FAR23 Loads Professional Version 2003

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Abstract

The FAR23 Loads program was developed by Hal C. McMaster of Aero Science Software in 1989 and was updated in 1990, 1993 and 1995. McGettrick Structural Engineering, Inc. is the current owner of FAR23 Loads. This program provides a procedure for calculating the loads on an airplane according to CFR 14 FAR Part 23. The original version of the FAR23 Loads Program was written in Q-Basic. This new version of the program (FAR23 Loads Professional Version 2003) was developed at Wichita State University. It uses Visual Basic 6.0 and can be executed on the Windows Operating System version 95 or higher. This version is a single user-friendly integrated program that combines the original 20 individual modules.

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Executive Summary

The FAR23 Loads program was developed by Hal C. McMaster of Aero Science Software in 1989 and was updated in 1990, 1993 and 1995. McGettrick Structural Engineering, Inc. is the current owner of FAR23 Loads. This program provides a procedure for calculating the loads on an airplane according to the Code of Federal Regulations, Title 14 – Aeronautics and Space, Chapter I – Federal Aviation Administration, Subchapter C – Aircraft, Part 23 – Airworthiness Standards: Normal, Utility, Acrobatic and Commuter Category Airplanes, Subpart C – Structures. This is referred to as CFR 14 FAR Part 23. Most of the detailed flight loads are developed from the flight envelopes specified in CFR 14 sections 23.333 and 23.345. At every point specified in the flight envelope, the airplane is balanced by a tail load reacting to the specified linear normal acceleration and the aerodynamic lift, drag, and moment about the center of gravity. The data needed to make these balancing calculations consists of weight and center of gravity, aerodynamic surface geometry, structural speeds, and aerodynamic coefficients. Modules in the FAR23 Loads program develop these data. After these balancing loads data are developed, the critical structural loads are determined for each component. For the critical conditions, the air loads, inertia loads, and net loads are calculated. Aileron, flap, tab, engine mount, landing, and one engine out loads are also calculated. The landing loads are calculated from the landing gear geometry, landing load factor, weight, and center of gravity data.

The original version of the FAR23 Loads Program was written in Q-Basic. The program could be executed on MS DOS Operating System version 3.1 or higher. The program consisted of 20 different independent modules. When the program starts, the main menu appears. After selecting a module from the main menu, the input window for that module appears and parameters for the analysis are specified. The graphics programs FARPLOT and GEOMPLOT are stand-alone programs for plotting data and drawing airplane geometries. These programs read the output data from the FAR23 Loads programs and graph it using a variety of options. Users can customize the graphs for use in reports. For certain modules of the program, there is more than one input form, but there is no indication as to which form the user is currently on or what data have already been entered. If the analysis is done before entering all of the data, the user may get error messages and/or incorrect outputs.

Professional Version 2003 of the FAR23 Loads program was developed at Wichita State University using Visual Basic 6.0. This version can be executed on Windows Operating System version 95 or higher. It is a single integrated program that combines the 20 individual modules. The new program eliminates redundant data entry by making the data transfer process automatic, thus eliminating errors due to the reentry of data. Data is transferred from one module to another by the click of a button. The program also facilitates saving,

previewing and printing of all the inputs simultaneously. Although integrated, each module can be run independently by entering all required data into that module. This occurs without affecting the other modules. Professional Version 2003 of the FAR23 Loads program is a complete program that does not permit the user to run a module unless valid data for that particular module have been entered. A tool tip text provides information for each field that makes it convenient for the user to input valid data. Saving, previewing and printing output data for individual modules is also possible. The program provides additional utilities like a calculator, an interpolation module and a unit conversion facility. Help on how to use a particular module is also made available for the user to view and print. The overall user-friendliness and ease of use of the program have greatly improved over the original version.

1 Introduction

1.1 About FAR23 Loads

The FAR23 Loads program was developed by Aero Science Software to calculate the loads on an airplane using methods that are acceptable to the FAA. This new version of program developed at Wichita State University is an integrated package that uses consolidated data for the complete aircraft and performs calculations for all 20 modules independently.

Most of the detailed flight loads are developed from the flight envelopes specified in the federal requirements FARs 23.333 and 23.345. At every point specified in the flight envelope, the airplane is balanced by a tail load reacting to the specified linear normal acceleration and the aerodynamic lift, drag, and moment about the center of gravity. The data needed to make these balancing calculations consist of (1) weight and center of gravity, (2) aerodynamic surface geometry, (3) structural speeds, and (4) aerodynamic coefficients. These data are developed by modules in the FAR23 Loads program.

After the data needed to calculate the balancing loads are developed, the critical structural loads are determined for each component. For critical conditions, the air loads, inertia loads, and net loads are calculated. Aileron, flap, tab, engine mount, landing, and one engine out loads are also calculated.

Landing loads are calculated using landing gear geometry, landing load factor, weight, and center of gravity modules.

1.2 Federal Regulations

The program FAR23 Loads provides a procedure for calculating the loads on an airplane according to the Code of Federal Regulations, Title 14 – Aeronautics and Space, Chapter I – Federal Aviation Administration, Subchapter C – Aircraft, Part 23 – Airworthiness Standards: Normal, Utility, Acrobatic, and Commuter Category Airplanes, Subpart C – Structures. This is referred to as FAR Part23. The regulations through Amendment 42 have been included in the FAR23 Loads program.

1.3 Using this Report

This report is a guide to run the FAR23 Loads program and is intended to be a supplement to reference 1. Reference 1 provides the theoretical development of the equations used in the computer program.

Section 2 of this report describes how to install the program and run FAR23 Loads. The section also includes general information on the integrated module and its menu structure.

Sections 3 through 22 are each devoted to a separate module of the FAR23 Loads program and are organized according to the order that the modules appear in the main program menu. Each section has up to 3 subsections – description, running the module and output.

Table 1.1 is a summary of the modules in FAR23 Loads.

Table 1.1 Summary of Modules in the FAR23 Loads Program

Module Name	Section	Description
Weight Estimation	3	Estimates weight of airplane and its components
Weight and CG	4	Calculates Weight, center of gravity, and inertia
Envelope of Loads	5	Envelope of discretionary loading
Aerodynamic Surface Geometry	6	Calculates geometric properties of 16 aerodynamic surfaces
Structural Speed	7	Determines structural design speeds
Mach Limitations	8	Determines Mach limitations
Aero Coefficients	9	Calculates aerodynamic coefficients
Air Loads	10	Calculates wing air loads
Flight Loads	11	Calculates loads within the flight envelope
Select	12	Calculates the critical flight loads
Aileron Loads	13	Calculates loads on aileron

Flap Loads	14	Calculates loads on flaps
Wing Inertia	15	Calculates spanwise inertia loads for critical wing conditions
Net Loads	16	Calculates the net wing loads
Engine Mount Loads	17	Calculates engine mount loads
Landing Loads	18	Calculates landing loads
Landing Load Factor	19	Calculates landing load factor
Tail Loads Distribution	20	Calculates the chordwise distribution for tail loads
Tab Loads	21	Calculates tab loads
One Engine Out Loads	22	Calculates loads with one engine out

2 Getting Started System Requirements

2.1 Systems Requirements

The FAR23 Loads program is designed to run on a personal computer. The minimum system requirements are as follows:

- A Windows compatible personal computer
- MS-Windows 95 or higher
- A 44X (or better) CD-drive

2.2 FAR23 Loads Disk

The FAR23 Loads program is contained on one Compact Disk.

2.3 Installing the FAR23 Loads Program

The FAR23 Loads program must be installed using the following steps:

- Insert the CD into CD drive.
- From Windows Explorer, select the CD drive.
- Double click on the Setup.Exe file.
- Follow the on screen instructions to continue the installation.

2.4 Running the FAR23 Loads Program

To run the FAR23 Loads program, double click on the file with the name FAR23LOADS. When the program starts, a flash screen followed by the main window shown in figure 2.1 appears. The main menu window has 3 visible menu options and 1 invisible menu option. Details of visible menu options are as follows:

- File: Figure 2.2 shows the main menu window with File menu active. The File menu has 8 sub menus.
    - New – Creates a new file.
    - Open – Opens a previously saved file.
    - Save All Input – Saves inputs of all modules.

- Save All Input As – Saves existing active file with a different file name.
- Preview All Input – Enables user to preview all inputs. Input data for modules where data has not been entered will either appear blank or as a zero. When this menu option is selected, the fourth invisible menu “Input” becomes visible as shown in figure 2.12. Error Messages might pop up if modules are empty. Details of the input menu are given at the end of visible menu options.
- Plot Geometry – The Plot Geometry menu gets activated only after the file has been saved. It also becomes active if a previously saved file is opened. Messages pop up confirming the entry of data in relevant modules. If the user has not entered data in relevant modules, messages requesting entry of data in required modules appear. The Plot Geometry menu has 3 sub menus as shown in figure 2.3:

  1. Plot Wing Assembly – Plots the wing assembly. The wing geometry plot automatically includes the aileron geometry, aileron aft geometry, aileron forward geometry and flap geometry.
  2. Plot Horizontal Tail Assembly – Plots the horizontal tail assembly. The horizontal tail geometry plot automatically includes the elevator geometry, elevator tab geometry, elevator aft geometry, elevator forward geometry and horizontal stabilizer geometry.
  3. Plot Vertical Tail Assembly – Plots the vertical tail assembly. The vertical tail geometry plot automatically includes the rudder geometry, rudder aft geometry, rudder forward geometry and vertical stabilizer geometry.

- Close File – Enables the user to close an existing file. This menu will close the file but not terminate the program. The user can open another file or create a new file without exiting the program.
- Exit FAR23 Loads – Ends the FAR23 Loads program.
- Return to Main Menu – Enables user to close existing menu and return to the main menu from which other modules can be selected and executed. Note that this option is valid only for the individual modules that are described in Sections 3 through 22.

- Output: The Output menu is used to preview, save and print output and also to update input / output data to other modules. The Output menu has 5 sub menus. Details are as follows:
  - Preview Output – Enables user to preview output in the same window. Other options are enabled only after the preview output option is selected.
  - Print Output – Enables user to print output on paper. Errors might pop up if the printer is not installed or connected.

- Save Output as Notepad – Saves output data to notepad file (*.txt) format.
- Update Output To Other Modules – Updates output to respective modules. Refer to table 2.1 for details on data flow between modules. After the data is updated, a message box displaying the names of modules to which data has been updated appears.
- Close Output – Enables user to close the output menu and return to the main window. If the user has not updated data to other modules using the 'Update Data To Other Modules', selecting the 'Close Output' option results in a user prompt to determine whether the data needs to be updated to other modules.

- Utilities: Figure 2.4 shows the main menu window with Utilities menu active. The Utilities menu has 3 sub menus.
  - Calculator – Opens the calculator
  - Interpolation – Opens the interpolation window. This window helps the user interpolate a set of values. Figure 2.5 shows the interpolation window at startup. The window has three sections – Data Available, Data Required and Controls. In the Data Available section, X1 and X2 are the values for which corresponding data V1 and V2 are available. In the Data Required section, XR is the value for which an interpolated value is required. In the Controls section, Interpolate button interpolates the required value based on the available data, Reset button clears the window for fresh data entry and Close button closes the interpolation window.

    An example of using the Interpolation window is given below.

    Let, X1=10; X2=20; V1=100; V2=200 and XR=15.

