utilised later in this paper to drive the flight dynamic equations. Below we can see the basic methods

SIMPLE GEOMETRIC MODELLING

Simple geometric modelling is a term used at J2 Aircraft Dynamics to represent the construction of simple models to begin the investigations into an aircraft's stability and control and dynamic behaviour. This approach concentrates primarily upon the longitudinal characteristics of an aircraft by breaking it down into a series of trapezoidal wing shapes. Each of these wings has the 2-D lift, drag and pitching moment characteristics calculated for the appropriate airfoil section. These 2-D characteristics can be found from various estimating techniques or applications such as JavaFoil (Reference [2]) or XFoil (Reference [3]), or from experimental data contained in numerous texts or in-house databases.

The aspect ratio is used for an approximate correction of the results for a finite wing. The 3D lift coefficient CL is determined by adapting the 2D CL . Mach number and aspect ratio are taken into account. Then the 3D drag coefficient CD is calculated by adding the induced drag coefficient for a wing with elliptical lift distribution to the Cd of the airfoil.

Once each item has been estimated, a hierarchical model can be constructed using the location of each item, and approximations to the aerodynamic centre of each item. Each entity can be constructed with a local angle of attack, such that factors such as variable incidence can be considered. The final stage is to look at the influence of 1 surface on that of another. The most obvious aspects to consider are upwash from a canard or downwash due from the wing on a horizontal tail. This can be built into the model constructed with J2 Builder through User Defined parameters, and can be calculated from expressions and tables provided by Roskam (Reference [4]). This now provides a static model, but this does not provide any indication of dynamic characteristics. In order to introduce the dynamic characteristics we need to look to Roskam^[4] again and look at how dynamic derivatives are estimated.

For example, the main aspect of Pitch Damping is due to the change in angle of attack on the horizontal tail due to a change in velocities caused by the pitching motion. This resulting in a change in lift combined with the moment arm produces a pitching moment that opposes the pitching motion. This approach gives us the following equations.

$$\Delta W = Q * x_h * \bar{c} \quad 1$$

With small angle theory we can assume

$$\Delta \alpha = \Delta W / U \quad 2$$

This means that the change Lift can be estimated to

$$\Delta C_L = \Delta \alpha * C_{L\alpha} \quad 3$$

$$\Delta C_L = \frac{Q * x_h * \bar{c}}{U} * C_{L\alpha} \quad 4$$

$$\Delta C_L = 2\hat{q} * x_h * C_{L\alpha} \quad 5$$

Thus we can get the derivative for the change in lift due to non-dimensional pitch rate from:

$$C_{L\hat{q}} = 2 * x_h * C_{L\alpha} \quad 6$$

The derivative will also be the major factor in pitch damping. If necessary it is also possible to add in the change in the local pitching moment using the same approach, but this has a much small influence on the overall pitch damping.

Roskam^[4] also gives us similar expressions for the change in lift due to rate of change of angle of attack as a function of the downwash gradient.

When working with lateral dynamic derivatives, a similar approach to the Pitch Damping can be used to estimate the Roll Damping derivatives. In order to do this the aircraft model must have separate items for left and right wings, horizontal tail etc, and then use the lateral position of the aerodynamic centre to provide the changes in

vertical velocity. The effects of Yaw Rate on Horizontal surfaces is a little more complicated as the Yawing Moment results in a change in forward velocity rather than vertical velocity so this cannot be added as simply and needs to be incorporated into the local angle of attack values rather than as an increment.

$$\Delta U = R * y_w * b \quad 7$$

$$\Delta \alpha = \alpha - \frac{W}{U - \Delta U} \quad 8$$

$$\alpha_w = \alpha_{AF} + \Delta \alpha \quad 9$$

As can be seen, with very simple information relating to the basic aerodynamic characteristics of individual items as well as some simple geometric data it is possible to start looking at the behaviour of the aircraft.

AERODYNAMIC STRIP THEORY

Aerodynamic Strip Theory (AST) or Blade Element Theory (BET) is a method of approximating a 3-dimensional lifting surface by breaking it down into 2-dimensional airfoil strips along its span. Internally, within the J2 Elements software, each strip has a local angle of attack calculated based upon the strips individual motion through the air and its orientation to the overall airframe. This method can therefore automatically include the dynamic motions and angular rates as well as geometric characteristics and thus allow for dihedral and twist to be used. However there is no cross span aerodynamic interactions between strips. Figure 1 shows how a basic wing shape can be approximated into individual elements, and the distribution of the aerodynamic centre of each strip..

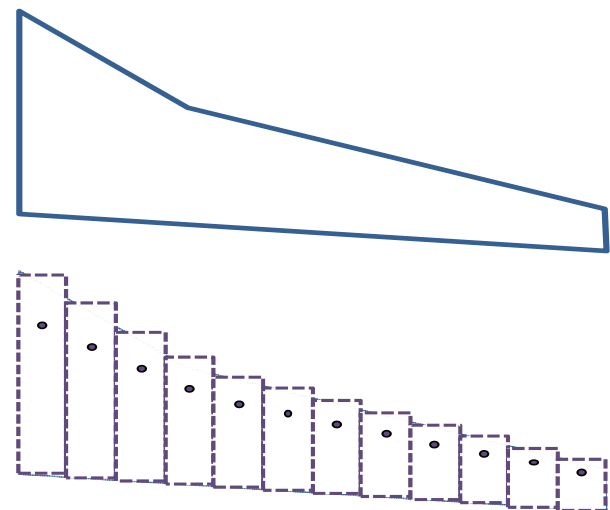


Figure 1 Lifting Surface Approximation into Strips

As with the Simple Geometric Models, the aerodynamic coefficients can be calculated in their standard non-dimensional forms C_L , C_D and C_M from any variety of methods. Each strip is assigned aerodynamic coefficients according to its airfoil section.

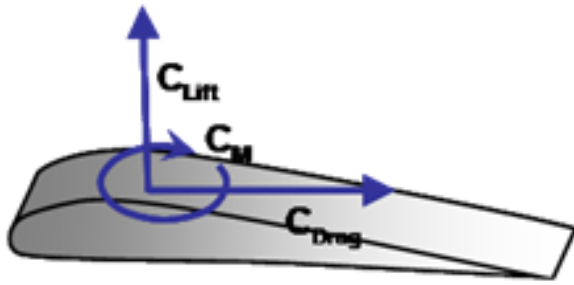


Figure 2 Local Forces acting on a Single Strip

Using Equations 10 to 12 we can then calculate the resultant local force and moment on each strip.

$$L = C_L * \frac{1}{2} \rho \text{ TAS}^2 S_s \quad 10$$

$$D = C_D * \frac{1}{2} \rho \text{ TAS}^2 S_s \quad 11$$

$$M = C_m * \frac{1}{2} \rho \text{ TAS}^2 S_s C_s \quad 12$$

It is a simple case of superposition to calculate the total effects of each element strip on the entire wing to generate a Single Lift, Drag and Pitch value for the lifting surface.

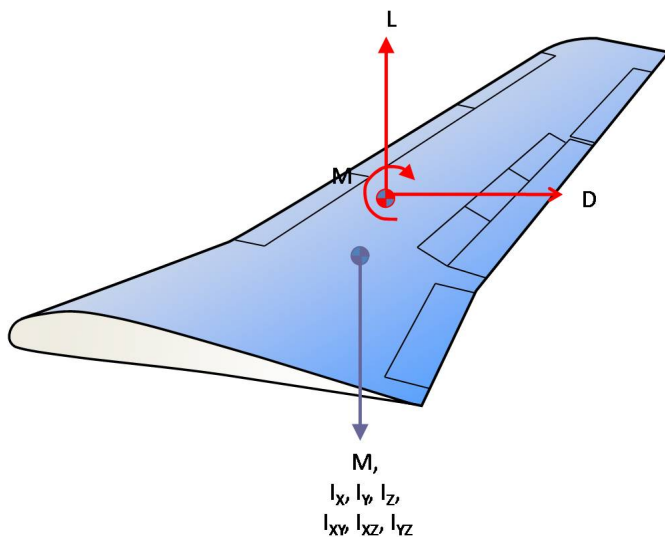


Figure 3 Total Forces acting on a Lifting Surface

USAF DIGITAL DATCOM

The Digital Datcom is a computerized version of the United States Air Force Data Compendium which is a significant collection of design notes used to predict fixed wing aircraft aerodynamic derivatives and stability & control characteristics. There are design limitations when designing an aircraft with the digital Datcom, multiple aircraft configurations may have to be analyzed to determine each contribution to the total aircraft. For example, in order to analyze a standard Wing-Body-

Horizontal Tail with movable flaps on the wing, the aircraft must be first analyzed without the Horizontal Tail to find the contribution the flaps will have, and then a second analysis that has the horizontal tail and no flaps. This is because the Digital Datcom assumes that the moving control surface is located on the most aft lifting surface which would have been the Horizontal Tail.

Once the Aircraft Body and lifting surfaces have been defined, the program is able to calculate the aircrafts aerodynamic Coefficients over a range of flight conditions, using the aircrafts Mach and Reynolds Numbers to define the flight conditions. These can be satisfied by mixing the following variables, Mach number, velocity, Reynolds number, altitude, and pressure and temperature. The Datcom program can then calculate the standard aerodynamic coefficients and using small perturbation theory, the aerodynamic derivatives can be manually calculated.

VORTEX LATTICE MODEL

The Vortex Lattice analysis model is a simplified computational fluid dynamic process that allows for an aircrafts aerodynamics to be calculated with little effort. A vortex-lattice model is best suited for aerodynamic configurations which consist mainly of thin lifting surfaces at small angles of attack and sideslip. These surfaces and their trailing wakes are represented as single-layer vortex sheets, discretized into horseshoe vortex filaments, whose trailing legs are assumed to be parallel to the x-axis, see Figure 4.

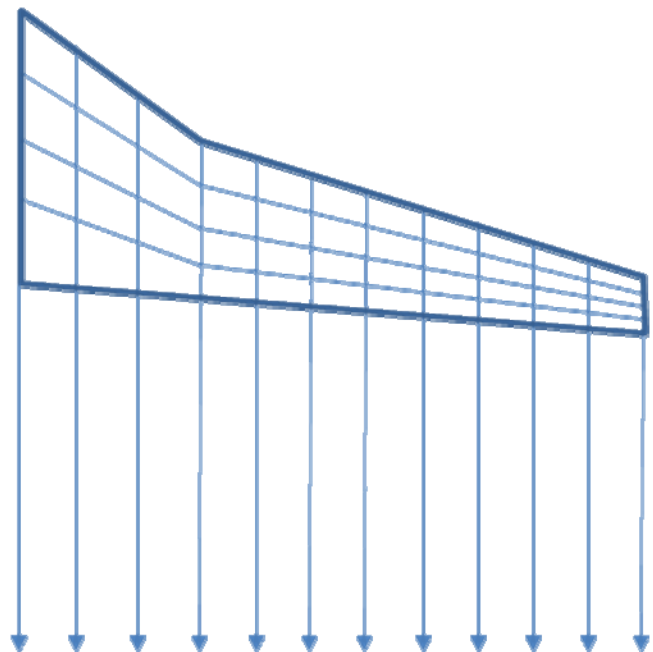


Figure 4 Vortex Lattice Panels

This technique uses the Biot-Savart Law to relate velocity induced by a vortex to the strength and orientation of the vortex. This method allows for downwash effects to be included if the upwash geometry is within the intersecting plane of the downwash object.

Building up individual panels that interact along the downwash lines can be used to calculate an aircrafts aerodynamic forces and moments.

COMPUTATIONAL FLUID DYNAMICS

Computational Fluid dynamics (CFD) is considered to be the most powerful and accurate way of modelling and simulating the aerodynamic forces around an object. Full 3 dimensional objects can be analyzed to determine the flow around each item. There are numerous methods available that can consider anything from simple laminar flow up to and including methods that are able to approximate turbulent effects. At present there are no analysis models that exist that can fully simulate turbulence, but approximate analysis models, when used by an experienced user can prove to be very accurate.

The 3-dimensional model and the control volume around it are initially discretized into small control volumes. Boundary conditions are set to initialize the objects flight conditions, and an iterative numerical method is used to calculate the effect of the boundary conditions and how they interact with the object. This method is the most costly in terms of computing power and time required to set up. A skilled operator with significant experience is often needed in order to create an accurate model.

WIND TUNNEL

Wind tunnel testing is the final method of simulating an aircrafts aerodynamics before actually constructing the full aircraft. A physical scaled model of the concept aircraft is created and placed in the wind tunnel. The atmospheric conditions are modified using standard Reynolds numbers to match the full scale flight atmospheric conditions. The aerodynamic forces and moments are easily derived using established practical techniques.

STABILITY ANALYSIS

The analyses performed on the models are based around assessing the static and dynamic stability utilising a 6-Degree of Freedom (6DoF) model. These models can be used to assess steady state conditions and trends such as angle of attack and elevator to trim; these trim conditions can then be used in time based simulations to look at the modes of motion of the aircraft as well as further investigations into FAA regulatory manoeuvres. Using these techniques it is possible to assess the complete envelope at any stage in the aircrafts lifecycle to ensure certification.

EQUATIONS OF MOTION

The 6DoF model allows for Forces and Moments in all 3 axes to affect the motion of the aircraft simultaneously. Equations 13 to 18 show the resultant forces and moments on the aircraft as a result of aerodynamic, inertial and gravitational effects.

$$X_R = X_A + m(WQ - VR) - mg \sin \theta \quad 13$$

$$Y_R = Y_A + m(UR - WP) + mg \cos \vartheta \sin \varphi \quad 14$$

$$Z_R = Z_A + m(VP - UQ) - mg \cos \vartheta \cos \varphi \quad 15$$

$$L_R = L_A - (-J_{xz}P - J_{yz}Q + I_zR)Q + (-J_{xy}P + I_yQ - J_{yz}R)R \quad 16$$

$$M_R = M_A - (I_xP - J_{xy}Q - J_{xz}R)R + (-J_{xz}P - J_{yz}Q + I_zR)P \quad 17$$

$$N_R = N_A - (-J_{xy}P + I_yQ - J_{yz}R)P + (I_xP - J_{yz}Q - J_{xz}R)Q \quad 18$$

These equations can be used to calculate the resultant linear and angular accelerations from Equations 19 to 24

$$m\dot{U} = X_R \quad 19$$

$$m\dot{V} = Y_R \quad 20$$

$$m\dot{W} = Z_R \quad 21$$

$$I_x\dot{P} - J_{xy}\dot{Q} - J_{xz}\dot{R} = L_R \quad 22$$

$$-J_{xy}\dot{P} + I_y\dot{Q} - J_{yz}\dot{R} = M_R \quad 23$$

$$-J_{xz}\dot{P} - J_{yz}\dot{Q} + I_z\dot{R} = N_R \quad 24$$

From the Accelerations it is possible to then integrate to find the Linear and Angular velocities and further integrate to obtain the resultant positions.

STATIC STABILITY

Static stability can be assessed by investigating the various states and control settings to trim the aircraft. Several types of analysis are available, but for the purposes of this paper we will investigate Angle of attack and Elevator Deflection required to trim over a range of airspeeds. For a conventional aircraft, such as the one being assessed, as the airspeed increases so the angle of attack required to trim should go from a decrease (go from a larger positive number to a smaller positive number) from however the elevator deflection is expected to increase (go from a larger negative number to a smaller negative number).

DYNAMIC STABILITY

The dynamic stability of a vehicle denotes the complete study of the motion occurring after the vehicle has been disturbed from its trimmed state. If the aircraft returns to equilibrium without overshoot, the motion is a simple subsidence. If the disturbing moment produced by the deviation tends to overshoot, then an oscillatory motion is induced. If the oscillations continue to increase, or the aircraft does not return to a steady state, then the motion is

diverging, and the aircraft is unstable. The different types of motion can be seen in Figure 5 Definition of Motion Types

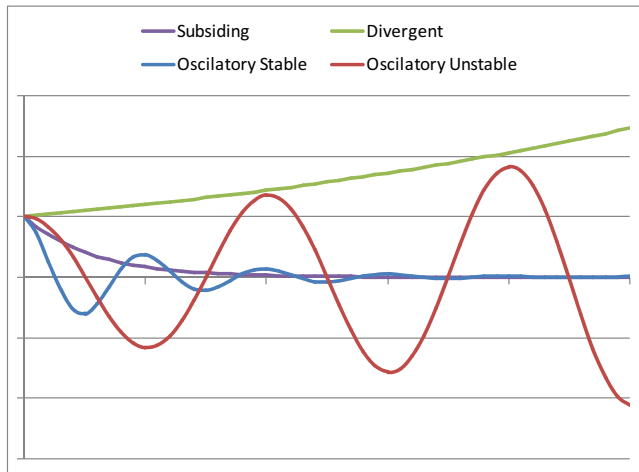


Figure 5 Definition of Motion Types

Several Standard modes of motion exist for conventional aircraft, these are:

- **Short Period - Longitudinal**

The Short Period Oscillation or rapid incidence adjustment is an aircraft mode that relates a sudden change to the aircrafts pitch and its corresponding response. In flight, the Short period can be typically initiated by either a vertical gust disturbance or by a pilot induced change to the elevator deflection. Typically the aircraft will return to its original pitch state in the region of seconds.

- **Phugoid - Longitudinal**

The Phugoid is best described as an exchange of an aircrafts kinetic and potential energy at an approximately constant angle of attack. This is visualized as speed and height oscillations. An example of initiating the Phugoid would be an increase in aircraft velocity that would see an increase in aircraft height followed by the aircraft decelerating; the aircraft will then proceed to descending and as a result, accelerate. The Phugoid is often referred to as the long period as it can take significantly longer to return to its original state, larger aircraft can have a Phugoid period in the region of minutes.

- **Spiral Mode - Lateral**

The Spiral Mode is a slow and typically unstable aircraft mode that can be potentially dangerous if unattended. The spiral mode can be initiated by a small perturbation that induces a rolling moment which has the coupled effect of causing a sideslip. This sideslip causes the aircraft to slowly drift downwards. The initial rolling moment is amplified as the tail plane, now with a side slip angle starts to produce a larger rolling moment. Thusly, left unchecked the aircraft can continue to spiral at an increasing rate and result in a loss of the aircraft.

- **Dutch Roll - Lateral**

The Dutch Roll is a damped oscillatory motion in yaw that is coupled with a roll motion. The Dutch Roll is initiated by for example a gust impacting on the vertical tail and creating a yawing moment. This Yawing moment is oscillatory and will attempt to return to its initial state. As this Yawing moment occurs, an oscillatory differential in Lift and Drag will occur on the horizontal lifting surfaces causing a roll moment. The combined motion of Yaw and Roll will create an oscillatory cycle that decays over time.

- **Roll Subsidence - Lateral**

The roll subsidence is simply a stable exponential mode that involves mainly the roll rate and corresponding bank angle. The aircraft roll angle response to lateral control inputs is an important part of the handling qualities requirements.

FEDERAL AVIATION REGULATIONS

The Federal Aviation Regulations (FAR) provide airworthiness standards for the issuance of type certificates for differing categories of aircraft. FAR 23 is applicable to small airplanes, and it is these that will be used to investigate the aircrafts behaviour. We will investigate 2 FAR manoeuvres during the analysis.

- **FAR 23.157a Rate of Roll**

It must be possible using a favourable combination of controls to roll the airplane from a steady 30° banked turn through an angle of 60° so as to reverse the direction of turn within 5.2s, with the critical engine inoperative.

METHODOLOGY

The Aim of this paper is to encourage the analysis of aircraft stability and control as early as possible during the design process and to show how this can be done at an earlier stage than is the standard design process. This paper has so far outlined a series of different methods concerning the generation of aerodynamic data for a concept aircraft and how the data can be used to analyze an aircraft's stability and control.

In order to demonstrate this approach, we will examine a Very Light Jet design (Figure 6 Very Light Jet Design) and compare the methods of aerodynamic data generation. The comparison will not be limited to data accuracy, but will also review the costs of using such processes in terms of time, financial cost and worker experience. The final review will show which process will be most cost effective given a concept aircrafts design stage.



Figure 6 Very Light Jet Design

Whilst a variety of software tools and approaches may be used in the generation of the aerodynamic data, all analysis will be performed using the J2 Universal Tool-Kit. This enables a consistent approach to be used with any aerodynamic data source from conceptual design right through to supporting flight test, as well as providing integrated graphics and visualisation to help fully understand the behaviour.

BASELINE AIRCRAFT MODEL BUILDING

All aircraft models were built using J2 Builder, the aircraft model construction tool of the J2 Universal Tool-Kit.

The first stage was to build a model that defines the structure of the aircraft, and contains references to all dynamic surfaces, and any structural items, as well as the

wing reference geometry. Engine Details are also added on to this model to provide a consistent aircraft for all cases. The resultant model was only lacking the aerodynamic data (Figure 7). The basic reference location for the aircraft has been placed at the ground and on the nose and along the centre line of the aircraft and all subsequent reference locations have been placed relative to this.

- **Mass and Inertias**

The airframe structure has been broken down into 5 mass components to enable different mass and payload scenarios to be investigated, these are:

- **Airframe**
- **Left/Right Fuel Tanks**
- **Pilot**
- **Payload**

The corresponding centre of gravity and mass values (Gross Weight) are shown in Table 1. The fuel tanks and payload are treated as point masses, and their inertial contributions will be calculated automatically. The airframe has inertia about its own cg position, with its overall contribution to the aircraft automatically calculated using the parallel axis theorem.