    Here, interpolated value for XR=15 is required. After entering all the data, when interpolation button is hit, the window appears as shown in figure 2.6. The interpolated value appears in the textbox with a yellow background. If any input box is left empty, the program gives a message to enter the corresponding value. An illustrative message that appears when the user leaves the XR box empty is shown in figure 2.7.
  - Unit Conversion – Enables conversion of units from one system to another. The 'Uconeer' application has been downloaded from the internet and is being used in this program since it is available for free to all users. Details on how to use the 'Uconeer' application are available in its help files. Figure 2.8 shows the 'Uconeer' window.
- Help: Figure 2.9 shows the main menu window with Help menu active. The Help menu has 2 sub menus.
  - User Manual – Opens this report in .pdf format.

   - About – Details about the license and permission for use of the software are mentioned. Figure 2.10 shows the About window.
- Input: Figure 2.12 shows the main menu window with the input menu visible and active. The input menu has 3 sub menus:
   - Print Input – Prints input data on paper. Errors might pop up if printer is not installed or connected.
   - Save Input as Notepad – Saves input data to notepad file (*.txt) format.
   - Close Input – Closes the input menu and returns to main window with 3 visible menus.

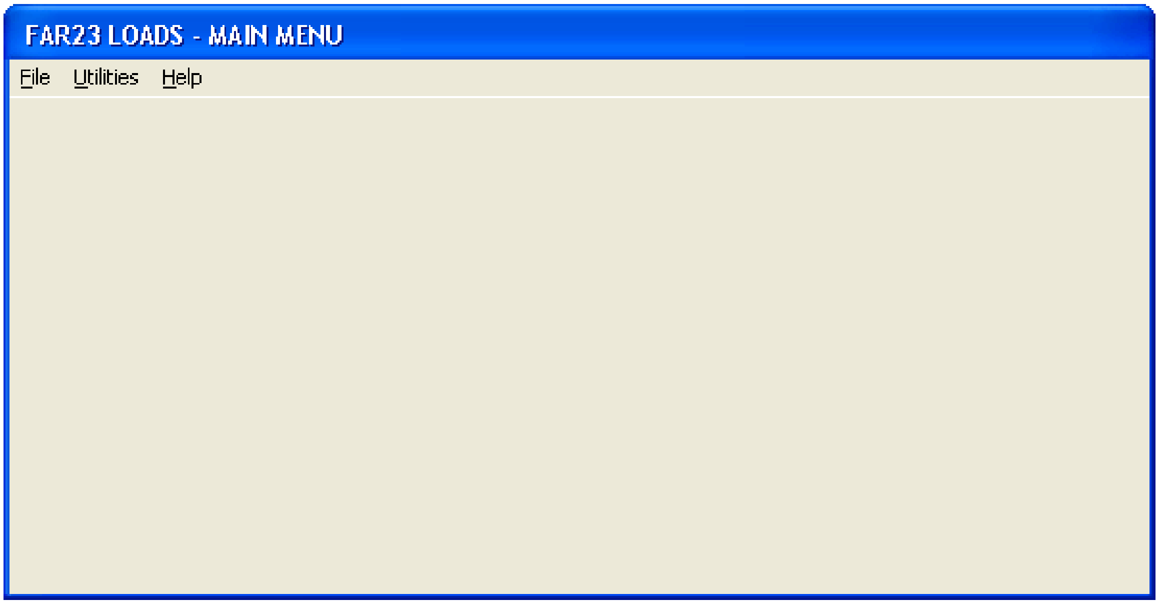


Figure 2.1 FAR23 LOADS Main Menu Window

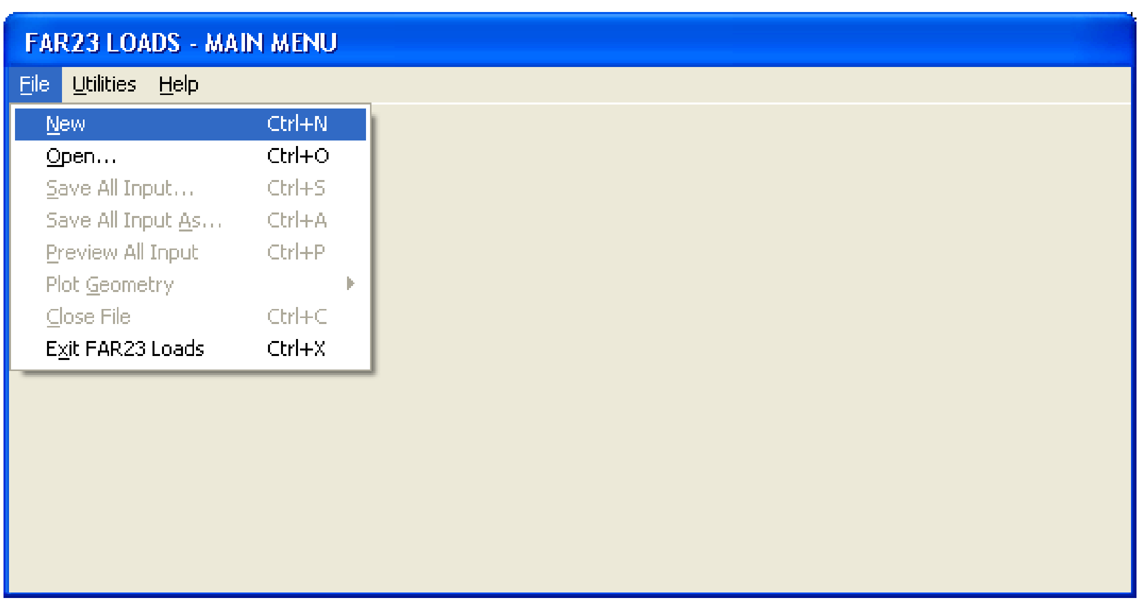


Figure 2.2 FAR23 LOADS Main Menu Window (File Menu Active)

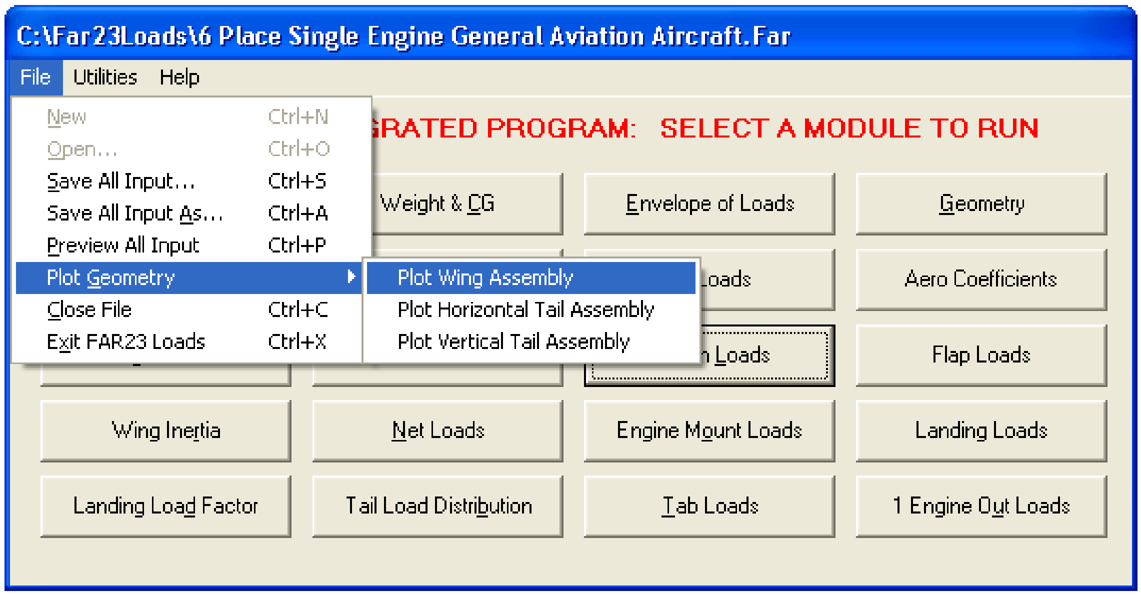


Figure 2.3 FAR23 LOADS Main Menu Window (Plot Geometry Menu Active)

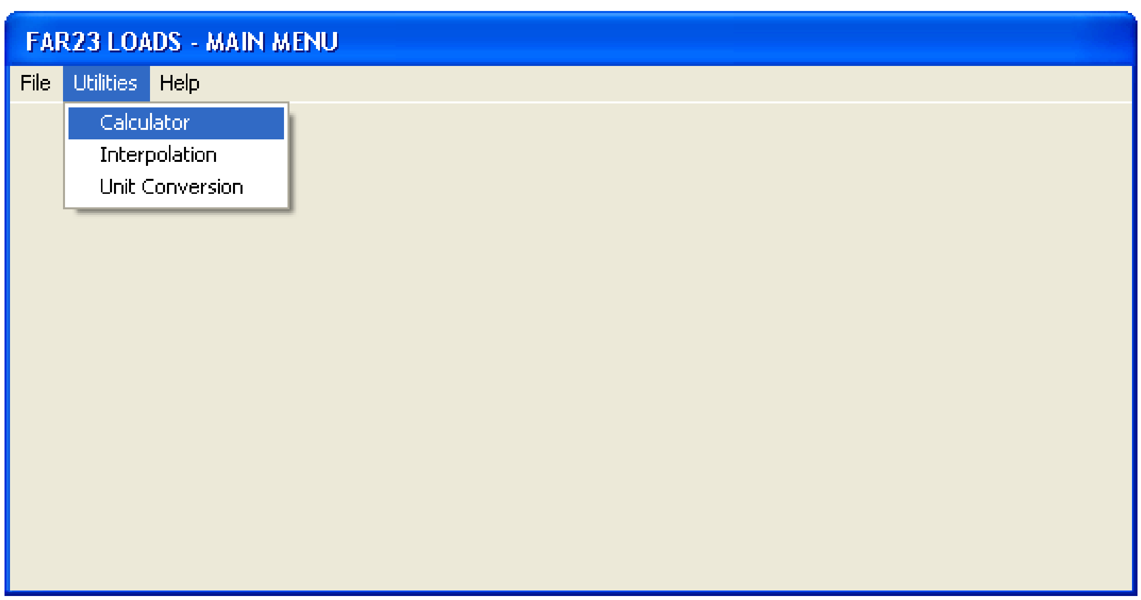


Figure 2.4 FAR23 LOADS Main Menu Window (Utilities Menu Active)

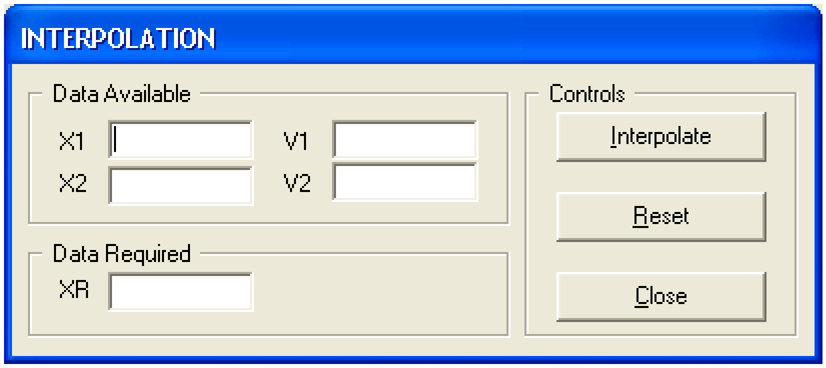


Figure 2.5 Utilities Menu (Interpolation Window]

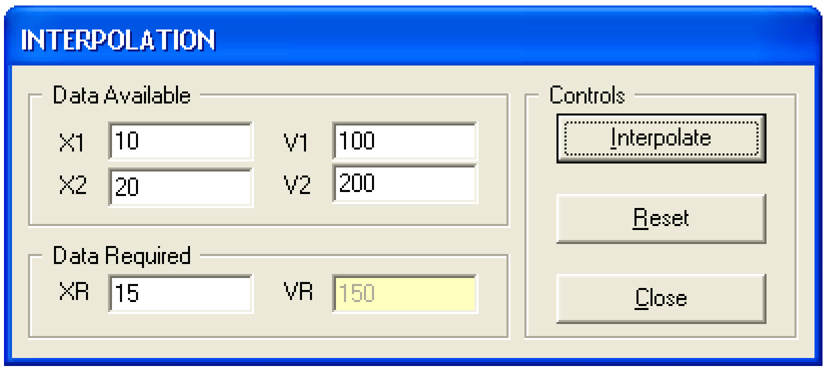


Figure 2.6 Interpolation Window after Interpolate button is hit

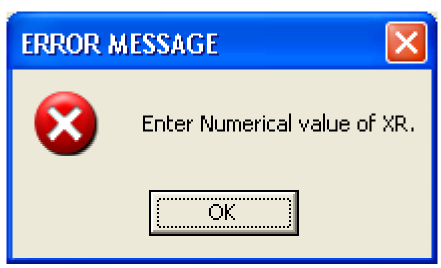


Figure 2.7 Error Message when XR box is left empty

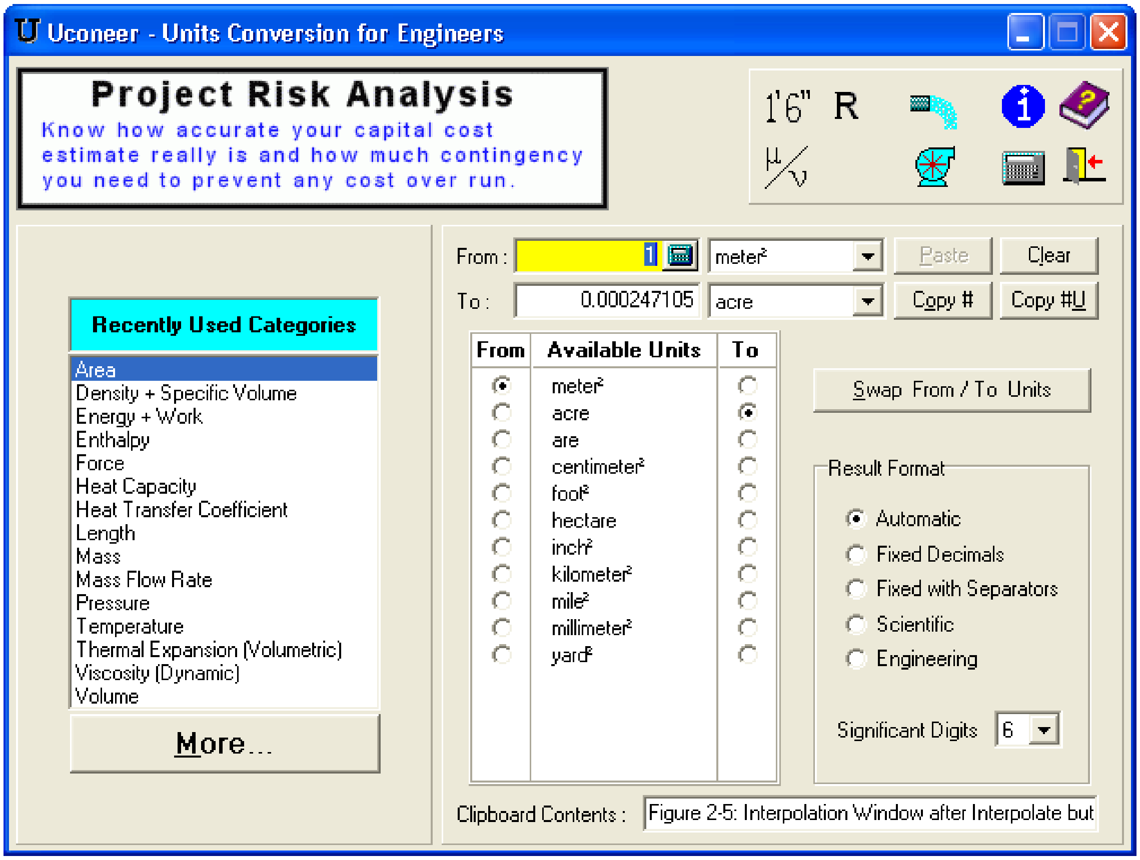


Figure 2.8 Unit Conversion Application 'Uconeer'

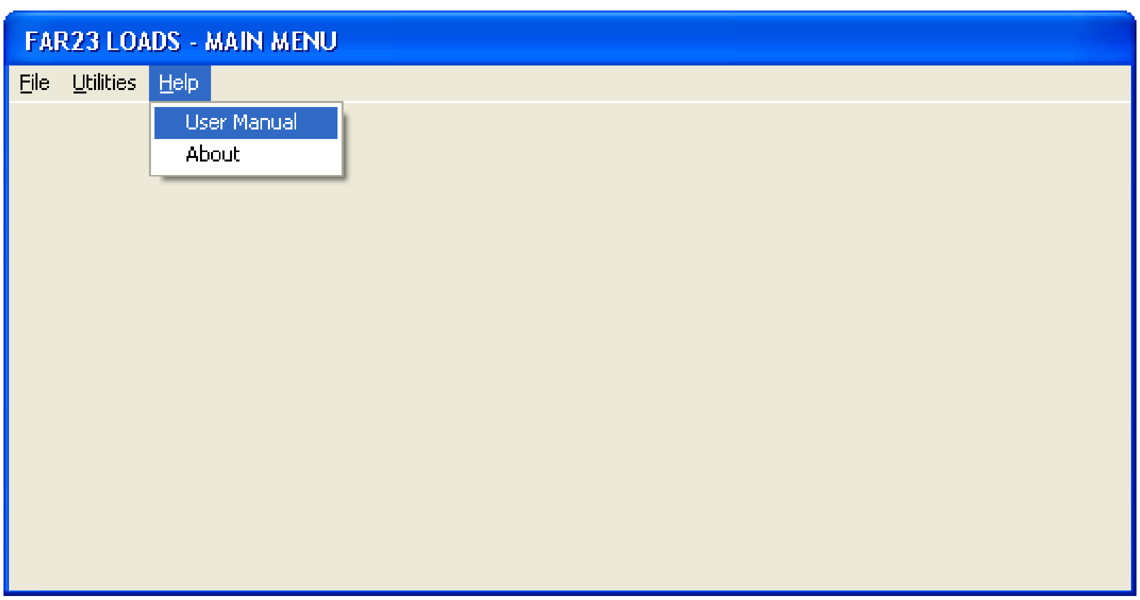


Figure 2.9 FAR23 LOADS Main Menu Window (Help Menu Active)

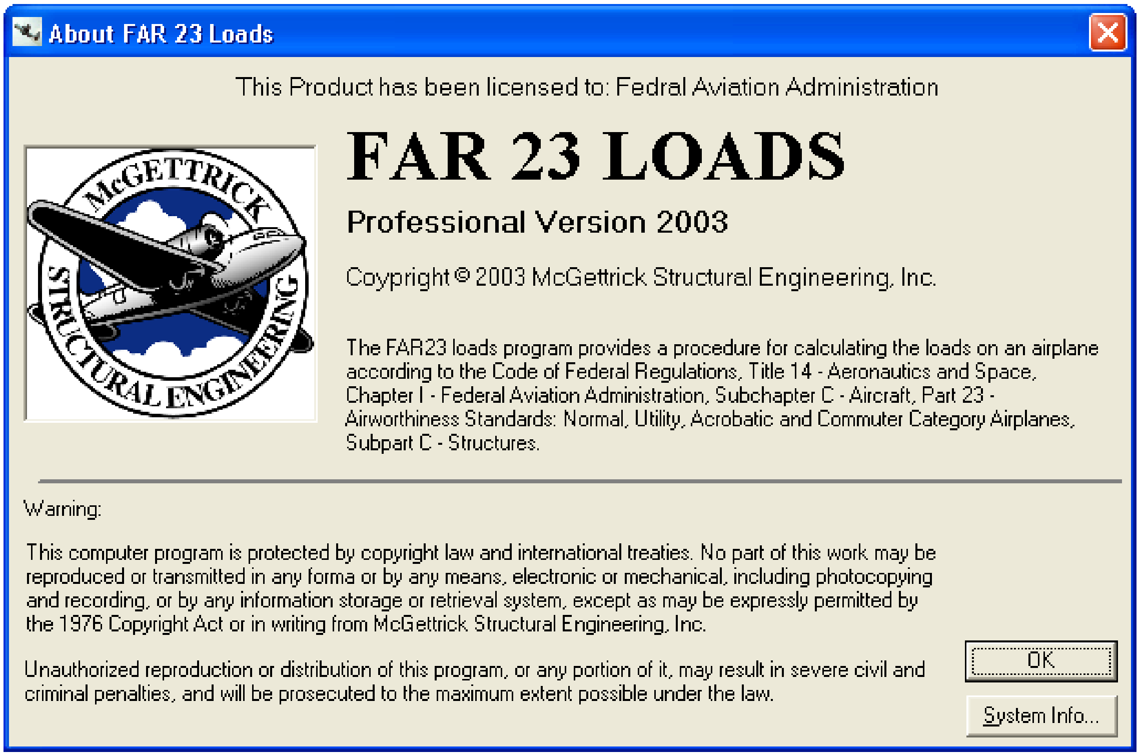


Figure 2.10 About Window

When the user selects New from the File menu, a window with all modules appears. Figure 2.11 shows the FAR23 Loads main menu with all modules. When Open is selected, the user is asked to select a file from the 'Open' dialogbox. A window as shown in figure 2.11 appears after an appropriate file is selected. If the file is not for the FAR23 Loads program, an error message will be given.