Airframe Inertia

- **$I_x = 188,270 \text{ lbft}^2$**
- **$I_y = 412,450 \text{ lbft}^2$**
- **$I_z = 149,674 \text{ lbft}^2$**

		Fuel	Fuel				
		Left	Right	Airframe	Pilot	Payload	Total
Mass	lb	831	831	3797	113	678	6250
Absolute CG Position							
x	in	-220	-220	-214.5	-110	-155	-207.62
y	in	-70	70	0	0	0	0.00
z	in	-70	-70	-105	-100	-100	-95.06

Table 1 VLJ Mass and Inertia Distribution

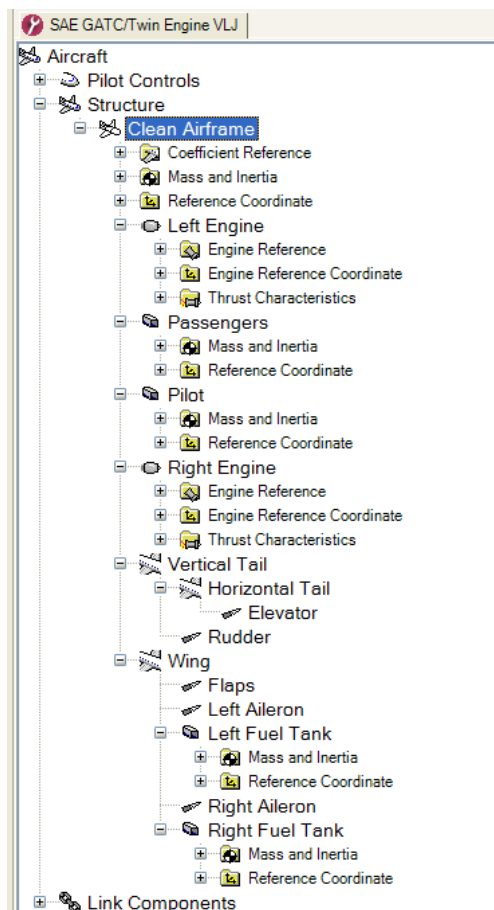


Figure 7 Basic Aircraft Structure

Engines

The engines are mounted on the fuselage and will also be given local reference coordinates to enable engine out scenarios and the impact of the centre of gravity and thrust line to be modelled. This information can be found in Table 2.

		Left Engine	Right Engine
Absolute C of T Position			
x	in	-239.116	-239.1
y	in	-49.515	49.52
z	in	-104.554	-104.6
Tilt	°	0.5	0.5
Toe	°	1	-1

Table 2 VLJ Engine Location Information

ADDING AERODYNAMIC DATA

Once the Baseline Model had been built, it was possible to add the Aerodynamic Data using the Delta functionality built into the J2 Universal Tool-Kit. This enabled us to take a virtual copy of the Baseline Aircraft and just add the remaining data without the need to produce multiple

models each containing the Baseline information. In this way we created several "delta" models each containing just the relevant aerodynamic data from each source. This also meant that if any corrections were needed to the Baseline model these were automatically transferred to the aerodynamic models without having to go through each one individually to ensure that the correct changes had been made.

The deltas were constructed as follows:

Simple Geometric Model

In this scenario and to speed up the model generation we just put together a Simple Geometric Model to consider the longitudinal characteristics of the aircraft. 2-D Aerodynamic coefficients were calculated for the Wing, Horizontal Tail, and Vertical Fin airfoils, about their quarter chord locations, using JavaFoil^[2] and these adjusted for aspect ratio. Contributions were then added to the wing and horizontal tail for flap deflections and elevator deflection.

We can add to the delta model the location of the reference coordinates for the wing, tail and fin, and from their geometry we can calculate the locations of their aerodynamic centres. With the reference information, it is then simply a case of adding the aerodynamic coefficients.

For the wing, a User Defined Parameter was created to produce a local angle of attack based upon the wing incidence and the airframe angle of attack. All coefficients were then described via look-up tables using the local value. In this way we are able to change the incidence of the wing and the aerodynamic contributions will be updated automatically. The flap contributions were also added, but these used the Flapped Wing Area only as a reference area.

For the Horizontal Tail, we used User Defined parameters to calculate the downwash gradient due to the wing, and this was then used to calculate a local angle of attack for the horizontal tail. As the Elevators were assumed to be full span, we could add the aerodynamic coefficients as 2-D Look-Up tables dependent upon the local angle of attack and the elevator deflection. As described previously the dynamic derivatives for Lift due to pitch rate and rate of change of angle of attack, we also added to the horizontal tail using the expressions described previously and the lift curve slope defined as a function of the local angle of attack.

Finally the Vertical Fin contributions were included. As we were only investigating longitudinal modes with this approach, it was only necessary to add the C_{D0} value for the Fin and its reference information. The contributions of this value to the overall pitching and drag of the aircraft will be automatically calculated.

- **Aerodynamic Strip Theory**

The aerodynamic strip theory model was constructed similarly to the Simple Geometric Model. However in this scenario, we could start to investigate the effect of lateral characteristics and automatically calculated dynamic derivatives. When describing the AST model, we were able to take greater account of the twist and dihedral of the wing as well as the sweep.

The AST models have been entered using the integrated Strip Theory capability of the J2 Elements part of the J2 Universal Tool-Kit.

In this case we add the wing and enter the twist distribution, aerodynamic centre distribution, of each strip in terms of their lateral location. Small sections of stripped items were added to account for the increments due to flap deflection and in this case aileron deflections. By giving each of these items physical locations relative to the reference location of the wing, the moment contributions in terms of lifting increment and moment arm about the CG are automatically calculated via the software.

The Horizontal Tail is added similarly, to the wing, but as for the simple geometric model, a full span elevator is assumed, and the coefficients are added as functions of the local angle of attack for each strip and the elevator deflection. In calculating the local angle of attack for a strip for the horizontal tail, this is performed automatically from the local velocities due to the geometry and the centre of gravity. The downwash contribution calculations are extended from a downwash gradient, to produce an additional velocity contribution that is automatically added to the existing velocities and resulting in a new local angle of attack.

The Vertical Fin is added as a stripped item excluding the dorsal strake, the rudder was again assumed to be full span. Once the coefficient data has been added in terms of sideslip and rudder deflection the software automatically takes into account locations and local velocities to calculate resulting moments and forces.

- **Digital Datcom**

The Digital DatCom is a programme developed from the USAF Data Compendium during the 1970's, which contained over 3000 pages on the analysis of aircraft aerodynamics, stability and handling characteristics. This digital version allowed for a significantly reduced design time.

Flight conditions are first set that are to be analysed, any combination of flight conditions can be defined as long as the Mach and Reynolds numbers can be calculated, as the DatCom uses these numbers for calculations. Only a maximum of 20 flight conditions can be calculated during any single run, thus multiple DatCom runs will be necessary to extend the flight envelope.

The aircraft fuselage is analysed as 20 elliptical cross sections along the length of the aircraft. The user defines the positioning of the wing, horizontal tail and vertical tail in the x and z directions, and the plan form of the aerodynamic item. Using these it is a simple procedure to define the variation of twist along the aerodynamic item by defining the root incidence angle and the twist angle across the span, which is of course an essential component to the aerodynamics. The horizontal and vertical tail were defined using NACA inputs, however the wing section is not a standard NACA section so the X chord, Y upper and Y lower positions where utilised, DatCom allows for 50 points to be defined.

Control surfaces can be defined on longitudinal surfaces, i.e. flaps, elevators and ailerons; however rudder deflections cannot be defined. If a control surface is defined then it is assumed to be the most aft lifting body. For the purposes of this analysis, only the elevator effects will be analysed. This is as a result of DatCom's restrictions in lateral motions, and only longitudinal trimming will be analysed. This will also simplify the DatCom post processing, as multiple control surface effects need multiple DatCom runs and their contributions to be manual adjusted.

With the aerodynamic data generated, the relevant data can easily be extracted into a useable format. The J2 aircraft model is developed to include lift, drag and pitch coefficients in relation to important variables such as angle of attack, air speed, altitude and elevator effects. We are now able to trim the aircraft longitudinally and develop a flight envelope.



Figure 8 DatCom Model

- **Vortex Lattice Model**

The vortex lattice method is among one of the earliest computational techniques for developing aircraft aerodynamics. The method is based on solving Laplace equations which will solve for the potential flow of the aircraft.

For this analysis, the Athena Vortex Lattice programme will be used which is available for free under the GNU General Public License. It was developed during the late 1980's at MIT and has been consistently used in industry since with positive results

The aircraft analysis is split into 3 distinct sections. The configuration geometry is defined in a file with each aerofoil is split into surface sections. For a symmetrical aircraft it is possible to duplicate about a Y location which can be user defined, in this case as in many cases, the Y location was set to zero.

Positioning a surface section involves defining the leading edge point in the x, y and z axis, sections can be created to account for sweep and dihedral changes. The chord and angle inputs define chord length and local angle incidence respectively. The programme will linearly interpolate these values for defining sections. The aerofoil definition is stored as a series of coordinates in a separate data file.

Control surfaces are defined within an aerofoil at sections within the surface. The hinge location is defined as a percentage of the chord and the line at which the hinge acts is defined as a vector in terms of x, y and z.

The fuselage is defined as a series of circular cross sections at various X locations (Fuselage Stations)

AVL allows for the mass and inertias of individual component to be defined and the overall aircraft mass and inertia to be calculated. In this situation the total aircraft mass and inertia values are already known.

A file is created for defining the aircraft flight conditions, the run file needs to be generated by first loading the aircraft and mass files and exporting from Athena. It is easy to modify the run file to set the Mach and velocity values which would be zero as a default, and to create multiple case conditions, i.e. alpha or beta angles.

For the analysis performed here 2 flight conditions were considered a low speed low altitude, (100kts at 300ft) and a high speed high altitude case (300kts at 34,000ft) however to get a good range of points, for look-up tables all 4 corners were analysed. When considering it is also necessary to vary the aircraft states and surface deflections, the total number of cases generated to be analysed was 68. Each case has to be individually run and saved. This can take a significant amount of time to run, and post processing of the data into a useable format can take some time.

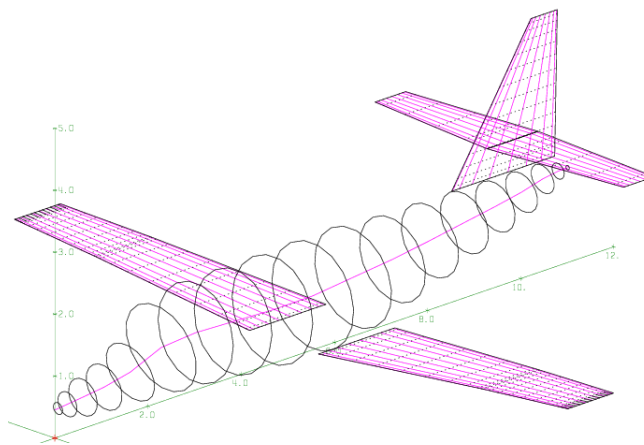


Figure 9 AVL Model

Once all the VL cases had been run, it was then necessary to establish the derivatives, and this was best performed using regression analysis this resulted in the following set of Derivatives and Factors found in order to calculate the resultant coefficients.

$$C_{Lift} = C_{Lift_0} + C_{Lift_\alpha}\alpha + C_{Lift_\beta}\beta^2 + C_{Lift_{\hat{q}}}\hat{q} + C_{Lift_{de}}de \quad 25$$

$$C_{Drag} = C_{Drag_0} + C_{Drag_{C_{Lift}}}C_{Lift} + C_{Drag_{C_{Lift^2}}}C_{Lift}^2 + C_{Drag_{da}}da^2 + C_{Drag_{de}}de^2 + C_{Drag_{dr}}dr^2 \quad 26$$

$$C_M = C_{M_\alpha}\alpha + C_{M_{\hat{q}}}\hat{q} + C_{M_{de}}de \quad 27$$

$$C_{Side} = C_{Side_\beta}\beta + C_{Side_{\hat{p}}}\hat{p} + C_{Side_{\hat{r}}}\hat{r} + C_{Side_{da}}da + C_{Side_{dr}}dr \quad 28$$

$$C_L = C_{L_\beta}\beta + C_{L_{\hat{p}}}\hat{p} + C_{L_{\hat{r}}}\hat{r} + C_{L_{da}}da + C_{L_{dr}}dr \quad 29$$

$$C_N = C_{N_\beta}\beta + C_{N_{\hat{p}}}\hat{p} + C_{N_{\hat{r}}}\hat{r} + C_{N_{da}}da + C_{N_{dr}}dr \quad 30$$

• Wind Tunnel

Once the wind tunnel data has been reduced, a spreadsheet is generated that provides all static derivatives for combinations of Flaps, Elevator, Aileron, and Rudder contributions. DAR Corp's AAA is also used with this data to generate dynamic derivatives. As all the coefficients and derivatives for the wind tunnel model have been calculated about the same point using the same reference areas, there is no need to provide additional geometric information to construct the model. All that is required is to add the location about which all coefficients are assumed to act (25% Wing m.a.c.).

From here it was simply a case of importing the look-up tables directly into the delta model.

During the construction of each of the models notes were taken in relation to the ease with which aerodynamic data could be generated, the time/cost associated with each method, the technical expertise required and the ease with which aircraft configuration changes could be made. Further notes were then taken as to how easy it is to add these data into the aircraft via J2 Builder.

ANALYSING THE AIRCRAFT

Once all the models have been created, it was necessary to perform the analysis. The first stage was to investigate the steady state stability. This was done by creating a Trim Model in the J2 Freedom component of the J2 Universal Tool-Kit that covered the complete flight envelope airspeed and altitude combinations. The full non-linear root solver built into J2 Freedom was then used to trim each aircraft model at each location defined in the Trim Model to establish the trim conditions.

In the cases where the coefficient data is simply linear derivatives and (e.g. Vortex Lattice), it was able to trim the aircraft at even the highest altitude and slowest airspeed combinations as the aircraft could effectively fly at any angle of attack to generate suitable lift. When the results were therefore plotted, angles of attack above 20° were discounted.

From the Trim conditions, we were able to Plot the Angle of Attack to Trim and Elevator to Trim values for each model across the complete flight envelope utilising the integrated graphics in J2 Visualize.

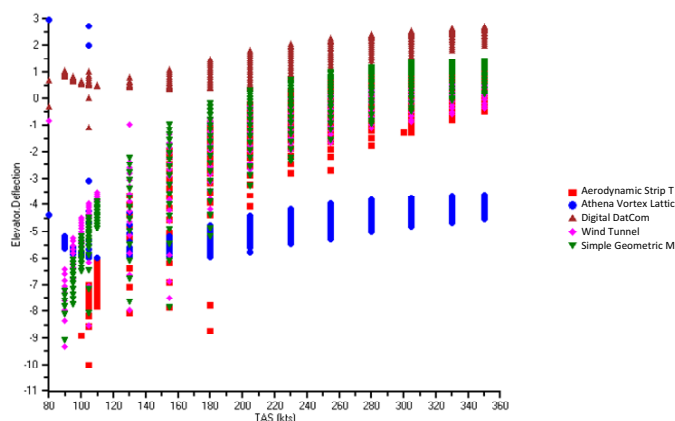


Figure 10 Elevator Deflection vs. TAS

Figure 10 shows the variation of elevator deflection along a range of TAS values. It can be noted that the simple geometric and elements models have a high degree of similarity to the high precision of the wind tunnel results. This accuracy can again be seen in figure 11 which shows the variation of angle of attack along a range of TAS values. It can be seen that the least accurate of methods are the Datcom and Athena methods which over and under predict the elevator deflection respectively.

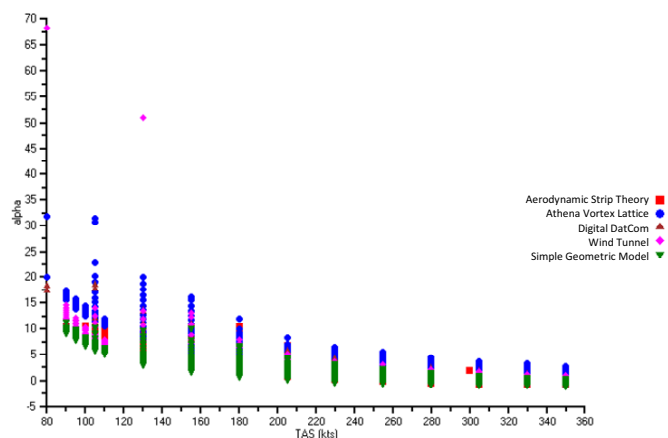


Figure 11 Angle of attack vs. TAS

Having established the steady state conditions for each model, these are automatically stored back into the database, and as such can be re-used as initial conditions for the dynamic analysis.

The initial dynamic analysis was to look at the modes of motion for the aircraft. This was performed on all models at the cruise condition and at a slower speed with the flaps in the take-off configuration. A series of Response Models were built using J2 Freedom, to excite each of the modes of motion. The manoeuvres were:

- Short Period**
 Initial disturbance of the Pitch Rate from the trimmed condition by $+10^\circ/\text{s}$
- Phugoid**
 Initial Airspeed disturbance from the trimmed condition of $+10\text{kts}$
- Spiral Divergence**
 Initial disturbance of the Bank Angle from the trimmed condition by $+10^\circ$
- Dutch Roll**
 Rudder Pedal Doublet after 2s. The pilot produce a stepped input into the pedals first in one direction then the other resulting in a 3° rudder deflection for 0.5s and then -3° rudder deflection for a further 0.5s then returning the rudder to the central position.
- Roll Subsidence**
 A small roll stick step input is added after 2s. The Objective is to view the maximum roll rate for a given aileron deflection

These manoeuvres were assessed over a range of airspeeds and altitudes to assess the complete envelope. The time histories of these manoeuvres were used to assess stability and the results compared across the different models.

Looking at the results of some of the dynamic behaviour we can compare the different methods against the wind tunnel data.

For Short Period characteristics, at low speed we can see that whilst all methods experience some damping, the AVL and DatCom methods diverge very quickly. However the AST and Simple Geometric approach both produce comparable damping characteristics.

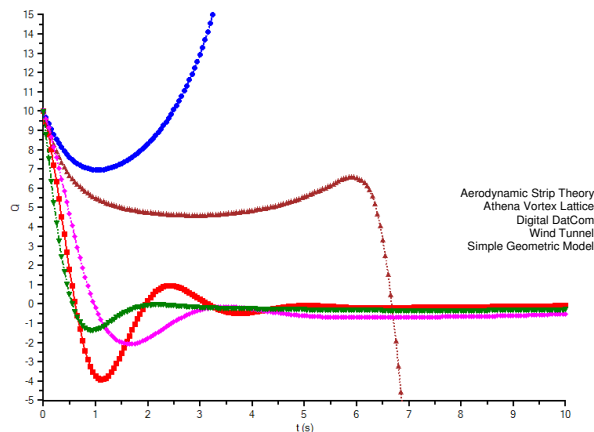


Figure 12 Pitch Rate Response for SPO

When looking at lateral behaviour, DatCom and Simple Geometric Models did not have sufficient data. The remaining methods showed lateral responses, but it can be seen that the AST has the best comparison to wind tunnel data. To keep the AST model simple, the dorsal strake was not added and only a constant fuselage drag.

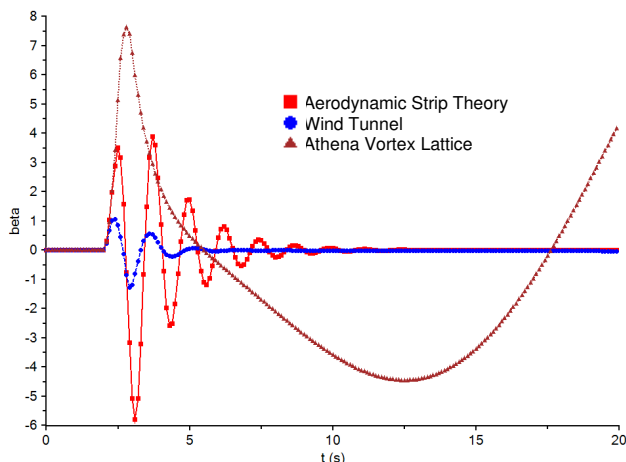


Figure 13 Sideslip Response to Dutch Roll

With Roll damping characteristics AST gives very good peak roll accelerations compared to the wind tunnel, and damping characteristics. The AVL value gives almost

double the Roll acceleration, and this is due in part to the fact that the differential aileron cannot be included in the AVL model, but can be added in the AST.

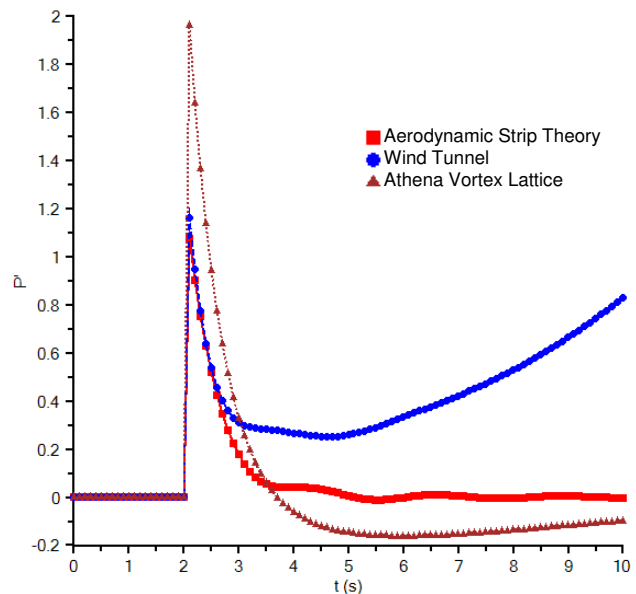


Figure 14 Roll Acceleration Response to Aileron Input

The final stage in the analysis was to investigate the ability to use a range of models and aerodynamic data generation techniques to see what can be used to assess the FAR 23 criteria.

The final FAR manoeuvre is to look at the time to bank, and so investigates further the roll authority defined within each of the models. In doing this, a further set of trim conditions were generated that trimmed the aircraft in a 30° Banked turn with One Engine Inoperative. A Response Model was then generated where the pilot moved to full roll stick after 2 seconds and maintained that until -30° Bank was achieved. The time to bank was compared to the regulations and to the Wind Tunnel Results.

SUMMARY OF METHODS & RECOMMENDATIONS

The first aspect when assessing the results is to look at the "costs" of the different aerodynamic calculation methods. These results can be found in Table 3. As can be expected, Wind Tunnel modelling is viewed as the most expensive method of obtaining aerodynamic data with the Aerodynamic Strip Theory method is viewed as the cheapest. This is due to the relatively simple set of data that is required for AST to enable dynamic responses to take place. In addition to this, the flexibility of both AST and Simple Geometric models, which can both be

parameterised so any changes made are automatically incorporated, makes them very powerful.

Methods such as the Digital Datcom and Athena Vortex Lattice Method which are still in use today are two good examples of aerodynamic generation techniques. However, the lengthy process needed to create and process data into a workable form can and will take more time than it would to create a workable aircraft model using other methods available. These methods can be used at later stages in design, but need a lot of work for every design change.

The Aerodynamic Strip theory is, during the preliminary stages of design, the most time efficient and flexible method, especially when using the J2 Elements plug-in. When using, well established aerofoil packages like JavaFoil, this paper has shown that aircraft aerodynamics can be calculated extremely quickly and with a good level

of accuracy. This allows for the simulation of an aircraft with very little information and still maintains simulation accuracy.