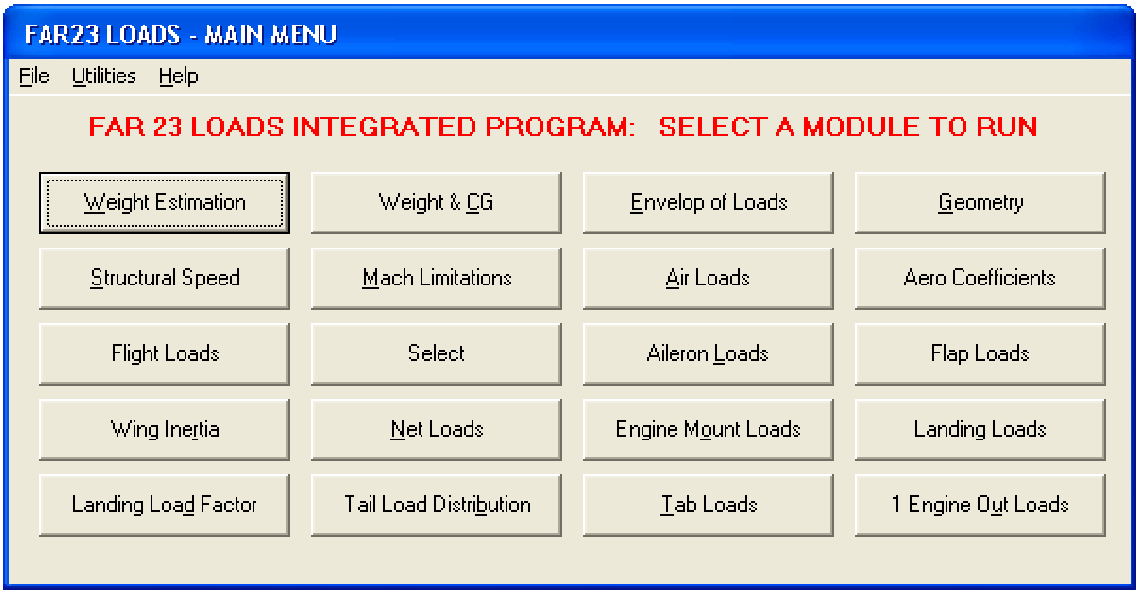


Figure 2.11 FAR23 Loads Main Menu (Module Names)

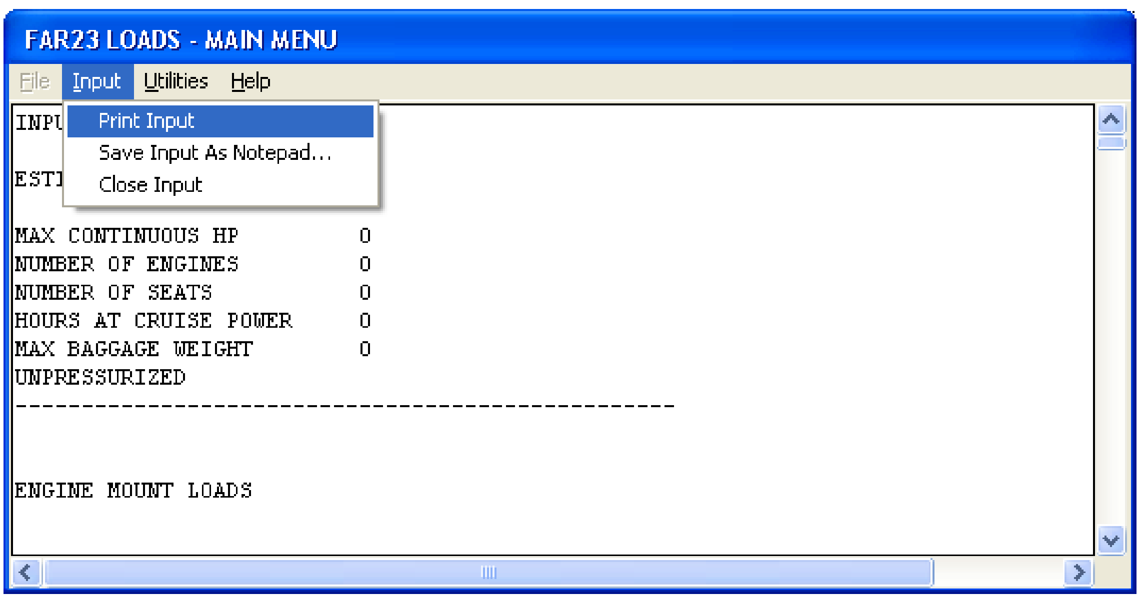


Figure 2.12 FAR23 LOADS Main Menu Window (Input Menu Active)

The modules shown in figure 2.11 are interconnected to each other. There are some modules which share common input data. For some modules the output data goes as input to another module. For a few cases both the input and the output data go to different modules. Table 2.1 shows the data that flows from one module to another in the FAR23 loads program.

Table 2.1 Data flow in the FAR23 Loads Program

From Module	To Module	Data Transferred
Weight & CG	Envelope of Load	Weight X Coordinate Y Coordinate Z Coordinate
Weight Estimation	Engine Mount Loads	Max. Engine HP
Landing Load Factor	Landing Loads	Landing Weight
Structural Speed	Landing Loads	Takeoff Weight
	Mach Limitations	M_C M_D Shoulder Altitude
	Select Critical Loads	Takeoff Weight Pos. N
	Flight Loads	Takeoff Weight Airplane Category V_A V_C V_D V_F M_D M_C
	Tab Loads	V_C
	Flap Loads	Takeoff Weight $V_{(Stall\ Clean)}$

		$V_{(Stall\ Flaps)}$ V_F
	Aileron Loads	V_A V_C V_D
Wing Geometry	Landing Load Factor	S_W
	Structural Speed	S_W
	Aerodynamic Coefficients	Leading Edge Points Trailing Edge Points DY
	Select Critical Loads	$2*S_W$ (Ft^2) MAC_W (Ft) AR_W
	Flight Loads	$2*S_W$ (Ft^2) MAC_W (Ft) $XLE_W + MAC_W * 0.25$
	Flap Loads	$2*S_W$ (Ft^2) MAC_F/ MAC_W
	Wing Inertia	Leading Edge Points Trailing Edge Points
Aileron Fwd Geometry	Aileron Loads	$S_{A(FWD)}$ (Ft^2)
Aileron Aft Geometry	Aileron Loads	$S_{A(AFT)}$ (Ft^2)
Flap Geometry	Flap Loads	S_F (Ft^2) MAC_F/ MAC_W
Vertical Tail Geometry	Select Critical Loads	S_{VT} (Ft^2)

		MAC_{VT}
		$XLE_{VT} + MAC_{VT} * 0.25$
		AR_{VT}
	Tail Load Distribution	S_{VT} (Ft^2)
	1 Engine Out Loads	S_{VT} (In^2)
		AR_{VT}
		$XLE_{VT} + MAC_{VT} * 0.25$
		$XLE_{VT} + MAC_{VT} * 0.50$
Rudder Geometry	Tail Load Distribution	S_R (In^2)
	Select Critical Loads	S_R (Ft^2)
	1 Engine Out Loads	S_R (In^2)
Rudder Aft Geometry	Tail Load Distribution	$S_{R(AFT)}$ (Ft^2)
	Select Critical Loads	$S_{R(AFT)}$ (Ft^2)
Rudder Fwd Geometry	Tail Load Distribution	$S_{R(FWD)}$ (Ft^2)
	Select Critical Loads	$S_{R(FWD)}$ (Ft^2)
Horizontal Tail Geometry	Select Critical Loads	$2*S_{HT}$ (Ft^2)
		AR_{HT}
		$XLE_{HT} + MAC_{HT} * 0.50$
		$XLE_{HT} + MAC_{HT} * 0.25$
		S_E/S_{HT}
	Flight Loads	$XLE_{HT} + MAC_{HT} * 0.05$
		$XLE_{HT} + MAC_{HT} * 0.25$
	Tail Load Distribution	S_{HT} (In^2)
Elevator Geometry	Select Critical Loads	S_E (Ft^2)
Elevator Fwd	Tail Load Distribution	$S_{E(FWD)}$ (In^2)

Geometry	Select Critical Loads	$S_{E(FWD)}$ (Ft^2)
Elevator Aft Geometry	Tail Load Distribution	$S_{E(AFT)}$ (In^2)
	Select Critical Loads	$S_{E(AFT)}$ (Ft^2)
Elevator Tab Geometry	Tab Loads	$S_{E(TAB)}$ (In^2)
Aero Coefficients	Air Loads	Cruise, Landing & Enroute Data
Flight Loads	Select Critical Loads	V-n Data File
Air Loads	Net Loads	Air Loads File
Wing Inertia	Net Loads	Wing Inertia File

3 Weight Estimation

3.1 Weight Estimation Description

There are three weight estimation modules in the FAR23 Loads program. The first module 'Weight Estimation' estimates the weight of the airplane and its major components. The other two modules, 'Weight & CG' and 'Envelope of Loads', are discussed in sections 4 and 5 respectively.

To estimate the weight of the airplane and its components, the following information is required:

- Number of engines
- Total horse power
- Type of engine
  - 4-cycle reciprocal
  - 2-cycle reciprocal
  - turbocharged
  - turboprop
  - liquid-cooled
- Hours of endurance at cruise speed
- Number of seats
- Total baggage weight
- Whether the cabin is pressurized or not

Based on this information, the weight of different components and the total aircraft weight are calculated.

3.2 Running the Weight Estimation Module

To run 'Weight Estimation', click the module from the main window. The main input window appears as shown in figure 3.1.

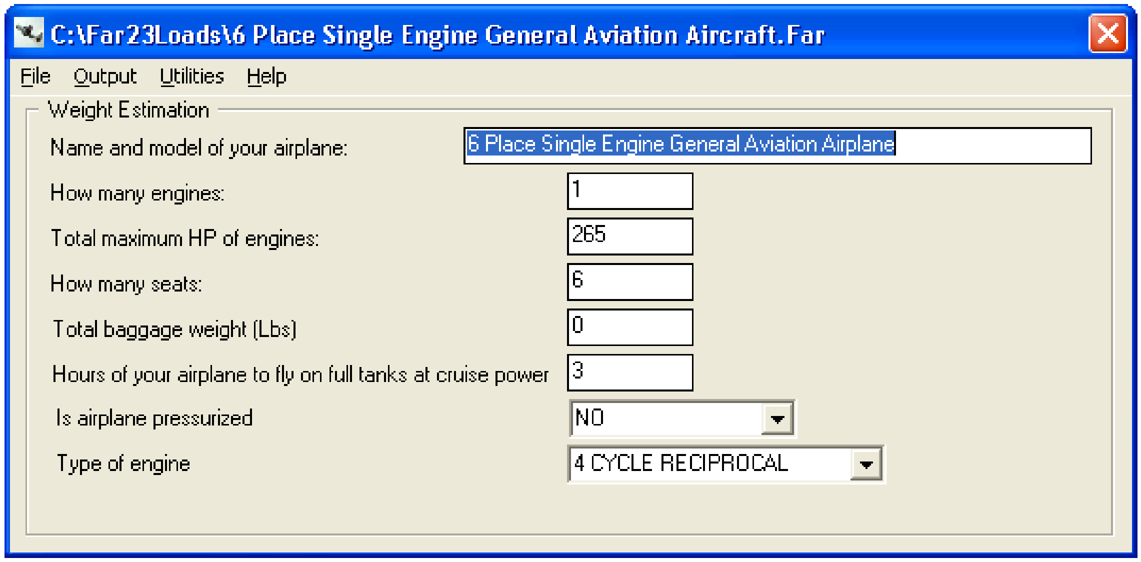


Figure 3.1 Weight Estimation Input Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the weight estimation module.

3.3 Weight Estimation Output

The output from the 'Weight Estimation' module consists of the maximum take-off weight and weights of the major components of the airplane. The components are grouped into structure, powerplant and systems.

The results of this module are used in the 'Weight & CG' and 'Envelope of Loads' modules to determine the envelope of useful loadings and to calculate weight, center of gravity and inertia of the airplane.

4 Weight & CG

4.1 Weight & CG Description

Weight & CG calculates the weight, center of gravity and inertia of the airplane for any specific loading configuration. These calculations are done at the four CG locations defining the weight structural limits diagram. These CG locations are the aft gross weight, the forward gross weight, the most forward reduced weight and the minimum weight. On an airplane with retractable landing gear, it is usually necessary to account for the shift in CG due to retraction of the gear with a second set of four loading conditions.

The weight limits are defined in FAR 23.25. The maximum weight must not be less than the empty weight plus 170 pounds for each seat for normal and commuter categories (or 190 pounds for utility and acrobatic category airplanes) plus oil at full capacity and a half hour of fuel at maximum continuous power. Also, the maximum weight must not be less than the empty weight plus minimum crew and full tank capacities of fuel and oil.

The minimum weight is not more than the empty weight (including unusable fuel, full oil, and fluids) plus the minimum crew (usually the pilot) and a half hour of fuel at maximum continuous power. For turbojet powered airplanes, the required fuel is 5 percent of the total fuel capacity.

The location of the weight components can be established from the three-view drawing or inboard profile drawing. The weight can be obtained from the component manufacturer, from actual weighing of the part, by calculation from drawing dimensions and material density, or from the 'Weight Estimation' module.

The inputs required are the coordinates, weight and moment of inertia for each of the components of the airplane, and useful loads for the condition. The inertia of small components may be neglected with no appreciable effect on the total inertia of the airplane.

The component data is entered in the weight database. This is the same database used by the 'Envelope of Loads' module.

The weight database contains the component weight data and the location dimensions. The default value for the maximum number of components in the database is 100, although the user may change this value. The database is divided into 3 sections: empty weight items, minimum weight items and discretionary items. Fifty percent of the items are considered empty weight items, ten percent are minimum weight items and forty percent are discretionary items. The type of item is indicated by the item number. Using the default value of 100 items results in the following: items 1 through 50 are empty weight items, items 51 through 60 are minimum weight items and items 61 through 100 are discretionary items. When entering data into the database, it must be entered at the proper item number.

4.2 Running the Weight & CG Module

To run 'Weight & CG' module, click the module from the main window. The main input window appears as shown in figure 4.1.

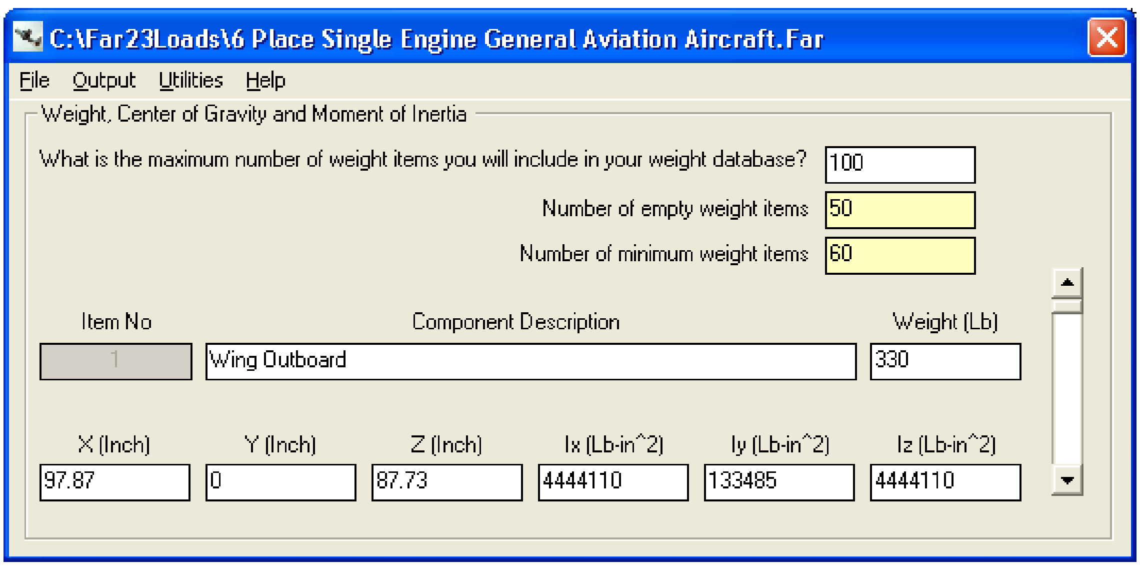


Figure 4.1 Weight & CG Input Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. When the number of weight items to be considered in the database is left empty, the default value of 100 is selected. The number of empty and minimum weight items is calculated based on the total items and displayed in the respective text boxes with a yellow background. The user does not have the flexibility to change these calculated values. Item numbers appear in the text box in grey color. Use the scrollbar to navigate through the item numbers. It is recommended to use the top and bottom arrows of the scrollbar rather than directly using the scroll tab to navigate through the item numbers since errors can occur due to scrolling with the tab.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the weight and CG module.

4.3 Weight & CG Output

The Weight & CG output includes the weight, center of gravity, the inertias with respect to airplane coordinates, the inertias with respect to principle axes and the angle θ for a single load configuration. Inertias are given in both sludg-ft^2 and lbs-in^2. The airplane weight and center of gravity are needed for the calculation of balanced flight and landing conditions in modules ‘Flight Loads’ and ‘Landing Loads’. The airplane inertia is needed for the calculation of maneuver and gust flight conditions and unbalanced landing conditions in modules ‘Select’ and ‘1 Engine Out Loads’.

5 Envelope of Loading Conditions

5.1 Weight Envelope Description

This module calculates the envelope of discretionary useful loading. It uses the same database as the 'Weight & CG' module. The database contains the component weight data and location dimensions. This weight data comes from the 'Weight Estimation' module or actual known weights; the location dimensions come from the three-view drawing.

This module directly receives data from the 'Weight & CG' module when the 'Update Data To Other Modules' menu option is selected from the 'Output' menu in 'Weight & CG' module. Although data is received automatically, this module is flexible enough to create or modify the database or add, delete, edit or move the component data. The minimum flight weight is calculated and the envelope enclosing all possible loadings is calculated. From the envelope plot of useful loadings, the four structural limit points can be selected to include the most desirable and practical loadings. See section 4 for additional information on the weight database.

5.2 Running the Weight Envelope Module

To run the 'Envelope of Loads' module, click the module from the main window. The main input window appears as shown in figure 5.1. The values in textboxes with yellow background are updated automatically from respective modules.

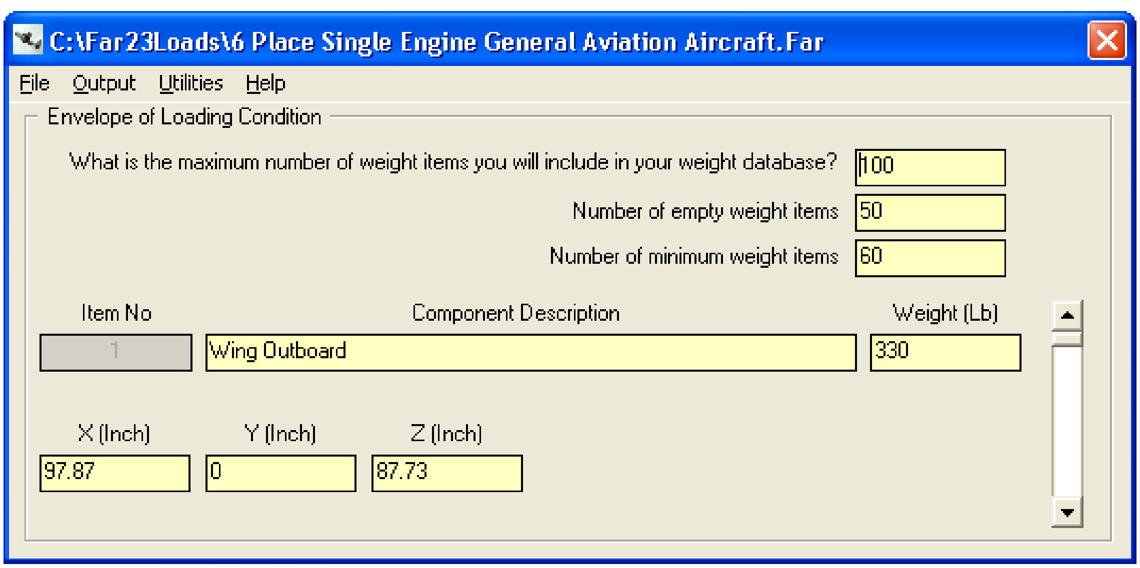


Figure 5.1 Envelope of Loads Input Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the weight & CG module.

5.3 Weight Envelope Output

The output from this module is the envelope of useful loads and includes the weight and CG of the airplane for given loading conditions. This data is used in the ‘Flight Loads’ module.

6 Aerodynamic Surface Geometry

6.1 Surface Geometry Description

The geometric properties for all aerodynamic surfaces on the airplane are calculated by the ‘Geometry’ module. The aerodynamic surfaces include the wing, aileron, aileron aft, aileron forward, flap, rudder, rudder aft, rudder forward, elevator, elevator aft, elevator forward, elevator tab, horizontal tail, horizontal stabilizer, vertical tail and vertical stabilizer.

This module must be used to analyze each aerodynamic surface. For each surface, the user enters the coordinates to define the leading and trailing edges of the surface. The program divides the surface into elements and calculates the area of each element.

The inputs required for this program includes the coordinates of the leading and trailing edge of the aerodynamic surface. Two points define a straight leading or trailing edge. Three points can be used to define the leading edge of a wing with a leading edge extension at the inboard end of the wing. Three points can also be used to define a straight leading edge with a raked tip. A complex or curved leading or trailing edge can be defined by a series of points assuming that the points are connected by short straight lines.

6.2 Running the Geometry Module

To run the ‘Geometry’ module, click the module from the main window. The main input window appears as shown in figure 6.1. This window displays all the aerodynamic surfaces. Select one surface at a time and enter the data required forr that surface. Since the majority of the input and output data from these modules goes to other modules, it is highly recommended that the user update the data to other modules. By doing so, errors due to reentry of redundant data are eliminated.

When the user selects a surface for data entry, an input window as shown in figure 6.2 appears. The inputs required are the name of the surface, whether the surface is symmetrical about the x-axis (y=0), the number of increments to divide the surface into, and the coordinates for the leading and trailing edge of the surface. For coordinates, x-value is the fuselage station and y-value is the wing station. Data should be entered starting from the lowest fuselage station since data in the same order will be used to plot the geometry. If the entered data is not in the proper order, absurd geometries will appear.

To enter the leading edge points, first enter the number of points. Move to the area for the x and y coordinates. For the first points, enter the x-value, then the y-value. Then use the mouse to click on the scroll bar next to the y-coordinate field. The point number should advance to the next number. Then enter the next pair of x-y coordinates. Continue until all data are entered. If an error occurs corrections can be made by clicking at the top of the bar. It is highly recommended that the scroll bar be positioned so that the first x-y coordinate pair is visible. Data for the trailing edge are entered in the same way.

Finally, click the check-box to preview or print the element data points.