This can be seen when comparing with the wind tunnel base line, which is of course widely regarded as a simulation technique with high levels of accuracy. However, unlike wind tunnel testing with the financial and time costs of running a simulation being expensive, elements will allow for rapid design alterations at a minimum cost. This will even lead to an increased efficiency in the use of wind tunnel testing, as a good understanding of the aircraft stability and handling will already be known and will reduce the number of wind tunnel model iterations needed before a design freeze can be implemented for manufacture and flight testing.

	Ease of Data Generation	Costs	Technical Expertise Required	Flexibility of Data to Changes	Quality of Data	Total Score	Ranking
	1=Easy 5=Hard	1=Low 5 = High	1=Low 5=High	1=Flexible 5=Inflexible	1=High 5=Low		1=Best 5=Worst
Simple Geometric Modelling	1	1	1	2	5	10	2
Aerodynamic Strip Theory	1	1	1	1	3	6	1
Digital Datcom	4	2	3	4	4	17	3
Vortex Lattice	3	2	3	4	4	15	4
Wind Tunnel	5	5	5	5	1	21	5

Table 3 Costs of Aerodynamic Data Methods

REFERENCES

- [1] Clifford Matthews, **AERONAUTICAL ENGINEERS DATA BOOK**, Butterworth-Heinemann Ltd, 2001
- [2] Martin Hepperle, **JAVAFOIL – ANALYSIS OF AIRFOILS**, <http://www.mh-aerotoools.de/airfoils/javafoil.htm>
- [3] Mark Drela, **XFOIL – SUBSONIC AIRFOIL DEVELOPMENT SYSTEM**, <http://web.mit.edu/drela/Public/web/xfoil>
- [4] Jan Roskam, **AIRCRAFT DYNAMICS, AIRPLANE FLIGHT DYNAMICS AND AUTOMATIC FLIGHT CONTROL SYSTEMS**, Roskam Aviation and Engineering Corporation, 1982
- [5] J B Russell, **PERFORMANCE & STABILITY OF AIRCRAFT**, Edward Arnold, 1996
- [6] Ronald L. Panton, **INCOMPRESSIBLE FLOW**, Wiley-Interscience, 1996
- [7] Joseph Katz and Allen Plotkin, **LOW-SPEED AERODYNAMICS: FROM WING THEORY TO PANEL METHODS**, McGraw-Hill Companies 1991
- [8] JC Gibson, **THE DEFINITION, UNDERSTANDING AND DESIGN OF AIRCRAFT HANDLING QUALITIES**, Rep. No. LUT LR-756, Delft University of Technology, 1995

APPENDIX

AERODYNAMIC DATA PLOTS

Wing with Aileron

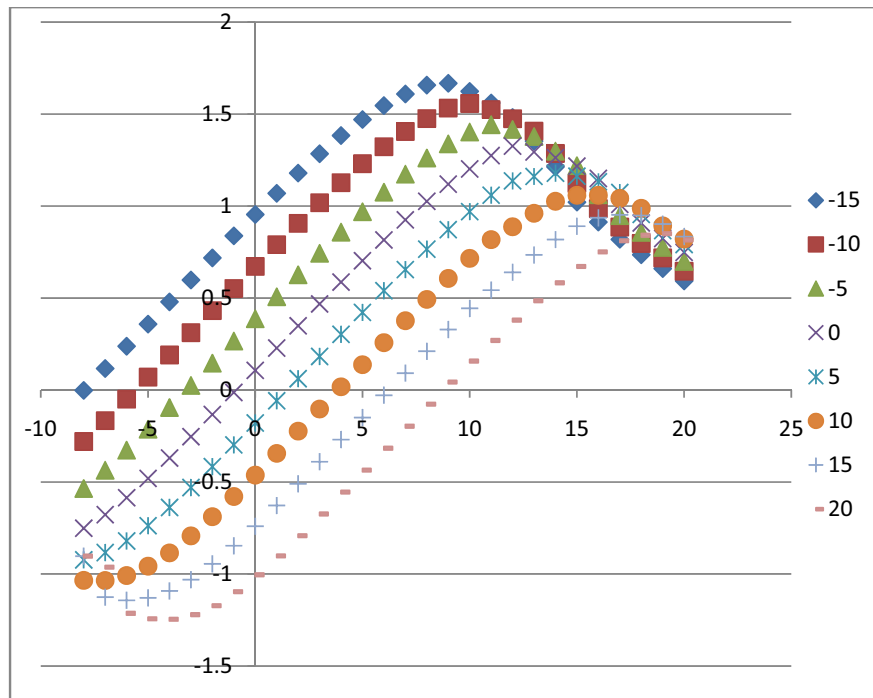


Figure O Lift Characteristics of the Wing Airfoil with Aileron Deflections

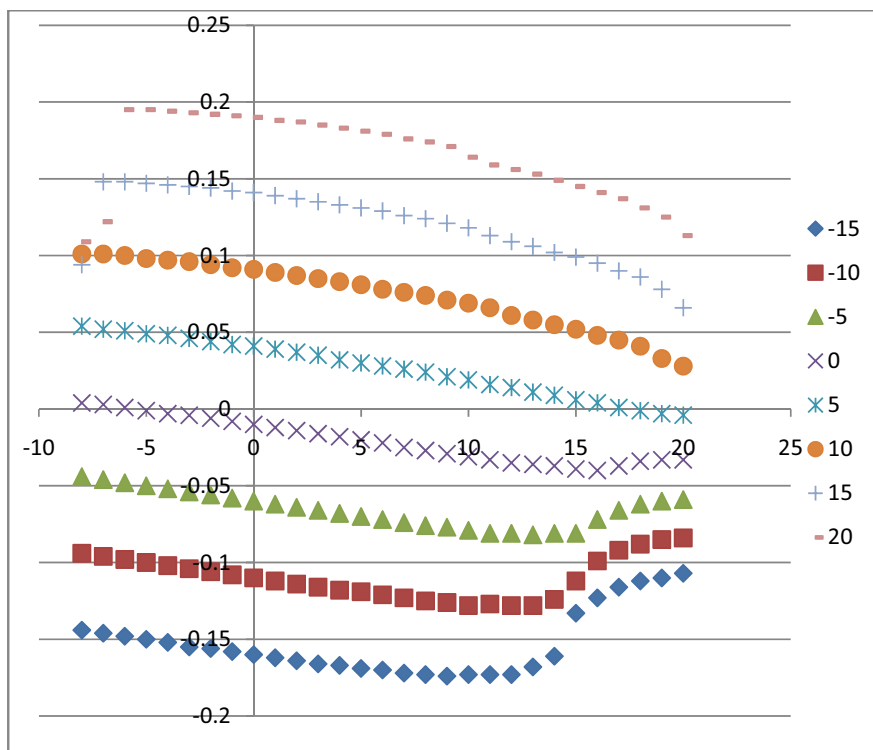


Figure P Pitch Characteristics of the Wing Airfoil with Aileron Deflections

Horizontal Tail with Elevator

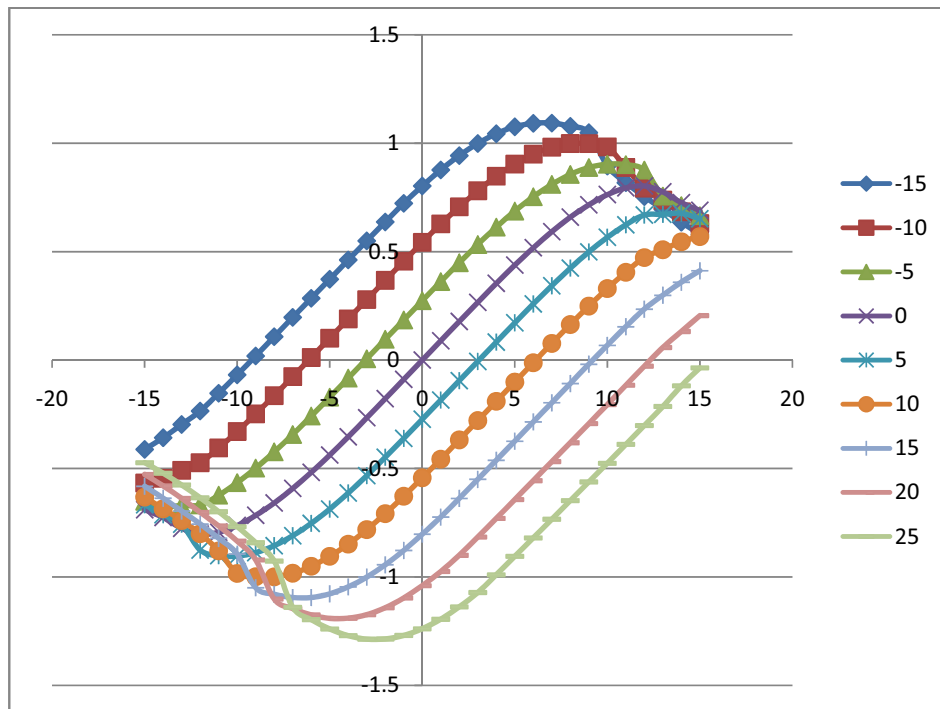


Figure Q Lift Characteristics of Horizontal Tail Airfoil with Elevator Deflections

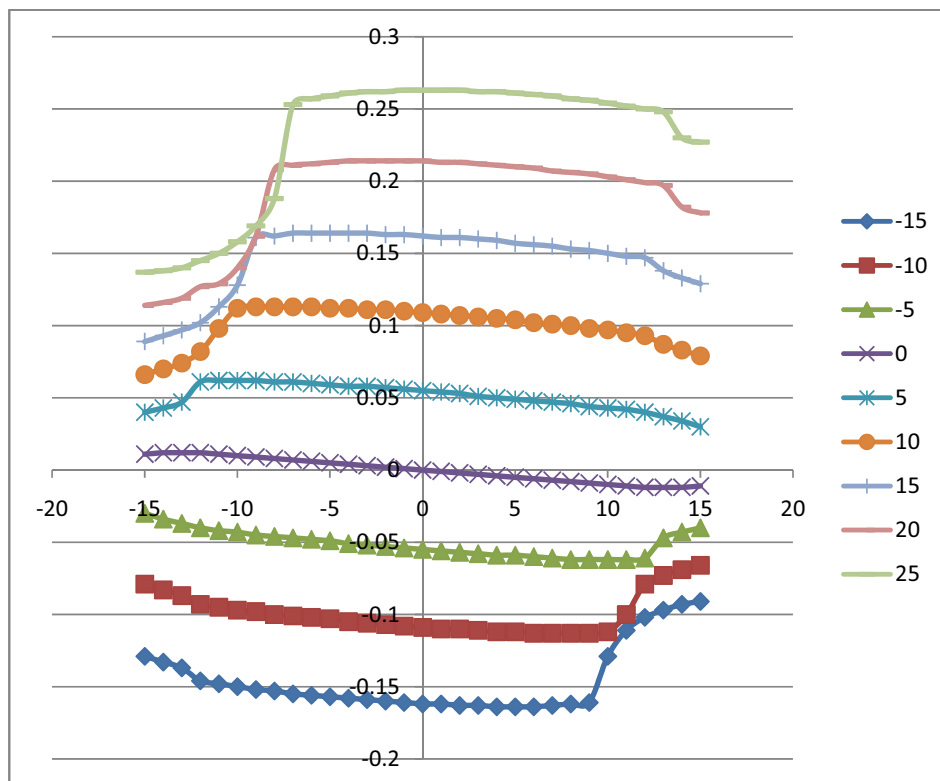


Figure R Pitch Characteristics of Horizontal Tail Airfoil with Elevator Deflections

Vertical Tail with Rudder

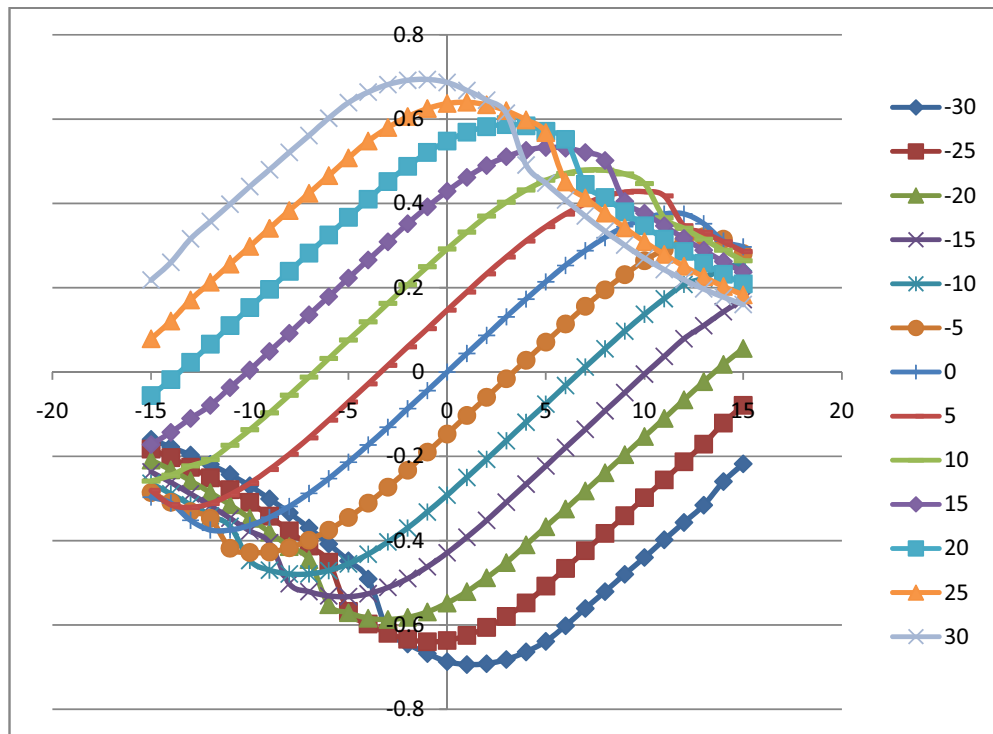


Figure S Lift Characteristics of the Vertical Tail Airfoil with Rudder Deflection

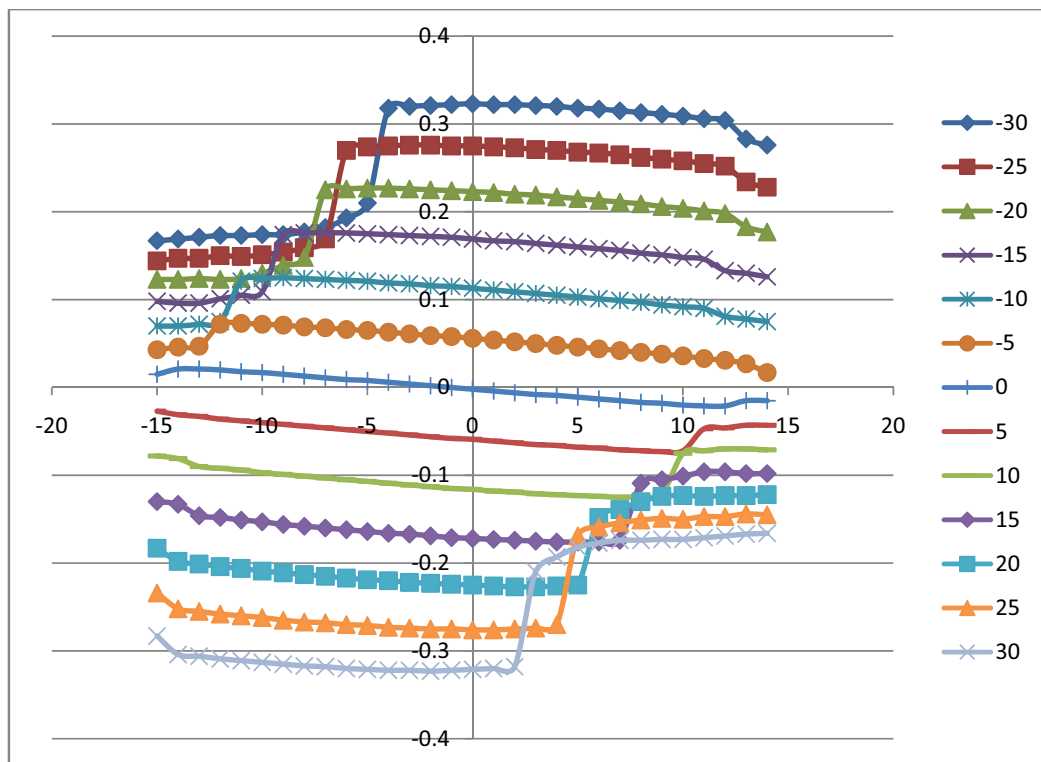


Figure T Pitch Characteristics of the Vertical Tail Airfoil with Rudder Deflection

DIGITAL DATCOM

*****INPUT DATA CARDS*****

NAMELIST

DIM M

PART

DERIV DEG

DUMP ALL

DAMP

\$FLTCON WT=1722.3, LOOP=1.0,
NMACH=4.0, MACH(1)=0.1512, 0.1726, 0.4538, 0.5178,
NALT=4.0, ALT(1)=91.44, 10363.0, 91.44, 10363.0,
NALPHA= 20.0,
ALSCHD(1)= -6.0, -4.0, -2.0, 0.0, 2.0, 4.0, 6.0, 8.0,
10.0, 12.0, 14.0, 16.0, 18.0, 20.0, 22.0, 24.0, 26.0, 28.0,
30.0, 32.0,
STMACH= 0.9, TSMACH= 1.2, TR= 0.0\$
\$OPTINS SREF=15.7, CBARR=1.3409, BLREF=11.99, ROUGFC=0.25E-3\$
\$SYNTHS XCG=5.1336, ZCG=1.1348,
XW=4.92, ZW=0.28, ALIW=2.5,
XH=9.28, ZH=2.28, ALIH=0.0,
XV=8.52, ZV=1.79,
SCALE=1.0, VERTUP=.TRUE.\$

\$BODY NX=20.0,
X(1)= 0.0, 0.583, 1.166, 1.749, 2.332, 2.915, 3.498,
4.081, 4.664, 5.247, 5.83, 6.413, 6.996, 7.579, 8.162,
8.745, 9.328, 9.911, 10.494, 11.077,
R(1)= 0.0, 0.483, 0.664, 0.745, 0.765, 0.765, 0.765,
0.765, 0.765, 0.765, 0.765, 0.745, 0.704, 0.644,
0.543, 0.422, 0.281, 0.100, 0.0,
ZU(1)= 0.442, 0.805, 0.966, 1.248, 1.651, 1.731, 1.731,
1.731, 1.731, 1.731, 1.771, 1.771, 1.731, 1.731, 1.731,
1.731, 1.731, 1.731, 1.731, 1.530,
ZL(1)= 0.442, 0.201, 0.161, 0.120, 0.120, 0.120, 0.080,
0.040, 0.0, 0.0, 0.0, 0.120, 0.241, 0.402, 0.523, 0.684,
0.845, 1.006, 1.248, 1.530,
ITYPE= 2.0, METHOD= 1.0\$
\$WGPLNF CHRDR=1.642, CHRDTP=0.985,
SSPN=5.94, SSPNE=5.2,
SAVSI=0.126,
CHSTAT=0.25, TWISTA=-3.0,
DHDADI=4.0,
TYPE=1.0\$

\$WGSCHR TYPEIN= 1.0, NPTS= 50.0,
XCORD(1)= 0.0,0.0002081,0.0004404,0.0009152,0.001396,
0.0023657,0.0048113,0.0097387,0.0295765,0.0395197,0.04947,
0.0594256,0.0793488,0.0893149,0.0992833,0.1242134,
0.1491534,0.1990572,0.2240189,0.2489862,0.2739588,0.2989362,
0.3239183,0.348905,0.3738961,0.4238911,0.4488952,0.4989173,
0.5239361,0.5489615,0.573994,0.5990348,0.6240846,0.6792255,
0.6992846,0.724361,0.7494381,0.7745122,0.7795268,0.8196388,
0.8246519,0.8497156,0.8747744,0.8998261,0.9248728,0.9499137,
0.9749509,0.9799575,0.9899698,1.0,
YUPPER(1)= 0.0,0.0030156,0.0042896,0.0061014,0.0074893,
0.0096695,0.0135796,0.0188041,0.0304662,0.0345513,0.0381238,
0.0413146,0.0468303,0.0492664,0.051529,0.056548,0.0608503,
0.0677428,0.0704807,0.0728205,0.0747781,0.0763869,0.0776567,
0.0785958,0.0792244,0.0795538,0.0792429,0.0776188,0.0762416,
0.0744017,0.0720512,0.0690951,0.0654873,0.0553066,0.0510353,
0.045524,0.0399538,0.0345993,0.0335463,0.0254604,0.0245092,

```

0.0199073,0.0156566,0.0119208,0.0085371,0.0055808,0.0028859,
0.002405,0.0015154,0.0006699,
YLOWER(1)= 0.0,-0.0015952,-0.0022948,-0.0033303,-0.0041686,
-0.0055288,-0.0078869,-0.01106,-0.0183788,-0.0209903,
-0.0232064,-0.025182,-0.0286494,-0.0302069,-0.0316799,
-0.0349892,-0.0378769,-0.0425688,-0.0445036,-0.046231,
-0.047753,-0.0490918,-0.0502523,-0.0512314,-0.0520125,
-0.0530064,-0.0532088,-0.0530259,-0.0527197,-0.052153,
-0.0513265,-0.0502214,-0.0487928,-0.0438963,-0.0411985,
-0.0362719,-0.0306382,-0.0259808,-0.0251533,-0.0191028,
-0.0184089,-0.015141,-0.0121685,-0.0093751,-0.0067255,
-0.004323,-0.0022855,-0.0019329,-0.0012757,-0.0006699$
NACA H 4 0010
$HTPLNF CHRDR=1.097, CHRDTF=0.658,
      SSPN=2.6685, SSPNE=2.6685,
      SAVSI=3.302,
      CHSTAT=0.25, TWISTA=0.0,
      DHDADI=0.0,
      TYPE=1.0$
$VTPLNF CHRDR=2.212, CHRDTF=0.567, CHSTAT=0.25, DHDADO=0.0,
      SAVSI=36.658, SSPN=2.2, SSPNE=2.2, TYPE=1.0$
$SYMFLP FTYPE=1.0,
      NDELTA=9.0, DELTA(1)=-15.0,-10.0,-5.0,0.0,5.0,10.0,15.0,20.0, 25.0,
      PHETE=0.0522, PHETEP=0.0523,
      CHRDFI=0.431, CHRDFO=0.263,
      SPANFI=0.0, SPANFO=2.6685,
      CB=0.84, TC=0.3, NTYPE=1.0$
NACA V 6 64A-012
CASEID TOTAL: VLJ Aircraft