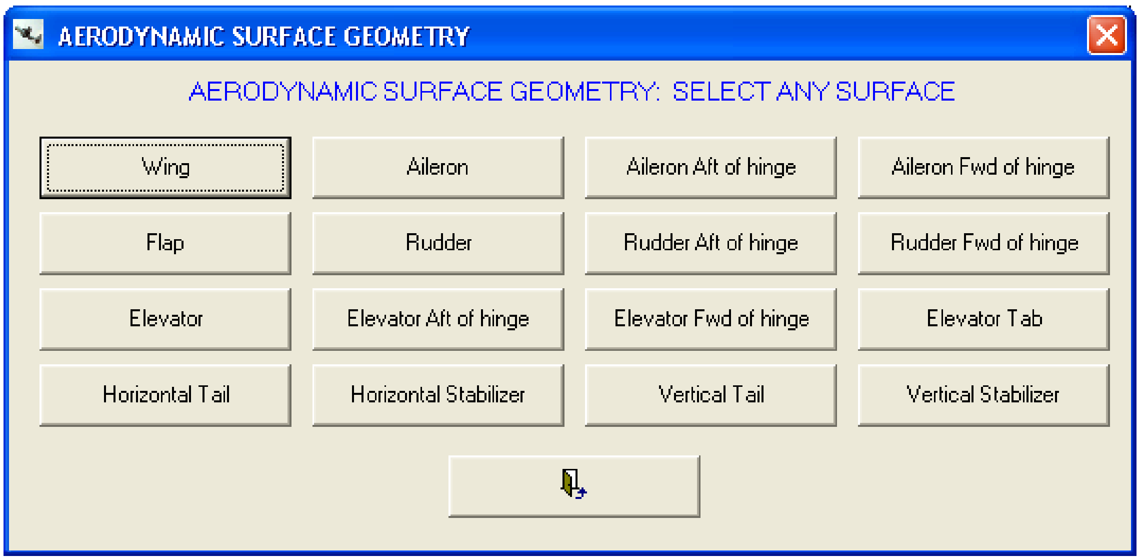


Figure 6.1 Aerodynamic Surface Geometry Main Window

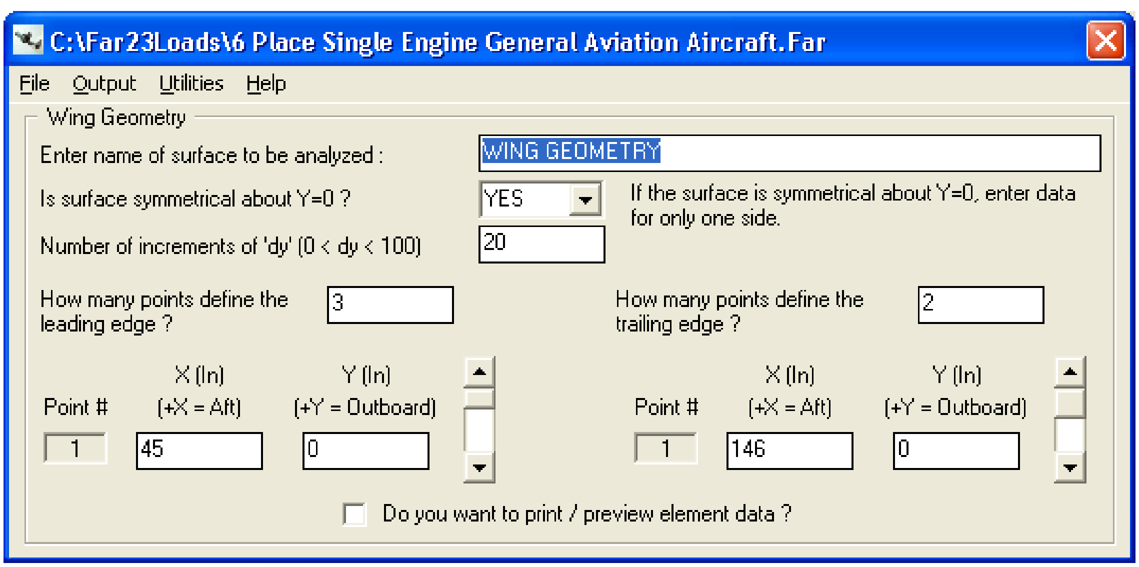


Figure 6.2 Wing Surface Geometry (Sub Module of Geometry Module)

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Output:
    - Plot Geometry – Plots the geometry on an X-Y axis. This menu gets activated only after the 'Preview Output' menu is run. Geometry of the surface is plotted and can be saved as a *.bmp file and opened and modified or printed from 'Paint' or any other application that opens graphic pictures.
- Help: The Help menu has only one option. It provides help on how to use the geometry module.

6.3 Geometry Module Output

The 'Geometry' module calculates the area, aspect ratio, mean aerodynamic cord (MAC) and the butt line and fuselage station of the leading edge of the MAC. This output data is needed as input to the modules 'Structural Speed', 'Air Loads', 'Flight Loads', 'Select', and '1 Engine Out Loads'.

7 Structural Speed

7.1 Structural Speed Description

The structural design speeds and the maneuvering load factors are calculated by the module ‘Structural Speed’. This module lets the user choose the airplane certification category: normal, utility, or acrobatic. The module calculates the minimum structural design speeds / load factors and verifies that the chosen structural design speeds are greater than the minimum requirements. It also checks that the margins between the speeds are greater than the requirements. If necessary, the speeds are adjusted to meet the requirements relative to cruise speed.

7.2 Running the Structural Speed Module

To run the ‘Structural Speed’ module, click the module from the main window. The main input window appears as shown in figure 7.1. The values in textboxes with yellow background are updated automatically from respective modules.

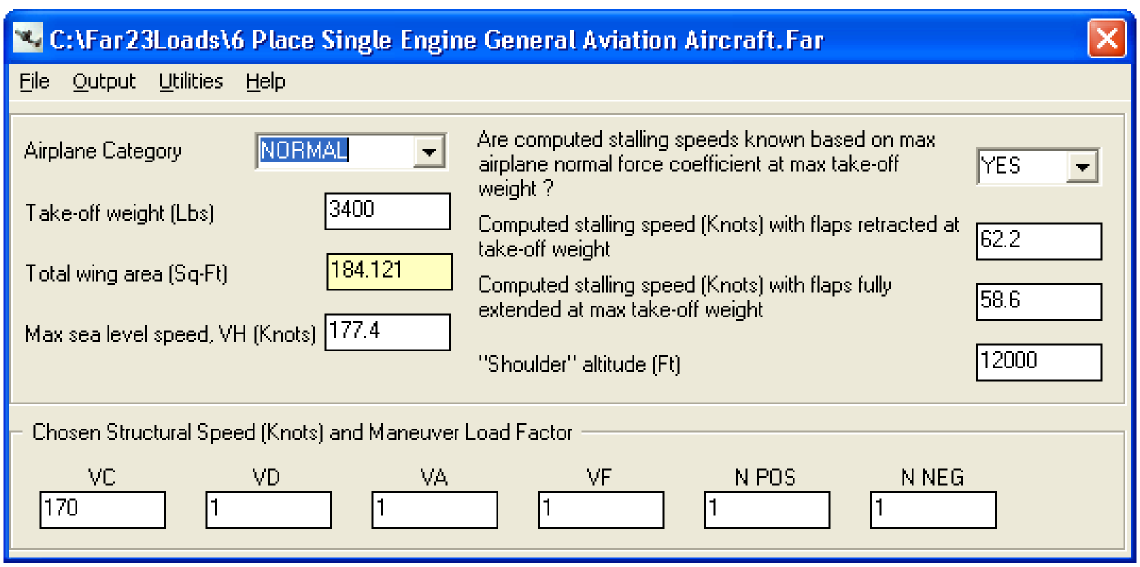


Figure 7.1 Structural Speed Main Window

The input required for this module includes the category of airplane, the take-off weight, wing area, maximum speed at sea level, and the shoulder altitude.

The stalling speeds with flaps extended and with flaps retracted can be entered if known, or they can be calculated by ‘Structural Speed’. For a known stalling speeds, select ‘YES’ and enter the values. To calculate the stalling speeds, select ‘NO’ and enter the maximum lift coefficients as shown in figure 7.2.

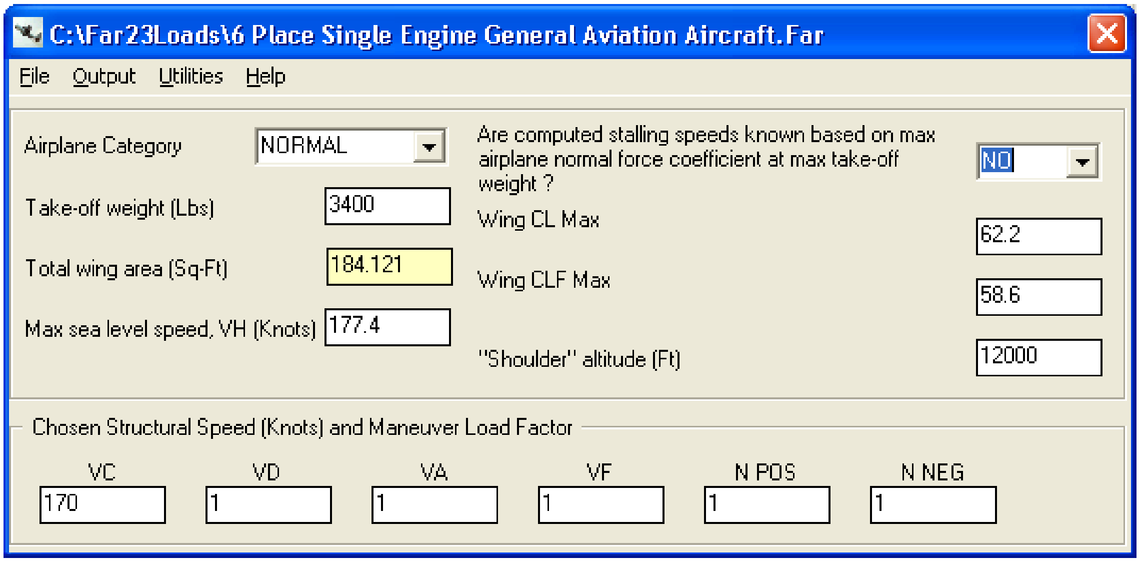


Figure 7.2 Structural Speed Main Window (Stalling Speeds Unknown)

The design speeds and load factors can be entered or calculated. Enter 1 for any calculated value. Some of the design speeds can be entered, and the program can calculate the remaining values.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the structural speed module.

7.3 Structural Speed Output

This module calculates the design speeds, load factors and Mach numbers (M_C, M_D). The minimum values are calculated for V_C, V_D, V_A and V_F, and for the positive and negative load factors. If a design speed is chosen then the other speeds are adjusted to meet the requirements. The calculated values are used in the ‘Mach Limitations’, ‘Flight Loads’, ‘Aileron Loads’, and ‘Flap Loads’ modules.

8 Mach Limitations

8.1 Mach Limitations Description

The 'Mach Limitations' module determines the Mach limitations for the flight envelope diagram. For a constant Mach number, the equivalent air speed is calculated at altitudes from the shoulder altitude to the maximum operating altitude.

8.2 Running the Mach Limitations Module

To run the 'Mach Limitations' module, click the module from the main window. The main input window appears as shown in figure 8.1. The values in textboxes with yellow background are updated automatically from respective modules.