```

ATHENA VORTEX LATTICE AIRCRAFT MODEL FILE

```

Vlj
0.5179
0      0      0.0
15.7  1.3409  11.99
4.92   0.0    0.76
#=====
#=====
BODY
Fuse
20  1.0
#
SCALE
1.0  1.0  1.0
TRANSLATE
0.0  0.0  0.0
#
BFIL

fuse.dat
#=====
SURFACE
Wing
10  1.0
#
YDUPLICATE
0.00000
#
SCALE
1.0  1.0  1.0
#
TRANSLATE
4.92  0.00000  0.28
#
# twist angle bias for whole surface
ANGLE
0.00000
#
#-----
# Xle      Yle      Zle      chord      angle  Nspan  Sspace
SECTION
0.0      0.74      0.0      1.642      2.5  10      1

AFIL
mainwing.dat
#-----
# Xle      Yle      Zle      chord      angle  Nspan  Sspace
SECTION
0.1009086  4.14      0.289      1.212      0.54  10      1

AFIL
mainwing.dat

CONTROL
aileron  1.0  0.85  0.0  1.0  0.0  -1.0
#-----
# Xle      Yle      Zle      chord      angle  Nspan  Sspace
SECTION

```

```

0.1495      5.78      0.404      1.005      -0.41    10      1

AFIL
mainwing.dat

CONTROL
aileron  1.0    0.85    0.0 1.0 0.0    -1.0

#-----

#      Xle      Yle      Zle      chord      angle    Nspan  Sspace
SECTION
0.1543      5.94      0.415      0.985      -0.5     2      1
AFIL
mainwing.dat
#=====
#
SURFACE
Stab
8  1.0  !
YDUPLICATE
0.00000
ANGLE
0.00000
#
TRANSLATE
9.28      0.00000      2.28
SCALE
1.000  1.0  1.0
#-----
#      Xle      Yle      Zle      chord      angle    Nspan  Sspace
SECTION
0          0.0      0.0      1.097      0.000    5      -1.50

AFIL
NACA0010.dat

CONTROL
elevator 1.0    0.60    0.0 1.0 0.0      1.0

#-----

SECTION
0.263906      2.6685      0.0      0.658      0.000    0      0

AFIL
NACA0010.dat

CONTROL
elevator 1.0    0.60    0.0 1.0 0.0      1.0

#
#=====
#
SURFACE
Tail
8  1.0  14  0.75
#
TRANSLATE
8.52      0.00000      1.79
#-----

```

#	Xle	Yle	Zle	chord	angle	Nspan	Sspace
SECTION							
0.0		0.0	0.0	2.212	0.000	4	1.50

AFIL
NACA64A.dat

CONTROL
rudder 1.0 0.70 0.0 0.0 1.0 1.0

#-----

SECTION							
1.7		0.0	2.35	0.567	0.000	6	-1.50

AFIL
NACA64A.dat

CONTROL
rudder 1.0 0.70 0.0 0.0 1.0 1.50

Destructive Evaluation of Four Aircraft Representative of the General Aging Aircraft Fleet

Melinda Laubach

National Institute for Aviation Research
Wichita State University

Copyright © 2008 SAE International

ABSTRACT

Due to current economic conditions, aircraft companies of today are experiencing an increasing need for their fleets to maintain safe operation beyond their original design life. The result is a growing percentage of aging aircraft that must maintain their airworthiness by utilizing standard methods of inspection and repair.

In order to determine if potential continuing airworthiness problems exist for the general aviation fleet as a function of the aging process, the Federal Aviation Administration (FAA) established a research program at the National Institute for Aviation Research (NIAR), Wichita State University, to conduct destructive evaluations on four aged general aviation airplanes. The intent of the program is to provide insight into the condition of a typical aged airplane and to see if a correlation exists between its maintenance history and current condition from a safety of flight perspective.

Four airplanes used in commuter service – a 1969 Cessna 402A, a 1979 Cessna 402C, a 1975 Piper Navajo Chieftain, and a 1993 Beechcraft 1900D Airliner – have been evaluated in the research program. In order to achieve the level of inquiry desired, the aircraft were assessed during two phases of research.

The “Inspection Phase” included a survey of the maintenance records; non-intrusive visual inspection of the aircraft structure, systems, and wiring; and supplemental nondestructive inspections (NDIs) of critical structural areas.

The “Teardown Phase” consisted of detailed structural disassembly; intrusive visual inspection of aircraft wiring and system components; close visual inspection of internal structure; microscopic examinations of suspect and critical structural areas; and fractographic analysis of defects identified during the “Teardown Phase.” Results of the teardown evaluations illustrate the typical condition of an aged general aviation aircraft and show how

differences in operational environment impact the aircraft’s airworthiness. The teardown evaluations involved a comprehensive look at each airplane as a whole, including airframe, wiring, and systems.

The results of the overall research program point to some recommended actions that may improve the continued airworthiness of aging general aviation airplanes. Some recommendations are specific to an airplane model, while others pertain to the teardown evaluation research program itself and its use in future investigations into the aged general aviation airplane fleet. Detailed results of the program are documented in technical reports for the FAA, the Cessna Aircraft Company, New Piper Aircraft Company, and Hawker Beechcraft Corporation. These results provide valuable information to guide the continued safe operation of aging aircraft and give new insight into the field of aging aircraft research.

INTRODUCTION

Economic and market conditions of present day aviation companies are requiring the use of airplanes far beyond their original design life objectives. This aging airplane concern exists for all types of airplanes including commercial, military, and general aviation. The concern is being amplified as more companies use aged aircraft and rely on standard inspection practices for an assurance of airworthiness. Standard practices to ensure continuing airworthiness include scheduled inspection and maintenance tasks contained in maintenance manuals, Instructions for Continued Airworthiness (ICA), and service bulletins. These practices are not just limited to structural integrity but also include wiring and systems integrity as well. These initiatives have provided timely preventive maintenance recommendations that permit the continued safe operation of aging airplanes until retirement from service for economic reasons.

Although the general public is primarily concerned with the airworthiness of commercial airplanes where most research funding resources and efforts have been focused, a growing concern also exists in the small airplane fleet. While numerous investigations are being conducted on the aging aspects of large transport and military airplanes, limited resources are being expended to understand the aging aspects of the small airplane fleet. The investigations on large transport and military airplanes have focused on the structural integrity as well as wiring and systems related aging problems. The results of these investigations can benefit a similar research program that investigates small airplanes.

Small airplanes are generally classified as general aviation airplanes. When one mentions general aviation, the traditional image usually involves a four-passenger airplane like a Cessna 172. However, general aviation covers a wide range of airplanes. In the context of this program, a general aviation airplane will be defined as a 14 CFR Part 23 (or predecessor CAR 3) airplane, which includes normal, utility, acrobatic, and commuter category airplanes. This classification includes airplanes operating commuter, cargo, or taxi service under 14 CFR Part 135.

The general aviation fleet includes approximately 210,000 fixed wing airplanes with about 71% being single-engine piston aircraft, 10% being multi-engine piston aircraft, 10% being experimental aircraft, and 9% being classified as turboprop, jet, glider, or lighter-than-air. Usage of the general aviation fleet is categorized as follows: 60% for personal use, 21% for business use, 6% for instruction, 4% for aerial application/observation, 3% for commuter service, 2% for public use, and 4% for other usage.

PROGRAM OBJECTIVES

Due to the large number of general aviation airplanes and their wide usage, the aging aspects of these aircraft must be addressed. In September 2002, the Small Airplane Directorate and William J. Hughes Technical Center of the FAA established a research program in order to begin addressing the aging concerns regarding general aviation airplanes. The research program focused on conducting destructive teardown evaluations on aged airplanes used in commuter service to determine if potential continuing airworthiness problems exist for the general aviation fleet as a function of the aging process. The intent of this program is to provide insight into the condition of a typical aged small airplane and to determine if a correlation exists between its maintenance history and the apparent condition of the airplane from a safety of flight perspective. The research program is primarily conducted by the Aging Aircraft Research Laboratory at NIAR, Wichita State University.

Much of the current concern related to aging effects on aircraft focuses on calendar age as opposed to flight hours. For instance, deterioration of wiring, aging effects on aircraft systems (control systems, seals, cables, etc.), and corrosion are calendar related, and these effects

may possibly be a continued safety of flight concern. For example, major attachments, such as wing-to-fuselage attachments, are typically never removed and inspected. These areas of concerns are the areas that the research program primarily focuses on to begin addressing and providing insight into the aging general aviation fleet.

The short term objective of the research program is to determine if potential continuing airworthiness problems exist for the general aviation fleet as a function of the aging process. The long term objective is to establish guidance to ensure current maintenance programs of small general aviation airplanes are providing acceptable levels of continued airworthiness. Achievement of the short term objective should determine if generic degradation indicators exist in the small airplane fleet. These indicators will likely include structural (corrosion/cracking); electrical systems or wiring; aircraft systems, such as fuel, hydraulic, pneumatic, mechanical, and flight control; and maintenance, service, and inspection quality. Determination of generic degradation indicators will assist in providing initial generic inspection guidance, such as:

- Do maintenance inspection programs address all areas of concern appropriately?
- What was found in areas that normal maintenance would not see?
- Are additional inspection criteria required for aged airplanes?
- Should specialized one-time inspections at some age be required?
- Should inspections and maintenance programs become more extensive as the airplane ages?

PROGRAM TEAM MEMBERS

NIAR and the FAA developed an advisory committee, consisting of FAA personnel, NIAR personnel, original equipment manufacturers (OEM) of general aviation aircraft, teardown experts, NDI experts, and a commuter aircraft operator to review the technical approach, findings, and conclusions of the teardown examination program. Figure 1 provides an overview of the key team members from each organization involved. This advisory committee was important to the success of the teardown examination program.



Figure 1 – Program Team Members

MAIN SECTION

AIRCRAFT SELECTION

The first teardown examination was performed on a 1969 Cessna 402A model (tail number N812BW), which had accumulated 19,698.9 airframe hours. The aircraft, shown in Figure 2, was most recently operated in commuter service by Sunshine Airlines for tourism flights from Las Vegas to the Grand Canyon. The second airplane selected was a 1979 Cessna 402C model (tail number N780EA), which is shown in Figure 3. This aircraft had 25,546.6 total airframe hours and was most recently used by Cape Air/Nantucket Airlines of Hyannis, Massachusetts, flying in and around Cape Cod, the Florida Keys, and the Caribbean Islands. The third aircraft selected for destructive evaluation was a 1975 Piper Navajo Chieftain PA31-350 model (tail number N61510), which is pictured in Figure 4 and had accumulated 17,269.9 total airframe hours. The Piper aircraft was operated all across the United States with the majority of the time spent on the West Coast and Alaska. The final aircraft selected for teardown examination was a 1993 Beechcraft 1900D Airliner (tail

number N856CA), which is shown in Figure 5. It had accumulated 15,203 total airframe hours and was operated on the East Coast. Logbooks, the Airworthiness Directive compliance list, and FAA 337 forms were included in the purchase of each airplane. Table 1 provides more detailed information regarding each aircraft.

The two Cessna 402s and Piper Navajo models were selected because they represent a large portion of the small airplane commuter fleet. These aircraft models share many design commonalities with other small twin class airplanes. The design concepts of the structures (layout and materials) are similar model-to-model and manufacturer-to-manufacturer. Therefore, findings from the destructive evaluation of these aircraft models would be applicable to all general aviation models regardless of manufacturer. The Beechcraft 1900D was selected since it was a larger commuter aircraft with a pressurized cabin.

Table 1 – Aircraft Specifications

Aircraft Model	Cessna 402A	Cessna 402C	Piper Navajo Chieftain PA31-350	Beechcraft 1900D
Aircraft Type	Turbo-Charged	Turbo-Charged	Turbo-Charged	Turbo Prop
Airframe Hours	19,698.9	25,546.6	17,269.9	15,203
Airframe Cycles	--	--	--	23,260
Number of Engines	2	2	2	2
Passenger Capacity	6-8	6-8	8	19
Range (Miles)	416	416	833	860
Length (ft)	36.5	36.5	34	57.9
Height (ft)	11.6	11.4	13	14.9
Wing Span (ft)	39	44	40.8	54.5
Take Off Weight (lbs)	6,500	6,500	7,000	15,245
Cabin Type	Not Pressurized	Not Pressurized	Not Pressurized	Pressurized
Operating Environment	Grand Canyon Tour Service (Fatigue)	Cape Air East Coast/ Island Commuter Service (Corrosive)	West Coast Alaska Commuter (Mixed Environment)	East Coast Commuter (Corrosive)



Figure 2 – 1969 Cessna 402A, Tail #N812BW



Figure 4 – 1975 Piper Navajo Chieftain, Tail #N61510



Figure 3 – 1979 Cessna 402C, Tail #N780EA



Figure 5 – 1993 Beechcraft 1900D, Tail #N856CA

OVERVIEW OF THE “INSPECTION PHASE” AND THE “TEARDOWN EXAMINATION PHASE”

The evaluation of each of the aircraft is divided into two main phases, the inspection phase and the teardown examination phase. In the inspection phase, three sub-tasks were performed: a survey of the aircraft maintenance records, routine visual inspection of the airframe and wiring, and structural supplemental inspections. A wiring assessment was not performed on the Beechcraft 1900D as most of the wiring and systems components were removed from the aircraft prior to its arrival at NIAR. The teardown examination phase involved the disassembly of the airframe, close visual inspection of the internal structure, post disassembly NDIs, microscopic examination of suspect and critical structural areas, and fractographic analysis of selected cracks found during the examination to determine extent and failure mode (i.e., fatigue, overload, stress corrosion, etc.).

A survey of the aircraft maintenance records was conducted in an effort to correlate aircraft condition with aircraft usage and maintenance history. Maintenance logbooks, service bulletins, airworthiness directives, and FAA 337 forms for major repairs and modifications were reviewed in this survey. The general condition of each aircraft was evaluated through the routine visual inspections of the airframe and wiring. Airframe visual inspections were performed according to the appropriate maintenance manual, which were supplied by the OEMs. These visual inspections are typically required on an annual basis. The wiring assessment was performed on the Cessna 402A and 402C, and Piper Navajo by a basic visual inspection of all accessible wiring to check the condition of the wires for any defects or flaws, which could pose a risk to flight safety. Wiring finding types included chaffing, rubbing, burning, tearing of the insulation, etc. The intent of the wiring and airframe visual inspections was to identify and characterize the severity and location of each finding prior to the teardown examination phase of the program. This approach would allow the NIAR team to determine which findings could be identified in the field. The results of the routine visual inspections were reviewed by a licensed airframe mechanic to determine which findings were “noteworthy.” Noteworthy findings are findings that would require further maintenance action. Supplemental inspections were performed on the Cessna 402A and 402C aircraft according to Cessna’s Supplemental Inspection Document (SID). Supplemental inspections are comprised of NDI techniques that target principal structural elements. The Piper Navajo did not have an OEM developed SID. NIAR engineers used engineering judgment, structural similarity to the Cessna aircraft, and maintenance history to develop a set of supplemental inspections that were performed on the Piper Navajo. To address structural concerns on the Beechcraft 1900D, the Hawker Beechcraft Structural Inspection Manual (SIM) was used during the routine visual inspections.

At the conclusion of the inspection phase, the teardown examination phase was performed in order to find defects not found by typical maintenance or supplemental inspections. Every finding or NDI indication identified during the inspection phase and teardown examination phase was further investigated during the microscopic examination portion of the teardown examination phase. The Cessna 402A, 402C, and Piper Navajo were disassembled into the major airframe sections including: wings, forward fuselage, aft fuselage, cabin, landing gear, horizontal stabilizer, and vertical stabilizer. All wiring and systems components were removed at the beginning of the teardown examination phase to facilitate easier detailed disassembly. At the time of acquisition, the Beechcraft 1900D was already disassembled into major sections and the systems components and wiring had already been removed. As detail parts were being removed from the airframe, close visual inspections were performed to document existing damage prior to coatings removal. All paint and primer were removed using mechanical techniques (solid media blasting). Fluorescent liquid penetrant (FLP) inspections were performed on all details after disassembly to identify structural defects. These details were then examined microscopically to identify and characterize any defects in the structure. Certain cracks were sectioned and mounted for fractographic analysis to determine the extent of the flaw and the mode of failure (i.e., fatigue, stress corrosion cracking, etc.).

SUMMARIES AND EXAMPLES OF TYPICAL ROUTINE INSPECTION FINDINGS

The routine visual airframe inspections were performed with the intent of finding and documenting every detectable flaw on the aircraft. This inspection approach led to a large number of documented flaws. The results of these inspections on each aircraft were reviewed by licensed airframe mechanics to determine which findings were noteworthy. Most of these flaws were deemed minor, meaning no further maintenance action was required. In general, the combined routine visual inspection results of the four evaluated aircraft revealed similar findings typically addressed in the maintenance field. With an aged aircraft, flaws such as minor cracks, scratches, dents, paint chips, and slight corrosion were to be expected and posed no immediate threat to the safety of the aircraft. In addition to dents and areas of missing paint, slight to moderate corrosion was noted on a majority of the screws on the assessed aircraft. It was also noted that some of the disposable parts on these aircraft such as seals and hoses were due to be replaced. Figure 6 and Figure 7 show examples of typical minor defects observed during the routine visual inspections.



Figure 6 – Paint Bubble in Skin that Cracked Paint



Figure 7 – Corroded Screw Heads

about the same frequency. It is important to note that since corrosion and rusted screws are related, at times these categories may have been used interchangeably. The breakdown of findings locations for the Cessna 402A and 402C are provided in Figure 10 and Figure 11.

These charts also illustrate similarities in the frequency of findings in major airframe components. These trends in both the finding type and finding location were observed on the Piper Navajo and Beechcraft 1900D as well.

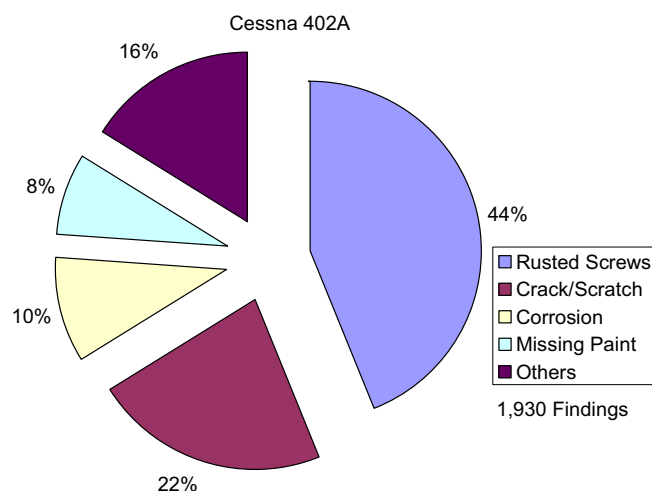


Figure 8 – Findings by Type, Cessna 402A

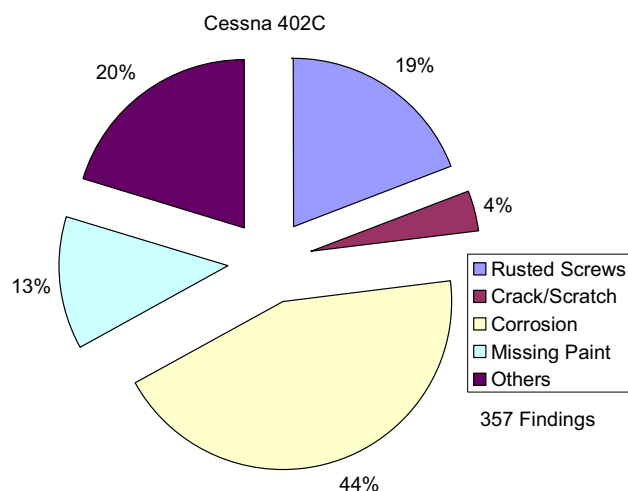


Figure 9 – Findings by Type, Cessna 402C

During the routine visual inspections on the Cessna 402A, 1,930 defects were identified with only four deemed noteworthy by a licensed airframe mechanic. This number of defects is much higher than any other aircraft; however, the inspections on the Cessna 402A were performed by different individuals than the Cessna 402C. Many more extremely minor defects were reported on the Cessna 402A than on any other aircraft; therefore, one should not conclude that the Cessna 402A was in worse condition than the other aircraft simply due to the high number of routine visual inspection findings. On the Cessna 402C, 357 findings were observed during the routine visual inspections with only three findings determined to be noteworthy. Only eleven noteworthy defects were identified on the Piper Navajo, while forty-nine of the 194 routine visual inspection findings on the Beechcraft 1900D were classified as noteworthy.

Figure 8 and Figure 9 show the frequency of different categories of findings for the Cessna 402A and 402C, respectively. These charts illustrate that the same finding types, in general, occurred on both aircraft with

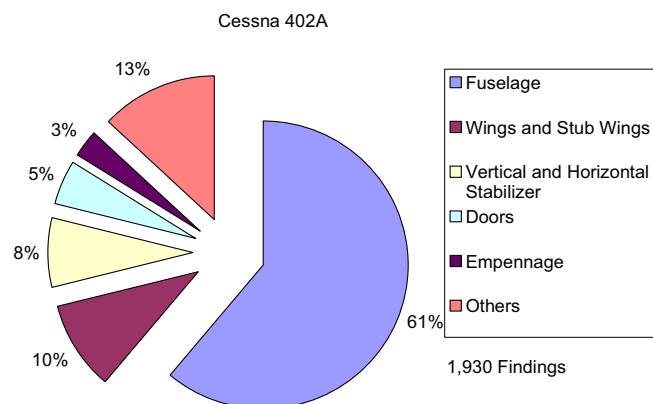


Figure 10 – Findings by Location, Cessna 402A



Figure 12 – Example of Cutting through Outer Insulation

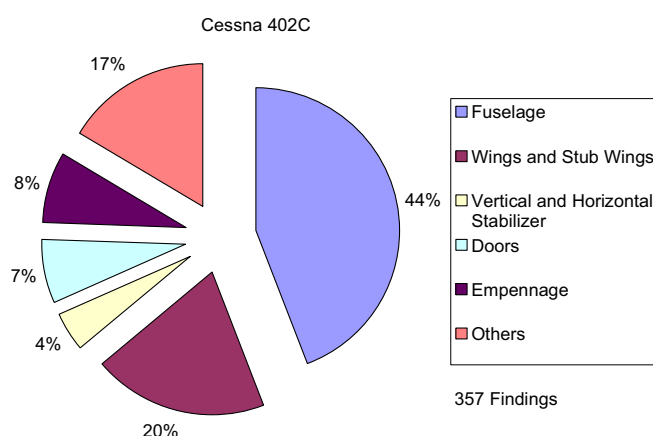


Figure 11 – Findings by Location, Cessna 402C



Figure 13 – Example of Exposed Inner Conductor



Figure 14 – Example of Heat Damage

The wiring routine visual inspections were performed without disturbing the condition of the wires in the aircraft. The objective of the inspection was to assess the general wiring condition, installation defects, terminations, and connectors. General wiring condition findings included: rubbing/chafing of the outer insulation, cutting through the outer insulation, exposed/damage shield, exposed/damage inner conductor, fluid/chemical/dust contamination, heat damage, and corroded shield/conductors. An example of cutting through outer insulation is provided in Figure 12, while an exposed inner conductor is pictured in Figure 13. An example of a heat damaged wire observed during the routine wiring visual inspections is shown in Figure 14. Wiring installation findings included: repaired wires, inadequate clearance to structure, wires riding on other wire bundles, missing/deteriorated grommets/ties, and improper bend radius. Figure 15 shows an example of inadequate clearance to surrounding structure, while Figure 16 provides an example of improper bend radii. A wiring assessment was not performed on the Beechcraft 1900D, as most of the wiring had been removed prior to the acquisition of the aircraft.



Figure 15 – Example of Inadequate Clearance to Structure

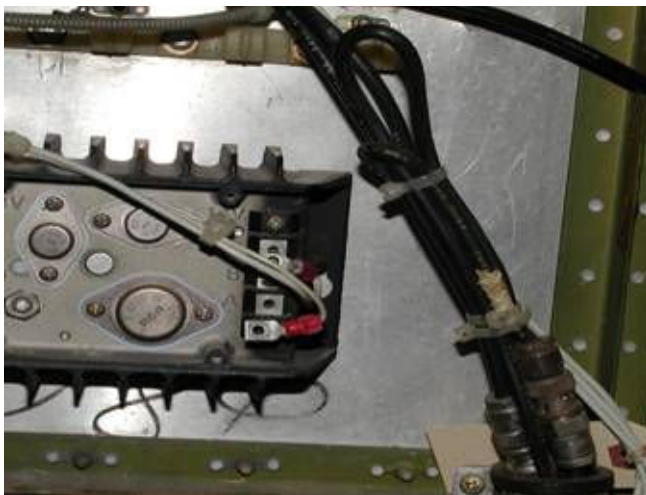


Figure 16 – Example of Improper Bend Radii

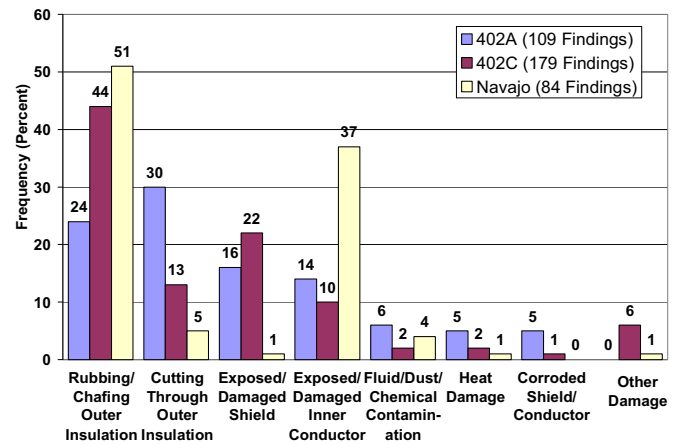


Figure 17 – Frequency of Wiring Condition Findings

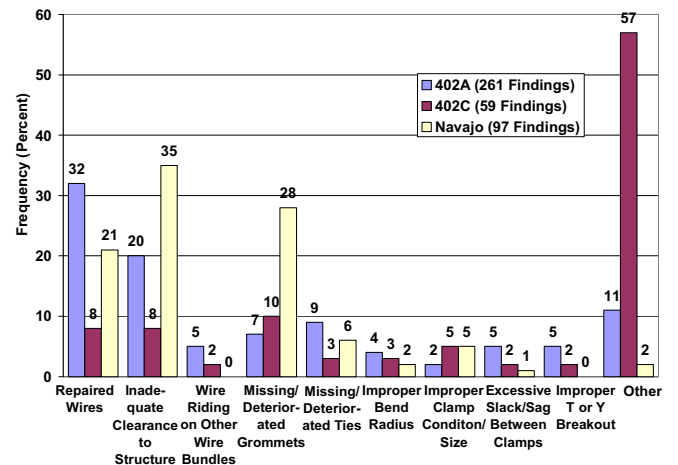


Figure 18 – Frequency of Installation Findings

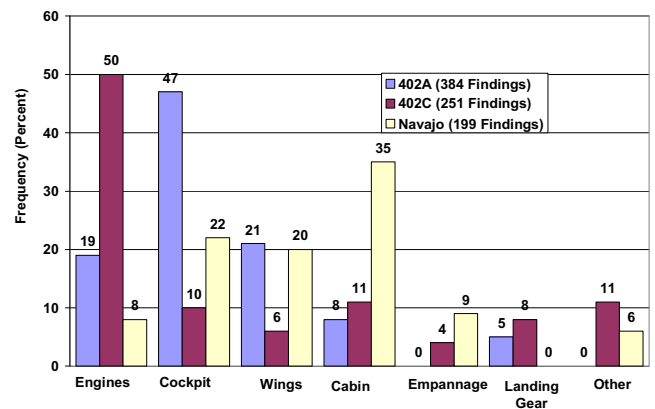


Figure 19 – Location of Wiring Findings

As with the routine visual airframe inspections, the wiring visual inspection results showed trends that were consistent from aircraft to aircraft. Figure 17 shows the frequency of wire condition findings on the Cessna 402a and the Piper Navajo. While the results vary some from aircraft to aircraft, the categories with the most findings are generally consistent. The frequency of installation findings on these three aircraft are compared in Figure 18, while Figure 19 provides a comparison of the location of all wiring defects on each aircraft.

SUPPLEMENTAL AIRFRAME INSPECTIONS

As general aviation aircraft approach and surpass design life objectives, in-service failures due to corrosion and fatigue become more probable. To address this issue for the Cessna 402A and 402C aircraft, the Cessna Aircraft Company developed a Supplemental Inspection Document (SID) for each aircraft. These supplemental inspections were developed to utilize NDI techniques for inspection of areas that were analytically predicted to have damage, that exhibited damage in a structural test, or that have had a history of maintenance issues. Stress, fatigue, and damage tolerance analyses were used to predict structural locations that were more likely to experience cracking. These analyses were supported with static and fatigue tests of structural components and full scale aircraft in order to determine areas where failure or cracks may occur. Three sources of information were used to determine the frequency of maintenance issues on Cessna 402s: survey of current Cessna 402 owners, service bulletins and service information letters, and FAA Service Difficulty Reports (SDRs).

For the Piper Navajo Chieftain model PA31-350, the NIAR team developed a supplemental inspection program based on engineering judgment and aircraft maintenance history since the Navajo does not have a SID. The supplemental inspections target principal structural elements (PSEs), which were identified for the Navajo based on its similarities to other models with supplemental inspections, specifically the Cessna 402 aircraft. Detailed NDI procedures were developed for each of the PSEs based on the type of defect to be found, access to the inspection area, and the type of material in the inspection area. It is important to note with a service history/engineering judgment approach, as was taken with the development of this set of supplemental inspections, no damage tolerance assessment was performed. Therefore, no initial inspection time or inspection intervals were established. This set of supplemental inspection provides an owner/operator with information on where to inspect a high-time Piper Navajo and provides inspection procedures developed for specific PSEs of this aircraft.

The supplemental inspections for the three aircraft employed several NDI techniques including visual inspection, eddy current surface and bolt hole inspection, liquid penetrant inspection, and magnetic particle inspection. These inspections were performed on specified areas of the wings, fuselage, horizontal stabilizer, vertical stabilizer, and landing gear. Each of the supplemental inspections were performed by certified airframe inspectors or NDI technicians. For the Cessna 402A aircraft, fifteen inspections were performed on each of the wings; two inspections were performed on the fuselage; eleven inspections were performed on the horizontal stabilizer and elevators; six inspections were performed on the vertical stabilizer and rudder; and nine

inspections were performed on the main/nose landing gear. For the Cessna 402C aircraft, fourteen inspections were performed on each of the wings; one inspection was performed on the fuselage; twelve inspections were performed on the horizontal stabilizer and elevator; six inspections were performed on the vertical stabilizer and rudder; and seven inspections were performed on the main/nose landing gear. The supplemental inspections for the Piper Navajo aircraft included twelve areas on each of the wings, five areas on the horizontal stabilizer and elevator, four areas on the vertical stabilizer and rudder, and two areas on the landing gear.

In order to address the fatigue and stress issues on the Beechcraft 1900D, the Hawker Beechcraft Structural Inspection Manual (SIM) was used during this phase. The Hawker Beechcraft SIM was developed by Hawker Beechcraft technical support personnel to guide and enable a thorough inspection of all airframe and components. The SIM was used as part of the visual inspection of the overall inspection phase. As previously reported in the visual inspections paragraph, there were 221 individual inspection tasks conducted on the structure of the fuselage, wings and empennage, with forty-nine discrepancies found and recorded.

TEARDOWN EXAMINATION

The teardown examination involved disassembly of the airframe and major aircraft sections, post disassembly NDI, and microscopic examination of all critical structural areas. The objective of the post disassembly FLP inspection was to conduct a broad area inspection of critical structural details, defects identified during detailed disassembly, and findings from the inspection phase to identify defects prior to the microscopic examination.

The microscopic examination provided verification and detailed quantification of all defects identified during the inspection phase, detailed disassembly, and post disassembly NDI. Microscopic examinations were conducted on suspect areas from all inspections, on defects found during disassembly, and on critical structural details to locate and characterize cracks and areas of corrosion. Microscopic characterization of cracks included the length of the crack and crack orientation. Fractographic analysis was performed on selected cracks to determine failure mode (i.e., fatigue, stress corrosion, overload, etc.). The characterization of areas of corrosion included identification of the type of corrosion present, a measurement of the surface area affected, maximum pit depth (if pitting corrosion), and an estimate of the thickness loss due to corrosion. Fifty-one findings were identified and characterized during the microscopic examination of the Cessna 402A, 292 findings were identified on the Cessna 402C, 303 findings were characterized on the Piper Navajo, and 172 findings were identified and characterized on the Beechcraft 1900D.

Figure 20 shows a comparison of the types of defects found during the teardown examination phase for each aircraft. This graph illustrates that the Cessna 402C exhibited more widespread corrosion than any other aircraft, which was expected since it was operated in a highly corrosive environment. As expected, the Cessna 402A showed the least amount of corrosion, as its primary usage was Grand Canyon tours (arid climate). One unanticipated outcome was the limited number of cracks found on the Cessna 402A. Grand Canyon usage is considered a severe fatigue usage by Cessna as the flights are short in duration, low to the ground (for sightseers), and turbulent. Figure 21 shows the number of findings by location on each aircraft.

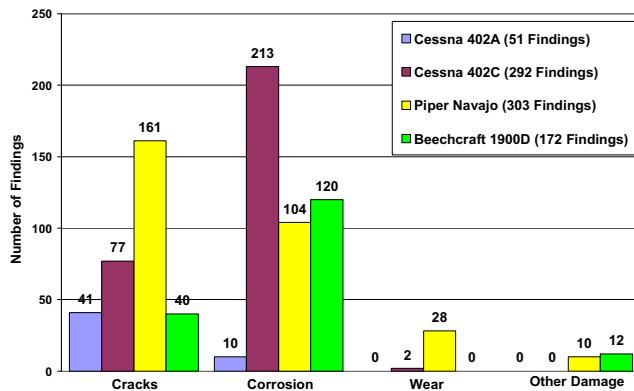


Figure 20 – Number of Findings by Type

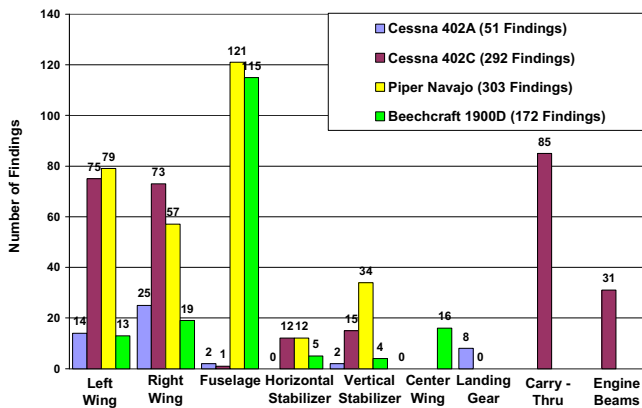


Figure 21 – Number of Findings by Location

MAJOR FINDINGS

Cessna 402A

During the inspection phase and teardown examination phase of the Cessna 402A, the following five major findings were identified and characterized:

- Broken teeth on fuel selector valve.
- Eight cracks on nose landing gear bellcrank.

- Crack on left stub wing upper front wing attachment fitting.
- Corrosion on right stub wing skin.
- Twenty cracks in auxiliary spars.

Figure 22 shows the broken teeth observed on the fuel selector valve during the routine airframe visual inspections on the Cessna 402A. A total of eight cracks were identified and characterized on the nose landing gear bellcrank, shown in Figure 23, during the supplemental inspections and the microscopic examination. These cracks ranged in length from 0.039 inches to 0.093 inches. Fractographic analysis revealed striations consistent with fatigue. Prior to this study, Cessna redesigned this component to resolve the cracking issue. The left stub wing upper front wing attachment fitting was found to have a 0.22-inch long 0.02-inch deep elliptical surface crack. Fractography of the fracture surface reveals 'blocky' intergranular features typical of stress corrosion cracking.

Severe corrosion was located in isolated areas on the right stub wing skin during disassembly. Small localized areas of 22.5% to 40% thickness loss were observed on the lower skin, while localized areas on the upper skin lost a maximum of 22.5% thickness due to pitting corrosion. A total of twenty cracks were found on the left and right auxiliary spars during disassembly, ranging in length from 0.041 inches to 0.740 inches. These cracks originated from the access plate, electrical wiring hole, and corners. An example of the cracks observed on the auxiliary spars is provided in Figure 24. Fractographic analysis of these cracks revealed a fatigue failure mode.



Figure 22 – Broken Teeth on Fuel Selector Valve (Cessna 402A)

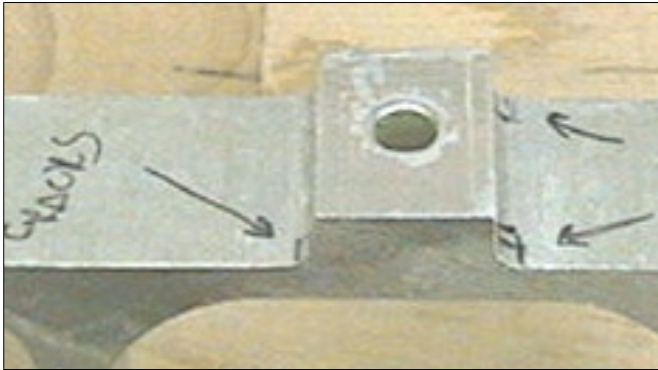


Figure 23 – Location of Cracks in Nose Gear Bellcrank (Cessna 402A)

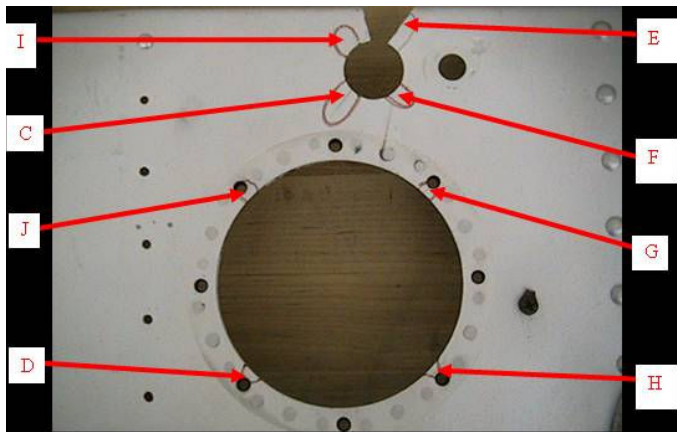


Figure 24 – Cracks around Access Panel and Wiring Hole on Right Wing Forward Auxiliary Spar (Cessna 402A)

Cessna 402C

The following five major findings were documented during the evaluation of the Cessna 402C teardown examination:

- Antenna corrosion.
- Crack on stub wing angle.
- Stub wing attachment fitting corrosion.
- Cracks on engine beams.
- Cracks in carry-thru structure.

During the disassembly of the Cessna 402C, corrosion was discovered under the antenna. The corrosion covered an area of 16.31 square inches resulting in a maximum thickness loss of 25%. Figure 25 shows this region of corrosion after removal of the antenna. Light surface corrosion under the antenna structure is common and was documented on all of the aircraft in this study, but the corrosion documented on the Cessna 402C was more severe than is typically observed.

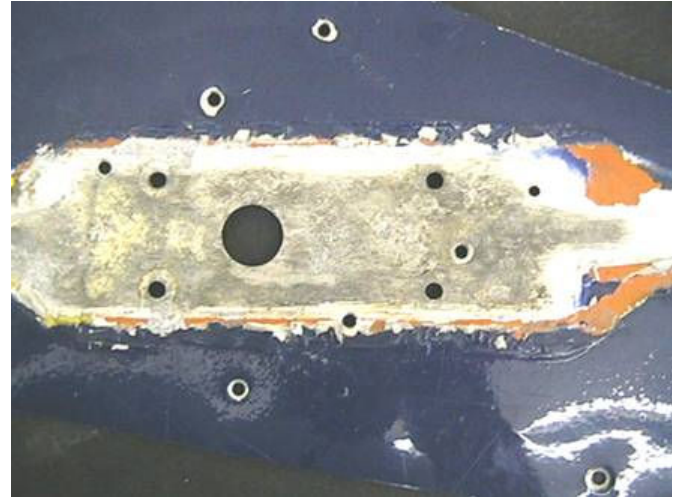


Figure 25 – Severe Corrosion Located Beneath Antenna (Cessna 402C)

During the supplemental inspections on the Cessna 402C, a 2-inch crack was discovered on the right stub wing rib angle. This angle is common to the right stub wing outboard rib and the forward carry-thru structure. Fractographic analysis was performed and it was determined that this crack was caused by fatigue.

Seven of the eight stub wing attachment fittings were found to have corrosion during the teardown examination. In general, the corrosion on the aft fittings was relatively light, but the forward fittings had much more corrosion, primarily at the attachment points. All of the wing attachment fittings had corrosion in several areas. The most significant corrosion was located around the attachment points, but light corrosion was found on different areas of the fittings. The maximum thickness loss measured on the stub wing and wing attachment fittings was 10%. An example of this corrosion is provided in Figure 26. Several of the fastener holes in the attachment fittings also had areas of metal separation due to mechanical damage during fastener removal.



Figure 26 – Example of Corrosion on Stub Wing and Wing Attachment Fittings (Cessna 402C)

Three of the four engine beams had a total of twenty-five cracks and several areas of corrosion. One of these cracks is shown in Figure 27. Fractographic analysis concluded that these cracks were caused by fatigue. Forty-three cracks were found on the carry-thru beams and webs. Light surface corrosion was also found on the carry-thru beams. An overview of the carry-thru structure is provided in Figure 28.

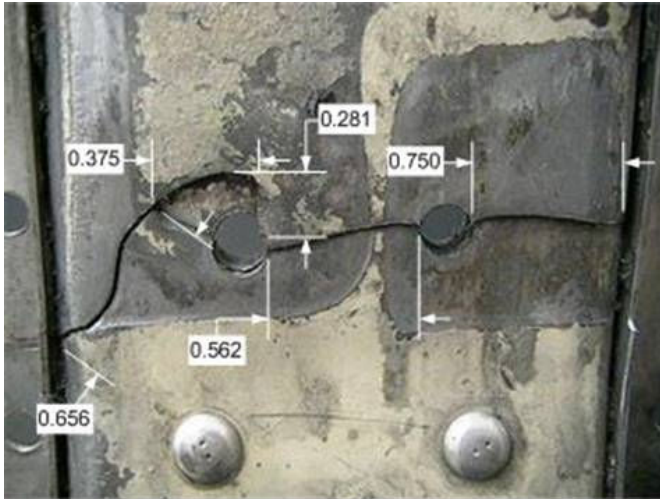


Figure 27 – Engine Beam Crack (Cessna 402C)

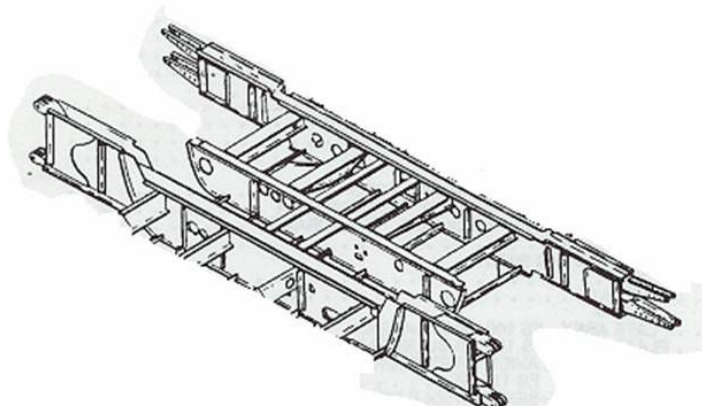


Figure 28 – Overview of the Wing Carry-Thru Structure (Cessna 402C)

Piper Navajo Chieftain

The following four major findings were identified and characterized during the teardown examination of the Piper Navajo Chieftain:

- Crack in the nose gear oleo strut housing.
- Corrosion on the rudder torque tube.
- Cracks in the main landing gear wheel wells.
- Cracks in the front spar fitting support brackets.

Figure 29 shows a 1.24-inch crack in the nose gear oleo strut housing. The survey of the aircraft maintenance records revealed that numerous reports have been made of similar damage fleet-wide due to improper towing procedures and exceeding the turn limitations of the aircraft.



Figure 29 – Crack in Nose Gear Oleo Strut Housing (Piper Navajo)

Figure 30 shows areas of through thickness corrosion on the rudder torque tube. Airworthiness Directive (AD) 2003-24-07 addresses this issue specifically; however, the NIAR team found that the true extent of corrosion was difficult to assess without removing the paint.

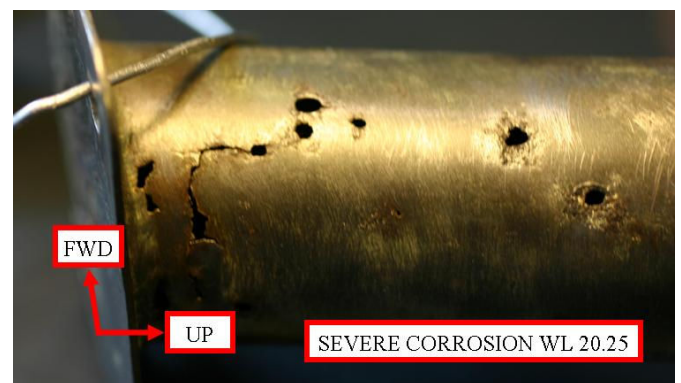


Figure 30 – Corrosion on Rudder Torque Tube (Piper Navajo)

Information obtained during the survey of the aircraft maintenance records revealed numerous instances of the wheel brake assemblies contacting and damaging the main landing gear wheel well ribs and stringers. On the Piper Navajo, this type of damage was observed on both the left and right main landing gear wheel wells. Pictures of this damage are provided in Figure 31 and Figure 32. Repairs have been performed in both the left and right main landing gear wheel wells to correct this issue; however, these repairs have also been damaged in the same manner.

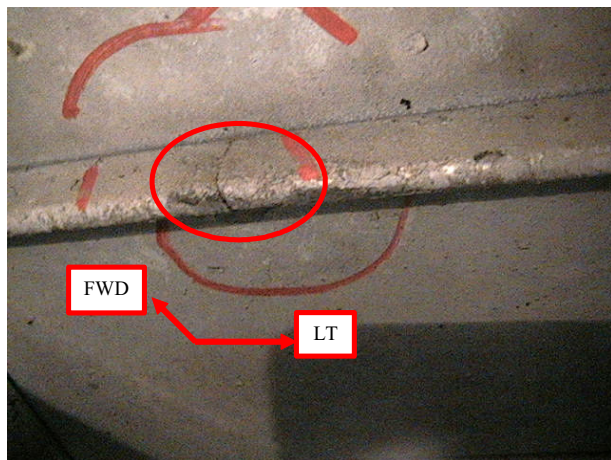


Figure 31 – Damaged and Cracked Right Wing Stringers in Main Landing Gear Wheel Well (Piper Navajo)

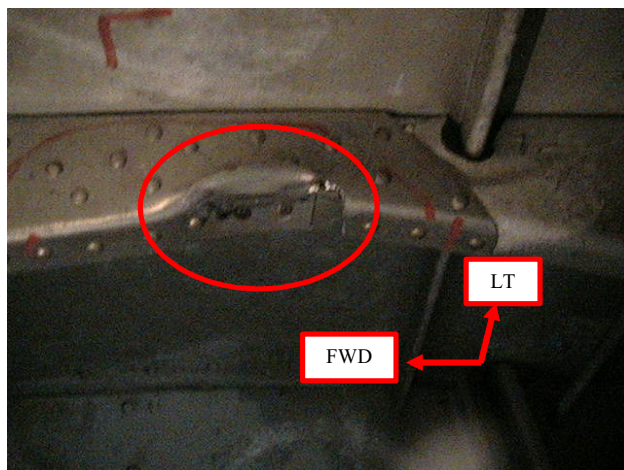


Figure 32 – Damaged Left Wing Ribs in Main Landing Gear Wheel Well (Piper Navajo)

Three cracks were identified on the right front spar fitting support bracket. These cracks measured 0.353 inches, 0.310 inches, and 0.112 inches. Three additional cracks were found on the left front spar fitting support bracket measuring 1.000 inches, 0.334 inches, and 0.250 inches. The location of the cracks is provided in Figure 33 and a picture of the orientation of the cracks observed is provided in Figure 34.

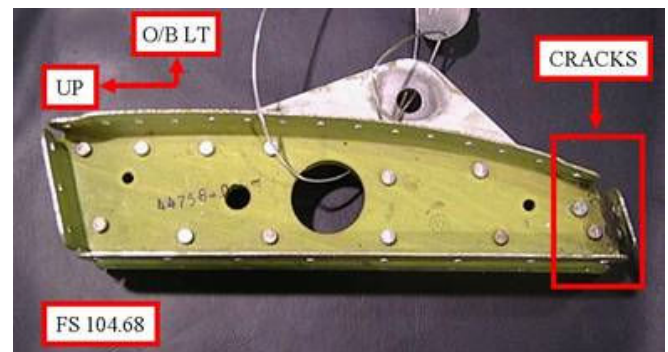


Figure 33 – Location of Front Spar Fitting Support Bracket Cracks (Piper Navajo)

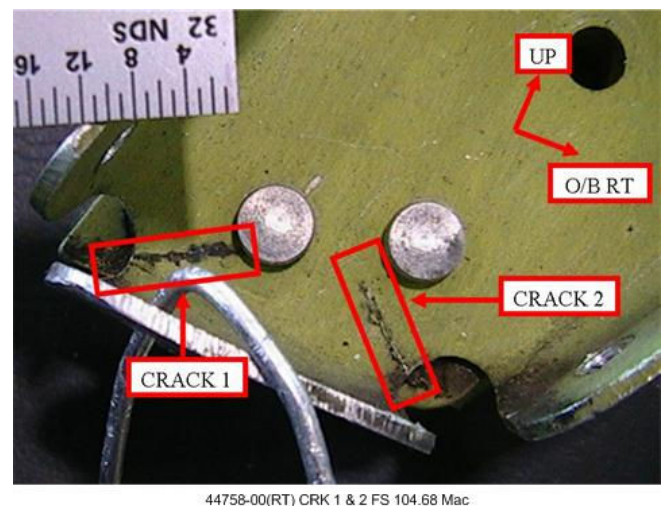


Figure 34 – Cracks on Front Spar Fitting Support Bracket (Piper Navajo)

Beechcraft 1900D

During the teardown examination of the Beechcraft 1900D, the following three major findings were observed:

- Cracks on the escape hatch gussets.
- Cracks on the seat track support assembly.
- Cracks on the cargo door hinges.

Six cracks ranging in length from 0.19 inches to 0.59 inches were identified on the escape hatch gussets. Three of these cracks are pictured in Figure 35. Two cracks, pictured in Figure 36, were identified and characterized on the seat track support assembly. These cracks measured 0.75 inches and 0.125 inches in length. Two cracks, measuring 0.358 inches and 0.115 inches, were observed on the cargo door hinge. These cracks are shown in Figure 37.

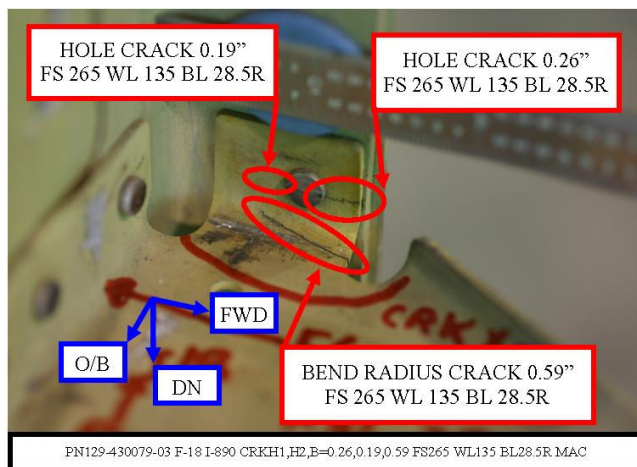


Figure 35 – Macroscopic View of Multiple Cracks on Escape Hatch Gusset (Beechcraft 1900D)

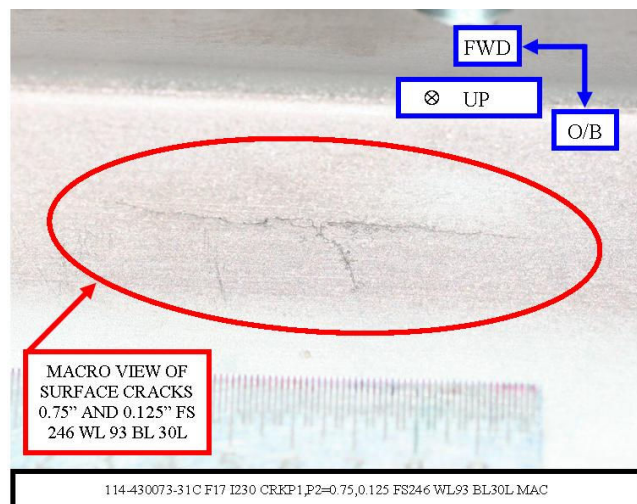


Figure 36 – Cracks on Seat Track Support Assembly (Beechcraft 1900D)

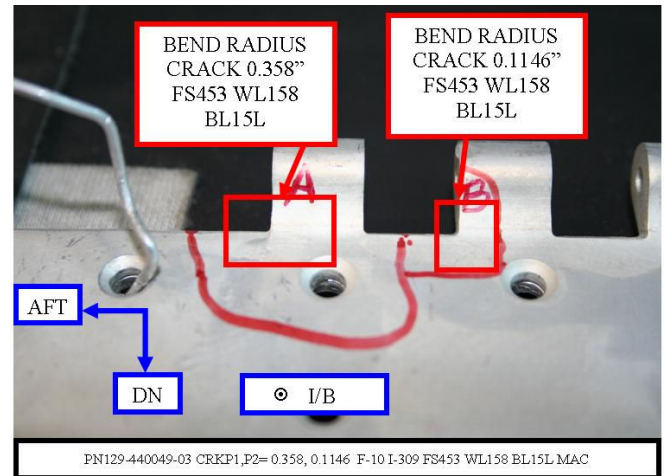


Figure 37 – Macroscopic View of Multiple Cracks on Cargo Door Hinges (Beechcraft 1900D)

EVOLUTION OF THE TEARDOWN PROCESS

Over the five year duration of the teardown program the following “lessons learned” emerged specific to the teardown process:

- Proper identification of detail parts in the disassembly phase of teardown is critical to understanding defect location/orientation.
- Defect documentation methods must be very thorough (document more information that you think you need).
- Establish a photographic documentation hierarchy.
- Add as much documentation (aircraft location, orientation, defect size, etc.) to the photographs as possible without cluttering.
- Optimize post disassembly inspection methods if possible.
- Develop corrosion categories that represent severity.

During detailed disassembly it is important to properly identify all detail parts as they are removed from the major aircraft section. Initially in the program, this was performed by simply documenting the part number of the piece removed. As the program evolved, NIAR teardown personnel determined more information was necessary. The following procedure was developed as the teardown program matured:

- With the detail part still attached to the major aircraft section or immediately following removal use a permanent marker to label aircraft orientation and aircraft location by the appropriate combination of fuselage station (FS), water line (WL), body line (BL), and wing station (WS).

- Create a pair of metal tags (one for each end of the part) containing part number, relevant reference photographs from the illustrated parts breakout and aircraft location by the appropriate combination of FS, WL, BL, and WS.
 - Two metal tags are used for redundancy, since all permanent marker information will be removed in the coating removal process.
 - Use steel tags and heavy gage aluminum wire to withstand the coating removal process.
 - Assure the metal tags both remain attached to the part for the duration of the teardown process. If one tag is lost, quickly replace it to avoid the possibility of unidentifiable detail parts.
 - Manufacturing of tags also evolved from a hammer and chisel combination, to a vibroetching technique, to a commercially manufactured tag machine.

Each defect should have a minimum of the following information documented:

- Part number.
- Defect type (crack, corrosion, mechanical damage, etc.).
- Defect location by FS, WL, BL, and WS.
- Defect orientation (if orientation is appropriate).
- Defect extent (length, depth, surface area, etc.).
- Defect detection method (visual inspection, SID, post disassembly NDI, microscope, etc.).

The more extensive the photographic documentation system is for each defect, the more information can be gained years after the teardown is complete. All pictures should include a scale for size reference and multiple pictures in each of the following categories should be taken to minimize lost data due to out-of-focus pictures, misplaced pictures, etc.:

- Illustrated parts breakout of total assembly or global photograph of assembly with local damage region identified.
- Macroscopic view of defective location.
- Microscopic view of defect.
- Fractography of defect, if applicable.

The addition of defect information to photographic documentation is an essential communication tool. This can be illustrated by comparing Figure 24 and Figure 37 of this paper. No vital information regarding the defect extent, location, orientation, or even type is provided in Figure 24. One would have to either look this information up in a table or rely on the verbiage describing the figure to obtain information. Figure 37, on the other hand, provides the defect type, extent, location, and aircraft orientation permanently on the photograph. A photographic template such as the one used to create Figure 37 also assures the same information is included on each picture for each defect. However, one must be careful not to clutter photographs with unnecessary information.

With the large number of parts to be inspected during the post disassembly NDI phase of the teardown examination, it is crucial to optimize the inspection process as much as possible.

This was evident by comparing the FLP inspections performed on the Cessna 402A and the Beechcraft 1900D. Most of the FLP inspections of the Cessna 402A were performed using an aerosol FLP can and lint-free wipes. While this method is acceptable it was not very efficient. A dip tank and rinse station FLP line was installed prior to the post disassembly NDI of the Piper Navajo. This station, pictured in Figure 38 allows inspectors to dip the part into a tank of FLP instead of using an aerosol can and remove the penetrant in a rinse station, as opposed to wiping the part off using rags. This investment drastically improved the efficiency and reduced the man-hour expenditures to perform the required FLP inspections.



Figure 38 – Fluorescent Liquid Penetrant Inspection Line

When this teardown examination program began, many techniques were available for corrosion categorization. Many of these methods consisted of measuring the depth at the location of maximum thickness lost (i.e., the depth of the deepest pit) and categorizing the corrosion based on that depth.

NIAR personnel felt it was more important to categorize corrosion based on the percentage of thickness lost. A 0.010-inch deep pit poses a different risk to structural integrity in a thin part compared to a thicker part. NIAR developed the following categories for corrosion classification based on maximum thickness loss:

- 0-2% Light Corrosion
- 2-5% Light-Moderate Corrosion
- 5-7% Moderate Corrosion
- 7-10% Moderate-Severe Corrosion
- >10% Severe Corrosion

NIAR AGING AIRCRAFT RESEARCH LAB GROWTH

During the duration of the five-year research program, twelve full time staff and twenty-two Wichita State University students were affiliated with the teardown examinations. One Masters Degree in Aerospace Engineering and three Level II NDI certifications were obtained to support this program. One airframe and power plant license with inspection authority was also maintained during the program.

The teardown examination experience NIAR personnel obtained allowed NIAR to participate in a number of large civilian and military teardown programs outside the scope of this research program. These teardowns included:

- T-34 Wings Teardown (FAA)
- Beechcraft Starship Teardown (FAA)
- Boeing 737 Composite Horizontal Stabilizer Teardown (FAA)
- Boeing B-52 Systems and Structural Teardown (Boeing Military)
- F-16 Static Test Article (Lockheed Martin)
- T-37 Wings Teardown (United States Air Force Academy)
- C-130 Center Wing Box Teardown (United States Air Force Academy)

- C-5 Teardown (United States Air Force)
- KC-135 Structural Teardown (United States Air Force)

CONCLUSION

Upon completion of the teardown evaluations of all four aircraft, it was apparent that the condition of the aircraft structure and systems are directly impacted by the usage and operating environment of the aircraft. The results of this research point to some recommended actions, which are listed below and may improve the continued airworthiness of aging general aviation airplanes.

1. **Aircraft antenna should be removed during inspections to inspect for corrosion.** The most severe corrosion found during these evaluations was found under the antenna. Input from experienced mechanics and engineers suggest this issue likely extends to the entire general aviation fleet. Since antennas need some electrical current to function, they create a local environment conducive to corrosion.
2. **Inspections of the Cessna 402C wing carry-thru and engine beam structures should be considered for additions to the SID.** Given the large number of cracks found in the carry-thru webs and engine beams of the Cessna 402C, inspection and possible repair of these areas could be beneficial to continued airworthiness.
3. **Periodic cycling of circuit breakers should be performed.** This research showed that the functionality of circuit breakers is an issue. When initially tested on the Cessna 402s, seven of sixty-seven circuit breakers did not meet their design specification for overload protection. The functionality of all but two of the seven circuit breakers improved to meet specification when they were thermally or mechanically cycled. This study shows that periodic cycling will improve the reliability of circuit breakers and also identify defective breakers to be replaced.
4. **Maintenance and inspection practices regarding wiring could be significantly improved.** The results obtained during this program show that a variety of wiring repairs and modifications are not being performed correctly. Through better training and with inspections targeted at specific wiring locations, wiring precursors, such as exposed conductors and heat damaged wires could be found and repaired prior to a problem with the aircraft's wiring system.

5. **Owners and operators should adhere to the manufacturer's instructions for continued airworthiness.** Routine visual inspections were not comprehensive or intrusive enough to identify all areas of concern found on the four aircraft. The supplemental inspections, service bulletins, and airworthiness directives do address a majority of the concerns due to aging effects that were discovered in the teardown evaluations.
6. **Maintenance inspection programs for general aviation aircraft should include supplemental inspections.** As noted above, the research showed that routine visual inspections were not enough to identify all areas of concern. Supplemental inspections should be predicated on engineering damage tolerance analysis and service history. These supplemental inspections should be comprehensive and should commence later in the aircraft's service life, but early enough to detect the onset of age related problems.
7. **Owners and machinist need to understand the importance of diligent Service Difficulty Reporting (SDR).** The SDR database for the models evaluated did not represent many of the major findings from the teardown examinations. Since SDR reporting is voluntary, many significant concerns found in the field go unreported. The service history is therefore not captured and does not benefit the collective fleet.
8. **Results from all teardown evaluations should be used to develop FAA guidance and changes to regulations.** This research highlighted several areas where maintenance and inspection practices could be improved with better guidance. This research also indicates that routine inspections for general aviation airplanes may not be adequate to safely maintain them as they age. This research can assist manufacturers and operators in developing supplemental inspection programs to enhance the current maintenance and inspection programs for aging airplanes.

ACKNOWLEDGMENTS

The author of this paper would like to thank Mr. Michael Shiao and Mr. Marvin Nuss (both FAA personnel) for their technical support and guidance during the teardown program.

REFERENCES

1. Cope, D.; Laubach, M.; Shiao, M. and Nuss, M., "Teardown Evaluation of a 1969 Cessna 402A Model Aircraft," Final Report submitted to FAA William J. Hughes Technical Center, December 2005
2. Cope, D.; Laubach, M.; Shiao, M. and Nuss, M., "Teardown Evaluation of a 1979 Cessna 402C Model Aircraft," Final Report submitted to FAA William J. Hughes Technical Center, December 2005
3. Cope, D.; Laubach, M.; Shiao, M. and Nuss, M., "Teardown Evaluation of a 1975 Piper Navajo Chieftain Model PA31-350 Aircraft," Draft Report submitted to FAA William J. Hughes Technical Center, January 2006
4. Cope, D., Moreno, M., Shiao, M. and Nuss, M., "Teardown Evaluation of a 1993 Beechcraft 1900D Aircraft," Draft Report submitted to FAA William J. Hughes Technical Center, May 2007
5. Cessna Aircraft Company, "Cessna 402A Maintenance Manual," Cessna Aircraft Company, Wichita, Kansas
6. Cessna Aircraft Company, "Cessna 402C Maintenance Manual," Cessna Aircraft Company, Wichita, Kansas
7. New Piper Aircraft Company, "Piper Navajo Chieftain Maintenance Manual," New Piper Aircraft Company, Vero Beach, Florida
8. Cessna Aircraft Company, "Cessna 402A Supplemental Inspection Document," Cessna Aircraft Company, Wichita, Kansas
9. Cessna Aircraft Company, "Cessna 402C Supplemental Inspection Document," Cessna Aircraft Company, Wichita, Kansas

CONTACT

- Melinda Laubach, MS Aerospace Engineering
Senior Research Engineer
National Institute for Aviation Research
Wichita State University
1845 Fairmount, Box 93
Wichita, Kansas 67260
(316) 978-8205
melinda.laubach@wichita.edu
www.niar.wichita.edu

DEFINITIONS, ACRONYMS, ABBREVIATIONS

AD	Airworthiness Directive
BL	Body Line
CAR	Civil Aviation Regulation
CFR	Code of Federal Regulations
etc.	etcetera
FAA	Federal Aviation Administration
FLP	Fluorescent Liquid Penetrant
FS	Fuselage Station
ft	feet
ICA	Instructions for Continued Airworthiness
i.e.	that is
lbs	pounds
NDI	Nondestructive Inspection
OEM	Original Equipment Manufacturers
PSE	Principal Structural Element
SDR	Service Difficulty Reporting
SID	Supplemental Inspection Document
SIM	Structural Inspection Manual
WL	Water Line
WS	Wing Station

The Fatigue Behavior of Fastener Joints

K. S. Raju, B. L. Smith and F. Caido
Wichita State University

C. Gomez
National Institute for Aviation Research

M. Shiao
FAA William J. Hughes Technical Center

ABSTRACT

The fatigue behavior of Hilok fastener joints under constant amplitude loading has been investigated experimentally. The effects of load transfer in an unbalanced joint configuration was characterized in terms of a stress severity factor relative to the open-hole configuration. The experimental data indicates that the clamp-up forces dominate the performance of fastener joints with the open-hole fatigue life being the lower bound at the stress levels investigated. The failure modes were observed to transition from a net-section type failure across the minimum section to a fretting induced failure at some distance from the hole. The experimental data has been used to develop stress severity factors to be used as a measure of the fatigue quality of the fastener joints.

INTRODUCTION

The fatigue behavior of fastener joints has received widespread attention for several years due to their extensive use in metallic airframe structures. The fastener joints serve the primary purpose of transferring loads across structural members through bearing and by clamp-up induced friction in some instances. The stress and displacement fields arising due to load transfer by fasteners has been studied in detail by Barrios[1]. The increase in stress concentration due to the combined effects of load bypassing a hole and the hoop-stress due to bearing load transfer have been shown to decrease the fatigue life of the joint[1-4].

The fatigue behavior of fastener joints with different levels of load transfer was studied by Lee [3]. The behavior of 7075-T7351 aluminum fastened by traditional (nut-bolt) fastener under constant amplitude fatigue loading ($R=0.05$) was addressed. The authors observed a decrease in both crack initiation life and total fatigue life with increasing levels of load transfer.

The clamp-up loads generated by the fasteners could have a significant effect on the load-transfer mechanism and the resulting failure modes and life under fatigue loading. Starikov [4] reports the influence of frictional forces arising due to fastener clamp-up on load-transfer and fatigue life of fastener joints. Using an instrumented fastener, they measured the load transferred by bearing and due to friction at the faying surfaces. The authors observed fretting fatigue as the major mechanism inducing the initiation of fatigue fracture in the joint.

The current AFS-120-73-2 document [5,6] uses Miner's linear cumulative damage theory [7] to evaluate the fatigue strength of a wing and associated structure of part 23 small airplanes using S-N curves from full-scale tests. The S-N curves contained in AFS-120-73-2 document are based on full-scale wing test results for a single type of wing. Full-scale tests were conducted because they incorporate important variables such as stress concentrations, multiple load paths due to adjacent elements, stress redistribution that occurs as damage propagates, fretting, load transfer rates, and material variability. The main problem with the current methodology is that it does not account for variations in stress concentrations and load transfer that occur from one wing to another. Furthermore, the current methodology does not accurately predict fatigue damage accumulation, especially in older aircraft that are subjected to severe flight conditions. This uncertainty results in high maintenance costs and potential catastrophic consequences and thus is a major concern for operators as well as regulatory agencies around the world. The methodology presented herein provides a direct approach for evaluating fatigue lives of fastener joints. Fatigue test data are used to establish an effective stress concentration factor that takes into account individual structural details. S-N curves are used with this effective stress concentration to evaluate the fatigue life of the structure.

fatigue life has been addressed. The details of the experimental program, results and observations are presented in this paper.

In the present study empirical factors β and θ were developed using fatigue tests conducted on open-hole (OH), filled-hole(FH) and load-transfer(LT) specimens. The nominal load transfer levels addressed in this study are 30% (Intermediate Load Transfer – ILT) and 50% (Medium Load Transfer – MLT). The details of the materials, specimen geometries, load transfer characterization and fatigue testing are described in this section.

Table 1. Nominal Mechanical Properties of clad 2024-T3 Aluminum [9]

Young's Modulus, E	72.4 GPa
Poisson's ratio, ν	0.33
Ultimate tensile Strength	427 MPa
Tensile yield strength	324 MPa
Ultimate shear strength	262 MPa
Failure strain	15%
Bearing yield ($e/d=1.5$)	482 MPa
Bearing yield ($e/d=2.0$)	579 MPa

Technical drawing of a mechanical part with dimensions in mm. The part has a total length of 330.2 mm and a height of 50.04 mm. It features a central hole with diameter $\varnothing d$ and a fillet radius R . Key dimensions include 68.33 mm, 40.55 mm, 165.1 mm, and $w/2$.

The load-transfer specimens used in the current study was a variant of the 1½ doppel specimen configuration

$$SSF = \frac{\sigma_{loc}}{\sigma_{ref}} = \frac{\alpha\beta}{\sigma_{ref}} \left\{ K_{tb} \frac{\Delta P}{dt} \theta + K_{tg} \frac{P}{wt} \right\} \quad (1)$$

While the effects of load transfer on the fatigue life of specific fastener joints under constant amplitude[3] and spectrum loading[4] has been reported in literature, no attempts have been made to develop the aforementioned constants based on the data. The development of fatigue based empirical constants will facilitate the use of Miner's damage rule while suitably accounting for the effects of load transfer. In the current study, the constant amplitude fatigue behavior of single-lap fastener joints have been investigated experimentally. The experimental data has been used to generate the empirical factors β and θ as a function of stress levels or fatigue life. The effects load transfer level and interference/clearance fit of the fastener on the

[2,3] as illustrated in figure 3. The specimen assembly consists of a main (or bypass) part which is the primary member and a load transfer (or secondary) part. The two parts are connected to each other using the fastener (under study) in the gage section and constraint pins in the gripping region. Ensuring compatibility of displacements in the gage region is necessary to force load transfer between the two parts [2]. The far-field loads are introduced into the main part of the assembly. Depending on the relative stiffness of the load transfer part, a specific amount of load is transferred to it via the fastener. The stiffness can be varied by altering the cross-section dimensions and the material itself. The widths of the load-transfer part of the ILT and MLT specimens are listed in table 2. Due to the inherent asymmetry in load transfer, the specimen is prone to bending loads, which is classified as secondary bending [2-4]. The amount of bending will depend on the load transfer levels and the applied load itself [2].

In the present study, the load-transfer and main parts were manufactured from the same material. The displacement compatibility in the gripping region was achieved satisfactorily by using four 6.35mm diameter steel pins as illustrated in figure 3. This combination was arrived at by limited experimentation with different configurations.

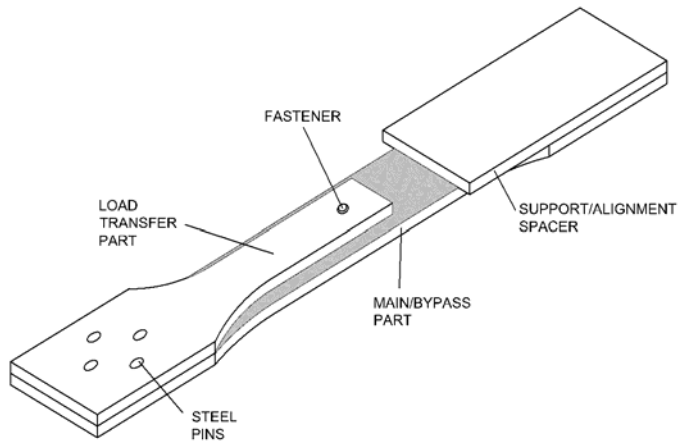


Figure 3. Load transfer specimen

Table 2. Nominal dimensions of test specimens

Specimen Type	Width of main part w_m (mm)	Width of load transfer part w_{LT} (mm)	Nominal Load transfer level
OH/FH	38.10	-	-
ILT	38.10	26.67	30%
MLT		38.10	50%

LOAD TRANSFER – The load transfer in the 1½ dogbone specimen occurs due to the enforced displacement compatibility between the main part and

the load transfer part. If the fastener is assumed to be rigid and the bearing deformations are negligible, the amount of load transferred is purely a function of the relative longitudinal stiffnesses of the main part and the load transfer part. However, owing to the deformation of the hole, bending of fastener and rotation of the joint itself, the load transfer level will vary as a function of applied loading[2]. The presence of fastener clamp-up loads and hole clearance will further complicate the load transfer process due to the participation of the frictional load transfer at the faying surfaces.

In the present study, the load transfer was characterized by conducting static tests on instrumented specimens. Since the far-field strains are elastic, they were used as a measure of load in the specimen. As the joint is unbalanced (single shear), secondary bending occurs and the strains are not uniform across the thickness of the two parts. Thus, back-to-back strain gages were used as shown in the figure 4.

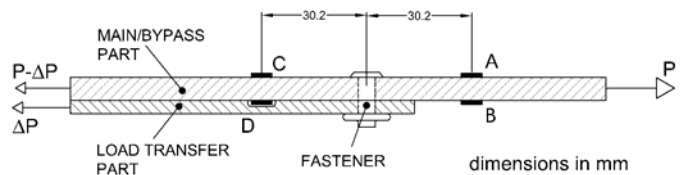


Figure 4. Details of load transfer specimen and location of strain gages along longitudinal plane of symmetry

The variation of load-transfer in the ILT test specimen subjected to a tension-compression loading cycle is shown in figure 5. Anti-buckling restraint supports were used to prevent buckling during the compression phase of the test. The load transfer in the specimen was computed by using the following equation

$$\% \text{ Load Transfer} = \left[1 - \frac{\epsilon_C + \epsilon_D}{\epsilon_A + \epsilon_B} \right] \times 100 \tag{2}$$

The load transfer computed using the above equation does not resolve the bearing and frictional components of the load transfer. The load transfer was observed to vary with the applied loading with a conspicuous initial drop in the load transferred. This can be attributed to the clearance hole used in the specimen. The friction at the faying surface dominates the initial load transfer process and bearing load transfer takes over when the frictional load transfer ceases and the fastener engages the hole surface. The maximum load transfer level achieved under tensile loading is about 26%. However, the load transfer under subsequent compression loading does not reach the levels achieved under tensile loading. The bearing stress corresponding to the applied tensile loading was higher than the nominal bearing yield stress listed in table 1. Thus the hole is enlarged due to plastic deformation, and in addition to the originally existing clearance results in poor load transfer response during

the compression phase of the test. Additional tests at lower tensile and compression stress levels resulted in comparable load transfer levels under both loading phases. The measured load transfer in MLT specimen is shown in figure 6. A load transfer of about 40% was achieved during the tension phase and about 50% during the compression phase of the loading. Based on a limited number of fatigue tests (up to 1000 cycles) during which the strains were monitored, it was observed that the load transfer levels stabilized after roughly 10 cycles.

The secondary bending in load transfer specimens were characterized by using the secondary bending ratio [2,3] which is defined as

$$\text{Secondary bending ratio} = \frac{\varepsilon_A - \varepsilon_B}{\varepsilon_A + \varepsilon_B} \quad (3)$$

The above equation applies to the load introduction side of the specimen. Secondary bending ratio can also be obtained for the bypass or load transfer side of the specimen using the strain data from gages C and D. The secondary bending ratio in the ILT test specimen during a tension-compression cycle is shown in figure 7. The secondary bending in the present study is based on far-field strains while the recommended AGARD points [2,3] are closer to the hole.

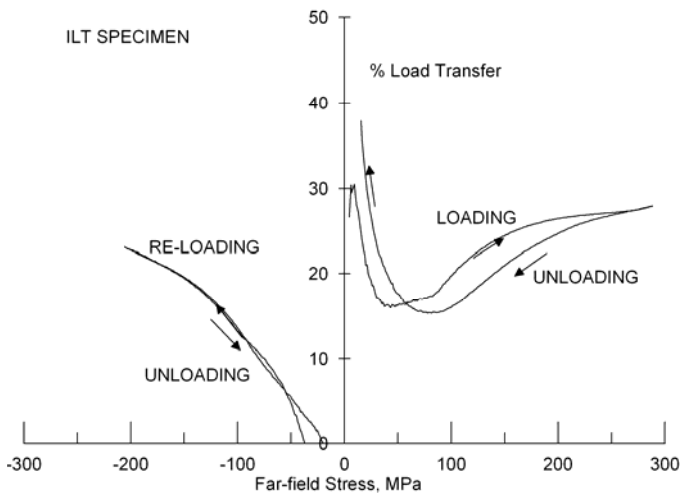


Figure 5. Measured load transfer in ILT specimen during one tension-compression loading cycle.

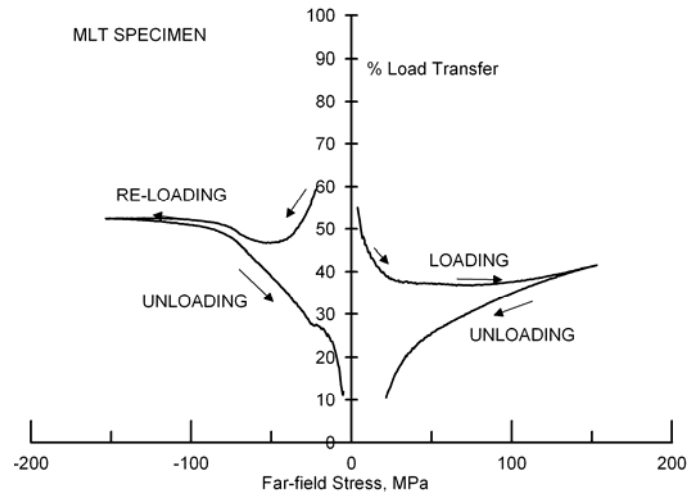


Figure 6. Measured load transfer in MLT specimen during one tension-compression loading cycle.

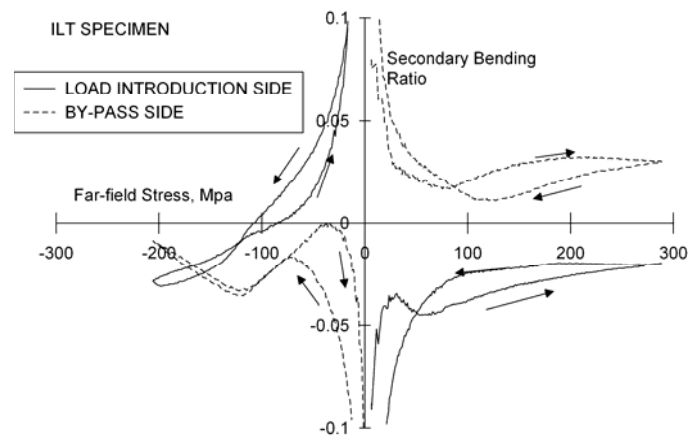


Figure 7. Secondary bending in ILT specimen during one tension-compression loading cycle

FATIGUE TESTING – The constant amplitude fatigue tests were conducted using 24kN capacity MTS servo hydraulic load frames. The test control was accomplished using MTS Multipurpose TestWare computer program. The test specimens were fatigued at mean stress levels of 20.68MPa and 41.37MPa, with a frequency of 20Hz. The test frequency was reduced to 10Hz for the higher stress amplitudes. Due to the choice of the mean stress levels, negative stress ratios were encountered for a major portion of the tests. To circumvent the possibility of global buckling of the specimen under compression loading, anti-buckling restraints were used during the tests.

The maximum stress levels were varied to achieve fatigue failure in the range of $1-10^3$, 10^3-10^4 , 10^4-10^5 , 10^5-10^6 and 10^6-10^7 cycles. A minimum of six specimens were tested at each maximum stress level to facilitate statistical analysis of the fatigue life. The number of cycles to failure and failure modes were recorded for each test.

DISCUSSION

The fatigue life of open-hole specimen as a function of the maximum stress level are illustrated in figure 8. The higher mean stress level increases the fatigue life which is consistent with the trends reported in literature. The test data ($K_{Tg} \approx 3.05$, $K_{Tn} \approx 2.72$) is compared with those reported in MIL-HDBK-5[9] but with a K_{Tg} of 1, 2 and 4. The curves were generated from the best-fit curve equations presented in the reference. The test data from the current study falls in between the curves corresponding to K_{Tg} of 2 and 4 as expected. The reference curves in the figures are for bare material and involve different test specimen geometries for a given K_{Tg} .

The fatigue life of filled-hole specimens using HiLOK fasteners are shown in figure 9. The open-hole data is also plotted in the same figure for comparison. The filled-hole specimens exhibited higher fatigue lives relative to the open-hole configuration. The effects of fastener interference at limited stress levels is illustrated in the same figure. An interference level of 0.038mm was used. A significant increase in fatigue life was observed with the interference holes at the two stress levels investigated. The generation of fatigue data for interference holes at other stress levels is presently under progress.

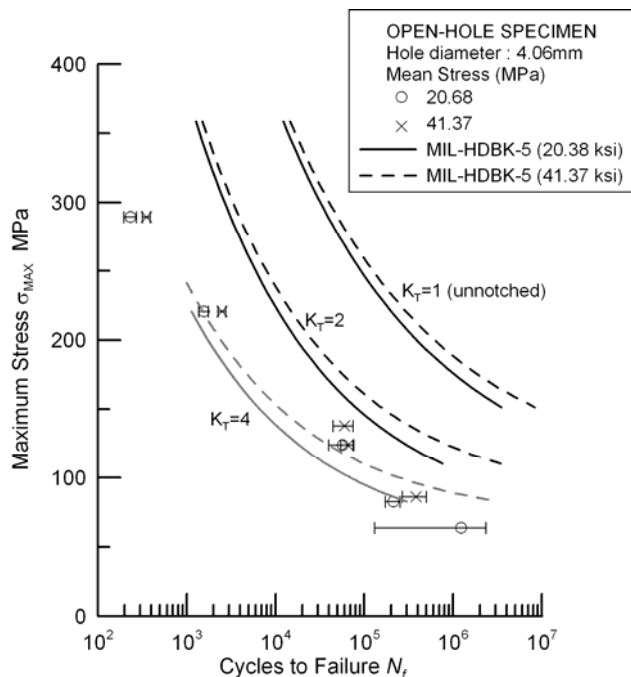


Figure 8. Fatigue life of open-hole specimens at two different mean stress levels.

The fatigue behavior of intermediate load transfer (ILT) and medium load transfer (MLT) specimens is presented in figures 10 to 13. Both open-hole and filled-hole data at the corresponding mean stress level are superposed on the load transfer data in these figures. Referring to figures 10 and 11, the fatigue life of ILT specimens tended to fall between the open-hole and filled hole

configurations. However, with interference holes, the fatigue life of ILT specimens tended to overlap or be lower than the open-hole fatigue life, especially at maximum stress levels above 150MPa. At lower stress levels, no significant differences were observed between the ILT specimens with interference and clearance holes.

The fatigue test data for ILT and MLT test specimens indicate that the fatigue life approaches that of the open-hole configuration with decreasing load-transfer levels. This contradicts with the observations of Lee [3] who reported a decrease in fatigue life with increasing levels of load transfer. However, the material, fastener type, fastener size, specimen width to fastener diameter ratio and maximum stress levels are significantly different from that of the present study. In addition, Lee's study involved tension-tension fatigue ($R=0.05$) unlike the present investigation where compression loads are present due to the choice of the mean stress levels.

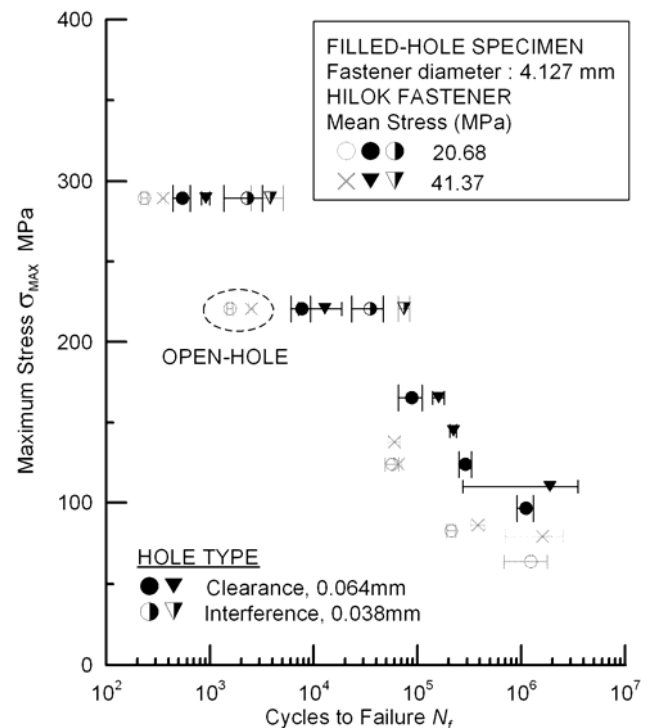


Figure 9. Fatigue life of filled-hole specimens at two different mean stress levels.

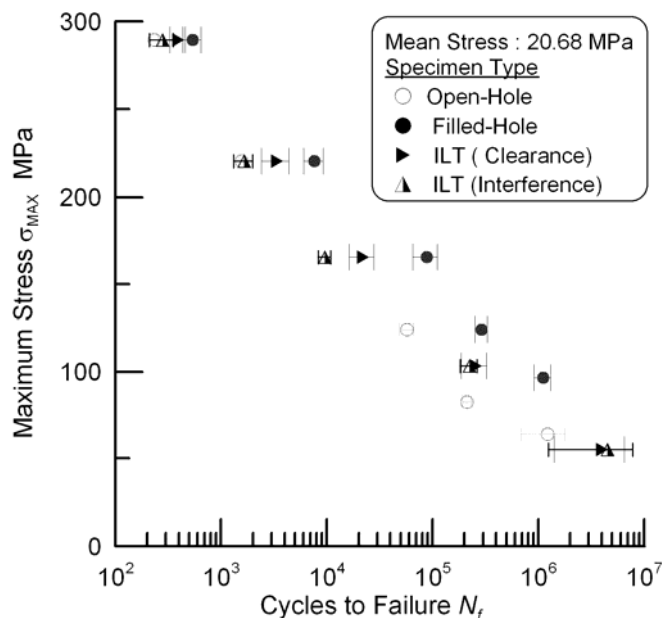


Figure 10. Fatigue life of ILT specimens at mean stress of 20.68MPa

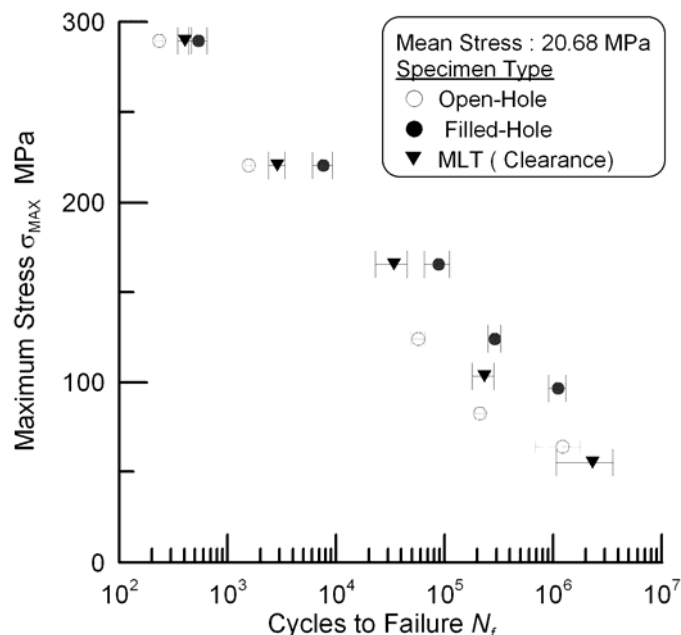


Figure 12. Fatigue life of MLT specimens at mean stress of 20.68MPa

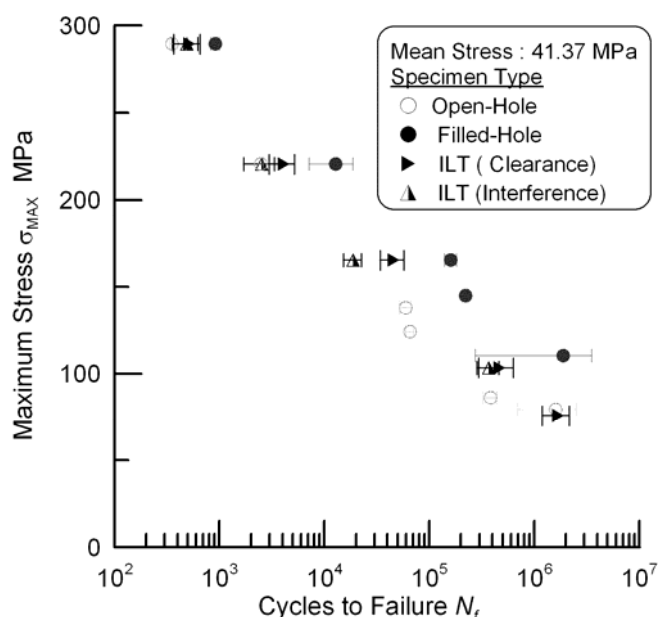


Figure 11. Fatigue life of ILT specimens at mean stress of 41.37 MPa

FAILURE MODES – The typical failure modes observed in load transfer specimens at different stress levels are illustrated in figure 14. The figure shows the faying surface of the main part of the specimen. The net-section fracture mode passing through the hole was characteristic of specimens failing below 5×10^5 cycles (an average value). The friction zone around the hole is evident in these specimens and is expected due to the presence of cladding layer. Frictional wear was also observed at localized regions away from the hole, where friction exists between the main and load transfer parts (edges) due to the tendency of the specimen to bend. The specimens tested at lower stress levels did indicate a friction zone around the hole but was not as pronounced as the previous case. However, the fracture was initiated due to fretting fatigue at a characteristic distance away from the fastener hole. These failure modes are similar to the observations of Starikov [4].

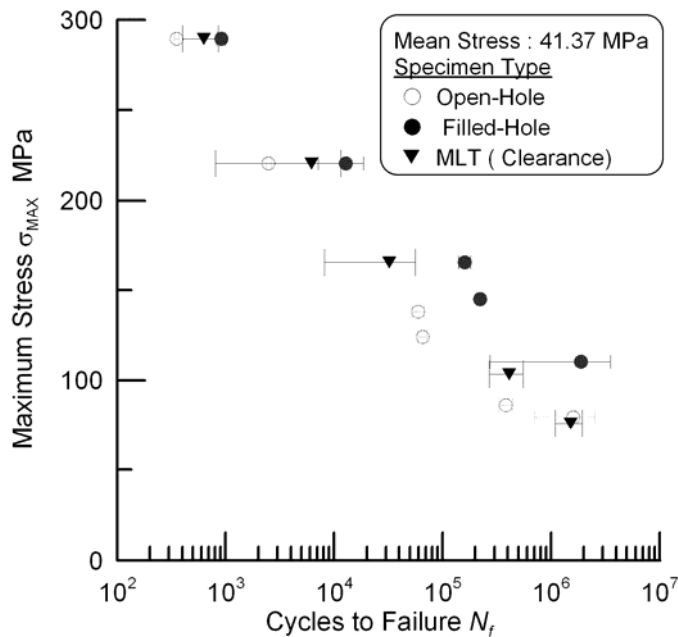


Figure 13. Fatigue life of MLT specimens at mean stress of 41.37 MPa

expressed as a function of the life. In the present study, the fatigue strength was expressed by the following equation

$$\ln(\sigma_{MAX}) = A + B \{ \ln(N_f) \}^C \quad (4)$$

The above equation was chosen heuristically, based on its ability to represent the test data. The typical stress-life curve for an open-hole configuration is shown in figure 15. The constants A, B and C in equation 4 were determined by curve fitting the experimental data using least-squares method. A normal distribution was assumed during the curve-fit process. Thus, the best-fit curve represents the fatigue strength as a function of mean life. The suspended test data and data at exploratory stress levels were not included in the curve-fit process. The parameters A, B and C obtained from curve-fitting of open-hole, filled-hole and load transfer test data are summarized in table 3. The least squares curve-fit was carried out using TableCurve 2D computer program [12]. It may be noted that similar stress-life curves could be generated for characteristic fatigue lives (e.g, B90, etc) by statistical analysis of the life data at each stress level investigated.

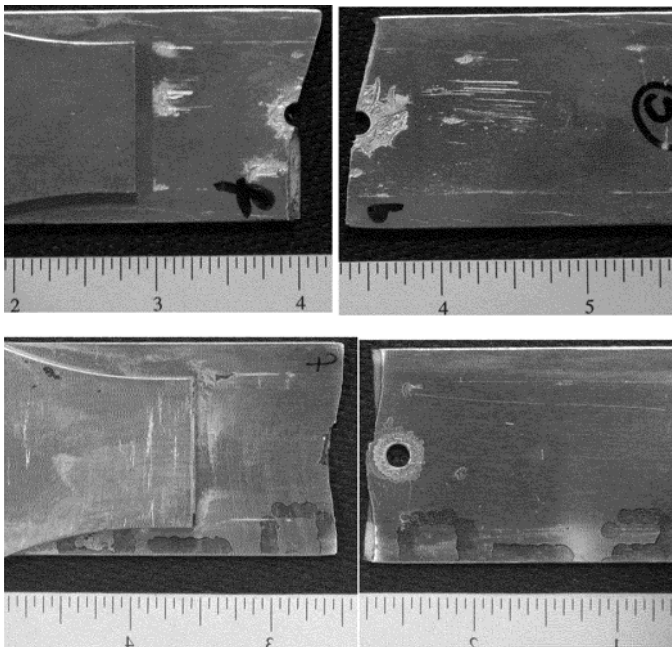


Figure 14. Typical failure modes in load transfer specimens : net-section fracture (top) and fretting induced fracture (bottom).

GENERATION OF EMPIRICAL CONSTANTS- The fatigue test data was subsequently used to generate the empirical factors α , β and θ for the ILT and MLT configurations, as a function of the maximum stress level or the fatigue life. The SSF at a specific fatigue life is defined as the ratio of the fatigue strength of the specimen with discontinuity (notch, load transfer, etc) under study to the fatigue strength of the open-hole configuration [8]. Since the fatigue stress levels rather than the lives were controlled during the tests, the generation of SSF requires that the fatigue strength be

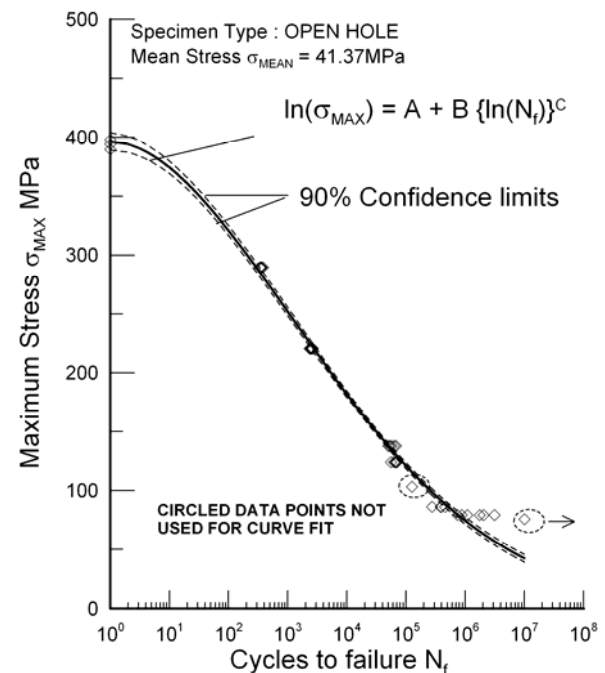


Figure 15. Typical S-n curve generated using least-squares curve fit.

Table 3. Summary of curve-fit parameters

Specimen Configuration	Mean Stress (MPa)	A	B	C
Open-Hole	20.68	5.978	-0.0133	1.877
	41.37	5.983	-0.0119	1.881
Filled Hole	20.68	5.978	-0.0089	1.915
	41.37	5.979	-0.0079	1.898
ILT (clearance)	20.68	5.981	-0.0095	1.965
	41.37	5.978	-0.0091	1.945
ILT (interference)	20.68	5.988	-0.0173	1.747
	41.37	5.987	-0.0120	1.863
MLT	20.68	5.984	-0.0088	2.003
	41.37	5.992	-0.0098	1.932

The stress-life curves were used to generate the empirical constants α , β and θ as a function of stress level or life. The fatigue life at a given stress level is inversely proportional to the SSF, or conversely, the fatigue strength for a given life is inversely proportional to the SSF. This is illustrated in figure 16.

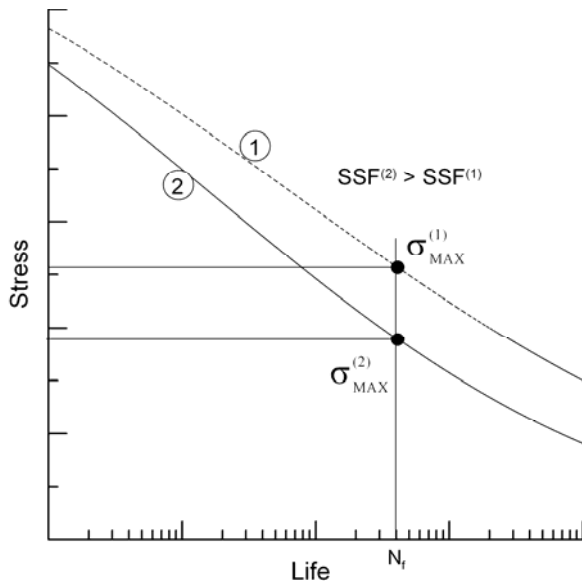


Figure 16. Fatigue strengths at a specified life for cases with different stress severity factors.

The fatigue strength for a given life may be written as

$$\sigma_{MAX}(n_f) \propto \frac{1}{SSF} \quad (5)$$

Thus, the ratio of fatigue strengths of the open-hole configuration and the filled hole configurations may be written as

$$\frac{\sigma_{MAX}^{OH}(n_f)}{\sigma_{MAX}^{FH}(n_f)} = \frac{SSF^{FH}}{SSF^{OH}} \quad (6)$$

where the superscripts OH and FH represent open-hole and filled-hole configurations respectively. The SSFs' can be expressed in terms of the stress concentration factor K_{tg} and the factors α and β as follows

$$\frac{\sigma_{MAX}^{OH}(n_f)}{\sigma_{MAX}^{FH}(n_f)} = \frac{\alpha_{FH} \beta_{FH} K_{tg}}{\alpha_{OH} \beta_{OH} K_{tg}} \quad (7)$$

The hole preparation factor α for the open-hole and filled-hole configurations are set equal to 1 (drilled hole) and 0.9 (reamed hole) respectively [9]. Further, the hole-filling factor β for the open-hole case is unity. Thus the hole filling factor β_{FH} may be written as

$$\beta_{FH}(n_f) = \frac{1}{0.9} \times \frac{\sigma_{MAX}^{OH}(n_f)}{\sigma_{MAX}^{FH}(n_f)} \quad (8)$$

The hole-filling factor may be recast in terms of the stress levels using the stress-life equation. The variation of the hole filling factor for the HiLOK fastener is illustrated in figure 17. The effects of interference hole is also illustrated in the figure, with a decrease in β . The observed trends must be interpreted with caution since the test data for the interference hole is limited and the corresponding stress-life curve will change with addition of new data at different stress levels. It must further be noted that the curves presented in the figure are dependent on the choice of stress-life equation. The hole-filling factors presented in the figure are associated with the mean life and similar curves may be generated for other characteristic lives.

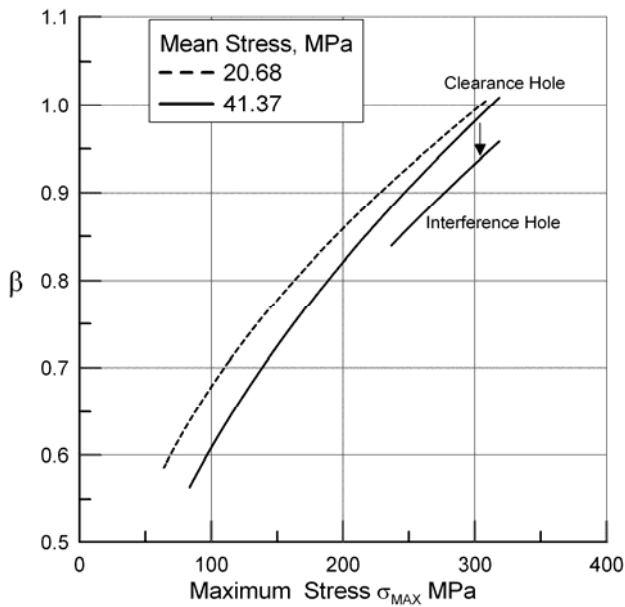


Figure 17. Hole-filling factor β for the Hilok fastener at two mean stress levels

The determination of the bearing factor θ , follows the same procedure used for the hole-filling factor. The ratio of the fatigue strengths of a load transfer and open-hole specimen can be written as

$$\frac{\sigma_{MAX}^{OH}(n_f)}{\sigma_{MAX}^{LT}(n_f)} = \frac{\alpha_{LT} \beta_{LT} \left(K_{th} \frac{\Delta P}{P} \frac{w}{d} \theta + K_{tg} \right)}{\alpha_{OH} \beta_{OH} K_{tg}} \quad (9)$$

where

$$\alpha_{LT} = \alpha_{FH}$$

$$\beta_{LT} = \beta_{FH}(n_f)$$

$$\frac{\Delta P}{P} = \text{nominal load transfer level}$$

Using the above equations and a nominal load transfer level (0.3 for ILT and 0.5 for MLT), the bearing factor θ may be expressed as a function of life or maximum stress under fatigue loading. The variation of bearing factor as a function of applied maximum stress is plotted in figure 18. The bearing factor can be observed to decrease as a function of the applied maximum fatigue stress level. In addition, the mean stress was observed to increase the bearing factor while an opposite effect was observed for the load transfer levels.

The SSF for the load transfer configurations may now be expressed as a function of the applied stress level using equation 1. The SSF can be used in conjunction with the open-hole data to predict the life of fastener joints. Further, Miner's rule may be utilized to predict the fatigue life of joints with known SSF, under general spectrum loading. Ostensibly, the SSF for the ILT and

MLT configurations will be about 3 or lower which contradicts the observations of Lee[3]. This may be attributed to several factors which include the use of clearance holes, clad material, negative stress ratios, and exceeding the bearing yield value for the material. Future work must address some of these issues, specifically the reduction of bearing stress levels and the use of interference holes. The bearing stress levels can be reduced while maintaining the percentage load transfer by reducing the actual load being transferred. This may be achieved by reducing the ratio of specimen width to hole diameter

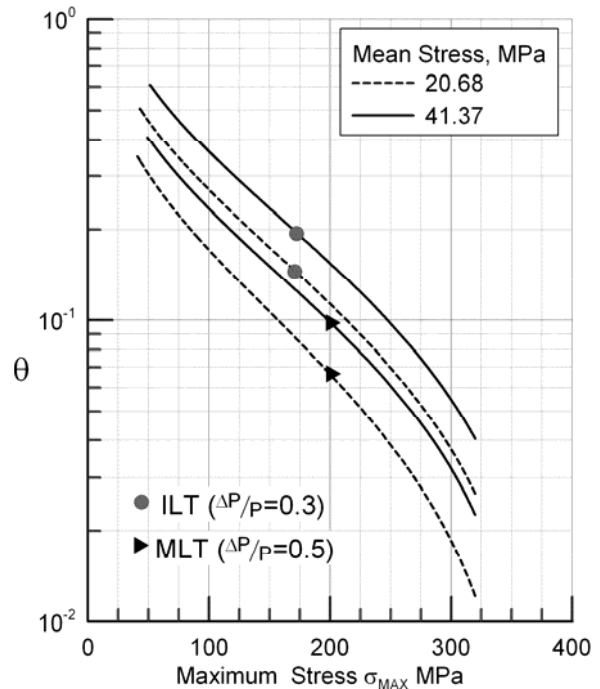


Figure 18. Bearing factor θ for ILT and MLT specimens at two mean stress levels.

CONCLUSION

The constant amplitude fatigue behavior of open-hole, filled-hole and load-transfer specimens fabricated using clad 2024-T3 aluminum has been investigated experimentally. The fatigue life data at two mean stress levels have been used to develop empirical factors β and θ which are used to predict the SSF or fatigue quality for the load-transfer specimens. The experimental data indicated a decrease in fatigue life for the ILT configuration compared to the MLT configuration. The load-transfer coupons failed in two characteristic failure modes, with net-section fracture dominating at higher stress levels and fretting induced fracture occurring at lower stress levels. The effects of load transfer on fatigue life observed in the present study conflict with those of Lee[3]. However, the differences in materials, fastener type, load levels and fatigue stress ratio used in the current study are significantly different than those reported in literature.

ACKNOWLEDGMENTS

The authors would like to acknowledge the useful inputs of Marvin Nuss and Mike Reyer of the FAA, Johnny Smith and Peter Gould, Hawker Beechcraft. The authors thank Prof. J.S. Tomblin, Executive Director, NIAR for providing the lab facilities and Ms. Lamia Salah, Manager, Fatigue & Fracture Labs, NIAR for efficient coordination/scheduling of the use of test equipment. The efforts of Kim-Leng Poon and K. Bangalore in machining the test specimens in a timely manner is appreciated.

REFERENCES

1. Barrios, W., "Stresses and Displacements due to Load Transfer by Fasteners in Structural Assemblies," *Engineering Fracture Mechanics*, 1978, Vol.10, pp. 115-176
2. Fatigue Rated Fastener Systems, AGARD Report No.721, 1985.
3. Lee, E. U., "Effect of Load Transfer on Fatigue of Mechanically Fastened Metallic Joints," *Fatigue in Mechanically Fastened Composite and Metallic Joints*, ASTM STP 927, John M. Potter, Ed., American Society for Testing and Materials, Philadelphia, 1986, pp. 95-117.
4. Starikov, R., "Fatigue behavior of mechanically fastened aluminum joints tested in spectrum loading," *International Journal of Fatigue* 26 (2004), pp.1115-1127.
5. Anderjaska, A. Nuert, and H. Leybolt, "Fatigue Evaluation of Wing and Associated Structures on Small Aircraft", Report No AFS-120-73-2, Federal Aviation Administration, May 1973.
6. "Fatigue and Fail-Safe Evaluation of Flight Structure and Pressurized Cabin for Part 23 Airplanes," FAA Advisory circular AC 23-13, April 1993.
7. Milton A. Miner, "Cumulative Damage in Fatigue", *Journal of Applied Mechanics*, September 1945.
8. Jarfall, L. E., "Optimum Design of Joints: The Stress Severity Factor Concept," The Aeronautical Research Institute of Sweden, 1967.
9. Metallic Materials Properties Development and Standardization (MMPDS)
10. Niu, M. C. Y., *Airframe Structural Design*, Conmilit Press LTD., 1993.
11. Standards Committee for HiLOK Products, 2600 Skypark Dr., Torrance, CA 90509
12. TableCurve® 2D, SYSTAT Software Inc., 501 Canal Boulevard, Suite F, Richmond, CA 94804-2028

Reduction of Jitter within Optical Tracking Data through an In-depth Survey of Smoothing Techniques

Schinnel Small, Kofi Nyarko and Craig Scott
Morgan State University

Copyright © 2008 SAE International

ABSTRACT

Signal Jitter is an impediment that continues to affect many technologies. Its presence in optical tracking devices is manifested in tracking data and affects the device's accuracy and latency. Previous studies have investigated several smoothing techniques to reduce jitter, while accounting for engine vibration as an external source. This paper will delve further into the research by exploring the characteristic behavior of the optical tracking data, so that a more intuitive selection of a smoother can be made. This process will not only justify the suitability of a smoothing technique for SVS, but also lead to increased tracking efficiency.

INTRODUCTION

Aircraft mishaps and fatalities remain a constant issue for general aviation pilots. The majority of these setbacks can be attributed to low visibility. In low visibility situations, such as night time or unpleasant weather, a pilot's perception of his or her environment, or situational awareness (SA), is drastically decreased. Synthetic Vision Systems (SVS) addresses this issue by offering 'enhanced' vision to the pilots. The enhanced vision is achieved by displaying three-dimensional representations of the outside environment, in spite of exterior conditions [1].

The successful implementation of an SVS is dependent on an accurate and reliable control mechanism. In this application, control is often provided by an optical tracking device. An assessment of the feasibility of a commercial-off-the-shelf (COTS) optical tracking device with SVS revealed that signal jitter was not only present, but also affected all six degrees of freedom (yaw, pitch roll, x, y, and z) for the device. These findings not only spurred the essential study of the extent of signal jitter effects but also the most appropriate means of reducing them [2]. Initial tasks involved the determination of signal jitter sources (both external and internal), as well as the utilization of common smoothing algorithms to reduce it. In some cases, effects from the attempted methods were visible and contributed to changes in latency and accuracy[3]. Although smoothing was achieved by each of the

attempted methods, a suitable trade off between jitter reduction and accuracy was not sufficiently attained.

This paper focuses on the determination of a suitable smoothing algorithm via the characterization of the tracking data obtained for this application. There has been reputable research that suggests the behavior of data sets and the desired criteria of the researcher are significant factors that can determine the selection of a smoothing algorithm [4]. By identifying the data's behavioral patterns, it is likely that a smoothing technique can be selected that reduces jitter effects and caters to the data's characteristics. This, in turn, can lead to improved performance for COTS optical tracking devices applied to SVS.

DATA AND SMOOTHER ANALYSIS

DATA CHARACTERIZATION. A key component in data analysis is the revelation of patterns within data. This process is necessary because data patterns allow meaningful observations and conclusions to be drawn about the variables that are under scrutiny. Smoothing is the suitable way to uncover data patterns as it removes the noisy components of data, and increases the visibility of any data pattern that may exist. If a pattern exists, it may be compared to traditional mathematical models, so that the data may be thoroughly classified and further understood [4].

There are three main classes of optical tracking data to consider within this study. The first class of data is obtained in near-stationary situations, where minute object movements may be observed. The second class consists of tracking data obtained while the object makes slow or gradual movements. The third data class is obtained when the tracking object makes sharp but intentional movements. A fourth class, which comprises data obtained with a stationary object, may also be considered. However, since the object is not moving, then it is expected that if the sampling frequency is large, the data obtained will adhere to linear properties, and so any smoother selected for smoothing should be sufficient for this type of data.