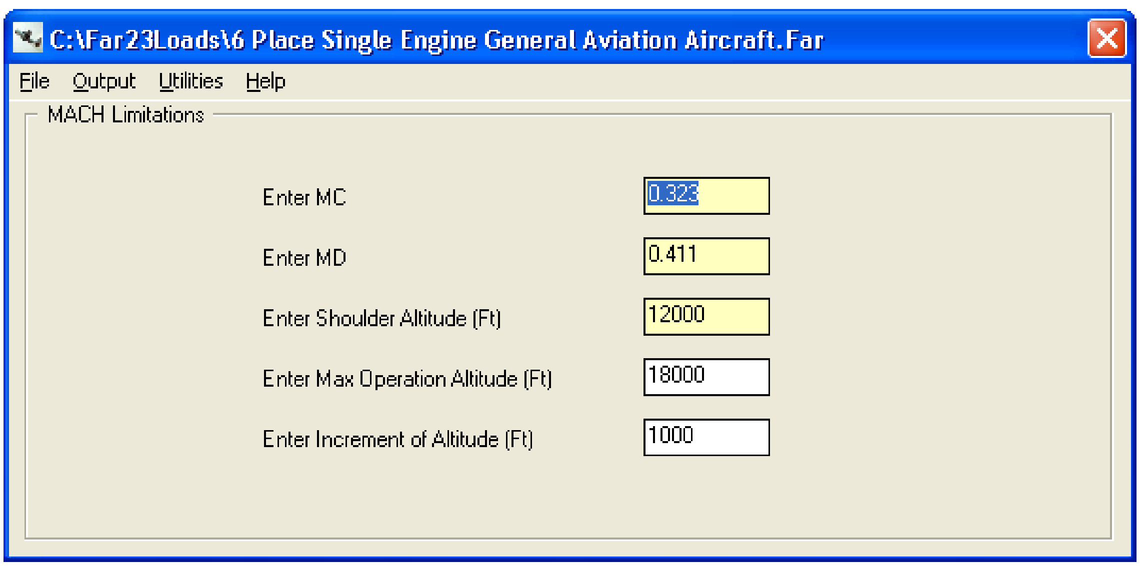


Figure 8.1 Mach Limitations Main Window

The input for this module includes the Mach number at design cruising speed (MC), the Mach number at design dive speed (MD), the shoulder altitude, and the maximum operating altitude. An altitude increment is also required; this is used to determine the altitudes of interest between the shoulder and operating altitudes. All altitudes are entered in feet.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the mach limitations module.

8.3 Mach Limitations Output

The output from 'Mach Limitations' includes the air speeds at altitudes between the shoulder altitude and the operating altitude. This data can be plotted on the flight limits diagram. All speeds are given in knots equivalent air speed (KEAS), and the altitudes are given in feet.

9 Aerodynamic Coefficients

9.1 Aerodynamic Coefficients Description

The aerodynamic coefficients module is used to calculate the aerodynamic coefficients for cruise, landing and enroute conditions. This module calculates the basic and additive spanwise aerodynamic lift coefficient distributions for the wing. It combines these with the spanwise lift coefficient distribution for any specific total wing lift coefficient and then calculates the associated spanwise drag and moment coefficients for that wing C_L.

The aerodynamic coefficients module also calculates the stall lift coefficient and the angle of attack for the wing. The pitching moment coefficient of the fuselage and nacelle is calculated and added to the total wing moment coefficient to provide lift, drag and moment coefficients for the airplane-less-tail configuration for any C_L. The drag moment of the extended landing gear is calculated and added to the airplane-less-tail. The sea level equations for lift, drag, and moment are formulated. These equations are used to make the balancing calculations for the V-n diagrams in the 'Flight Loads' module.

9.2 Running the Aerodynamic Coefficient Module

To run the 'Aerodynamic Coefficient' module, click the module from the main window. The main input window appears as shown in figure 9.1. The first window allows the selection of the cruise, landing or enroute submodules. It also provides the flexibility to return to the main menu by clicking on the close button. Additional data input windows for Cruise, Landing and Enroute conditions are shown in figures 9-2, 9-3 and 9-4 respectively. The values in textboxes with yellow background are updated automatically from respective modules.

Since this module has been customized for aero coefficients only, the user is not asked if calculations for air loads are desired.

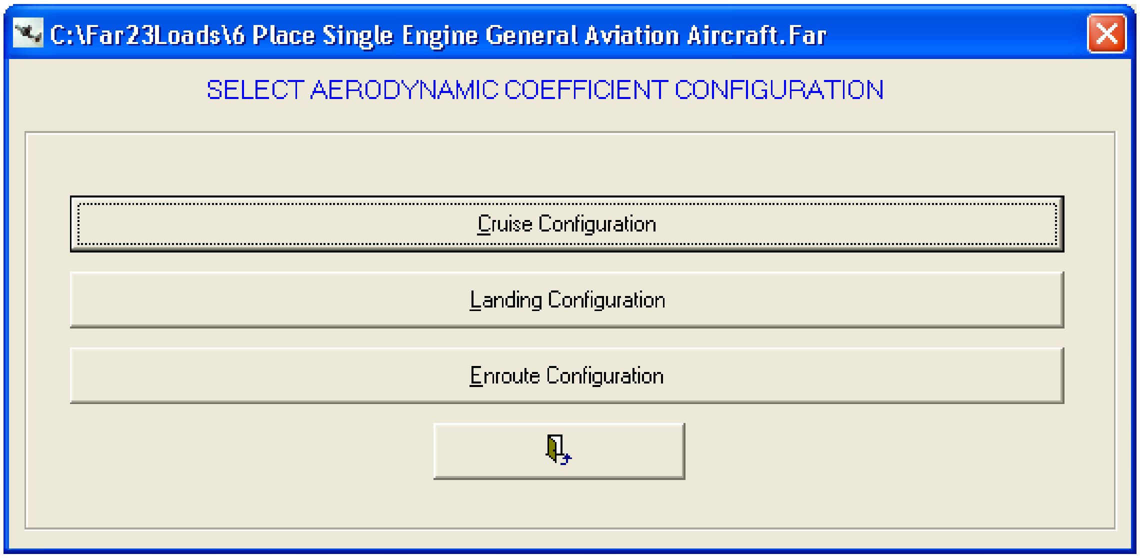


Figure 9.1 Aerodynamic Coefficients Main Window

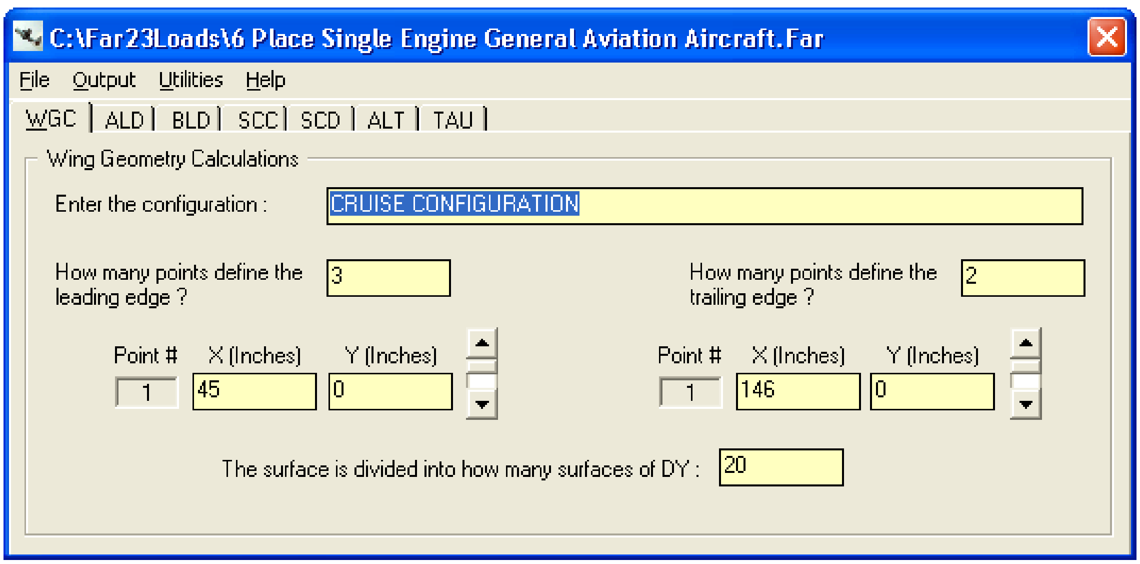


Figure 9.2 Aerodynamic Coefficients: Cruise Configuration Window

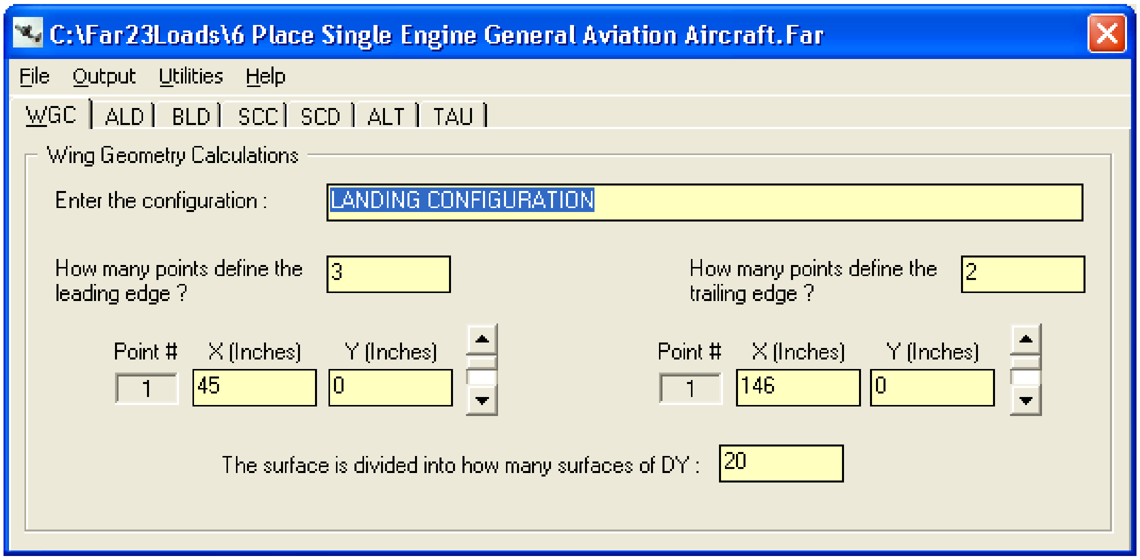


Figure 9.3 Aerodynamic Coefficients: Landing Configuration Window

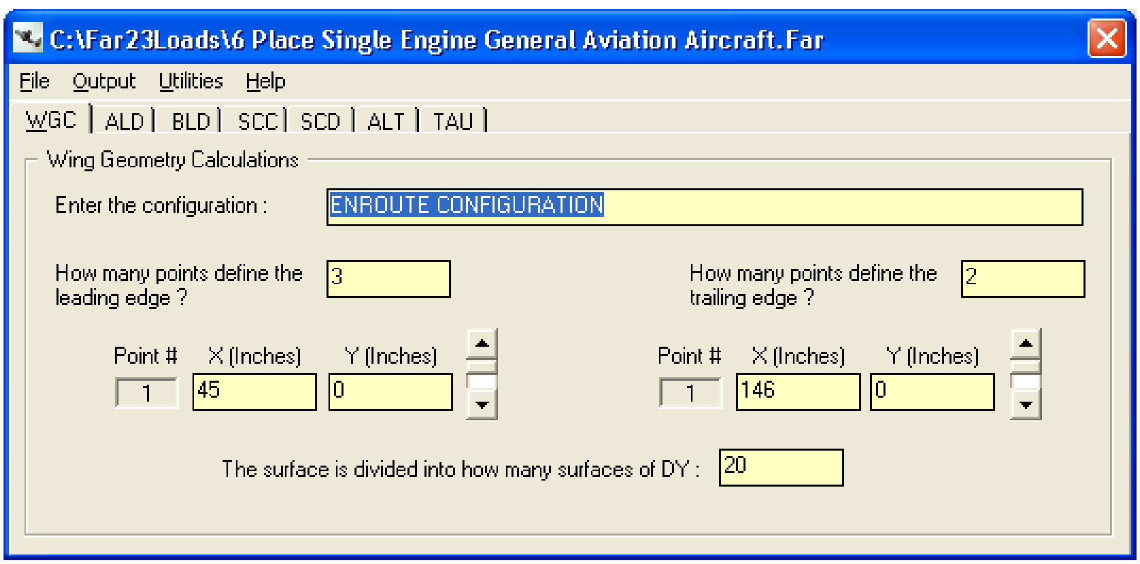


Figure 9.4 Aerodynamic Coefficients: Enroute Configuration Window

The data input window for any case has a similar appearance and menu structure. Each window has 7 data entry pages:

1. WGC: Wing Geometry Calculations
2. ALD: Additive Lift Distribution
3. BLD: Basic Lift Distribution
4. SCC: Stall C_L Calculations

5. SCD: Spanwise Coefficient Distribution
6. ALT: Airplane Less Tail Load Calculations
7. TAU: Correction factor Tau

In the WGC page, geometry data for the wing is entered including the coordinates for the leading and trailing edges. The window appears as shown in figure 9.2.

In the ALD page, parameters required for the additive lift distribution calculations are entered. The input window appears as shown in figure 9.5.

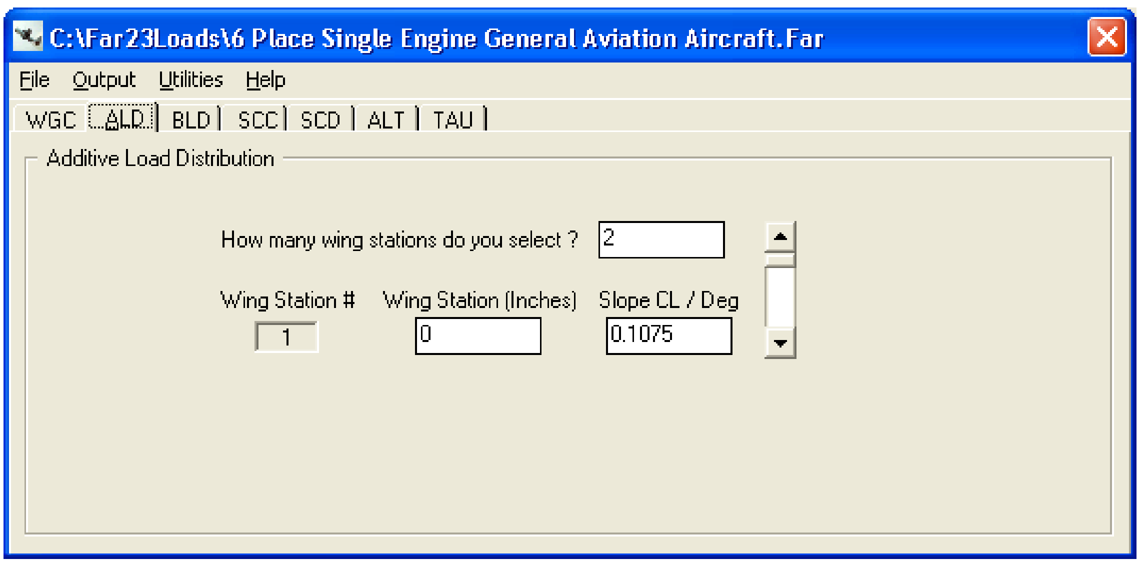


Figure 9.5 Aerodynamic Coefficients: Additive Lift Distribution Window

In the BLD page, parameters required for the basic lift distribution calculations are entered. The input window appears as shown in figure 9.6.

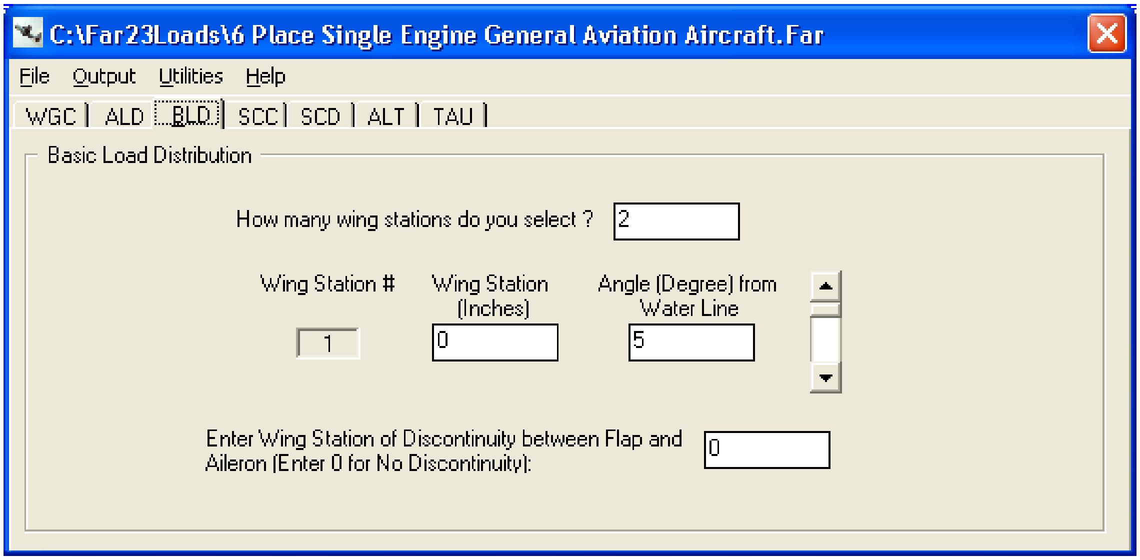


Figure 9.6 Aerodynamic Coefficients: Basic Lift Distribution Window

In the SCC page, parameters required for the spanwise C_L calculations are entered. The input window appears as shown in figure 9.6. The user is asked if stall C_L is to be calculated. If the user selects YES, a window appears as shown in figure 9.7, but if the user selects NO, a window appears as shown in figure 9.8.

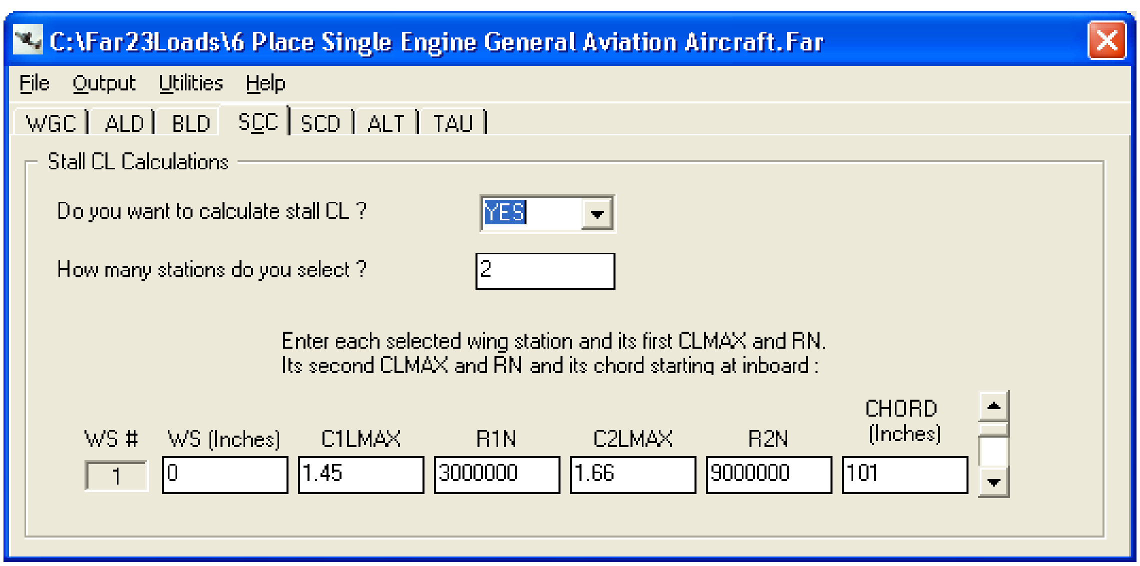


Figure 9.7 Aerodynamic Coefficients: Stall C_L Calculation Window when user selects YES

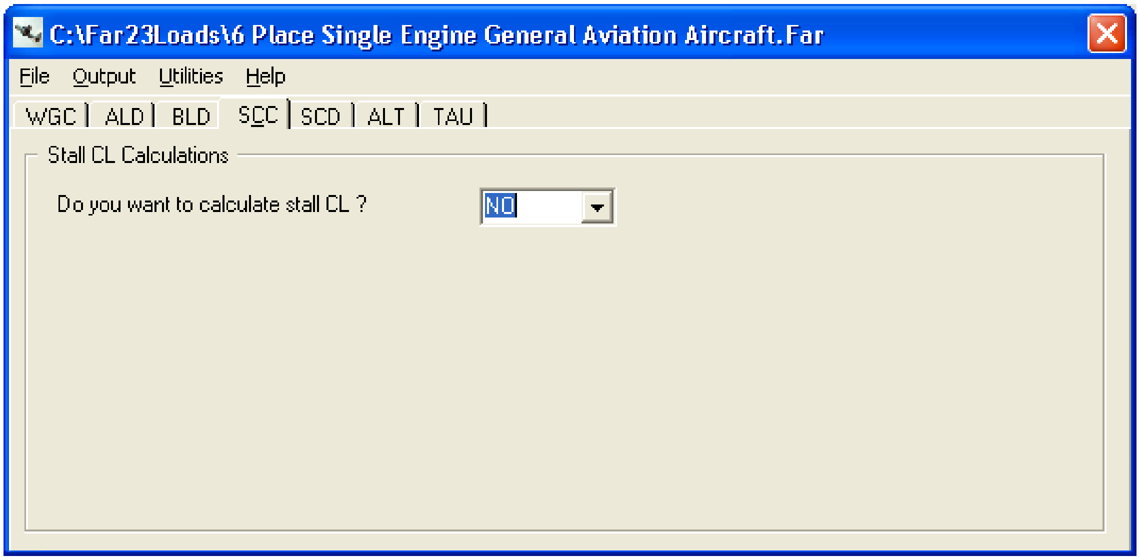


Figure 9.8 Aerodynamic Coefficients: Stall C_L Calculation Window when user selects NO

In the SCD page, parameters required for the spanwise coefficient distribution are entered. The input window appears as shown in figure 9.9. The user is required to input data for the spanwise drag and moment coefficients, C_D and C_M. A value for tau, which is a correction for the slope of the lift curve, is also entered on this page.

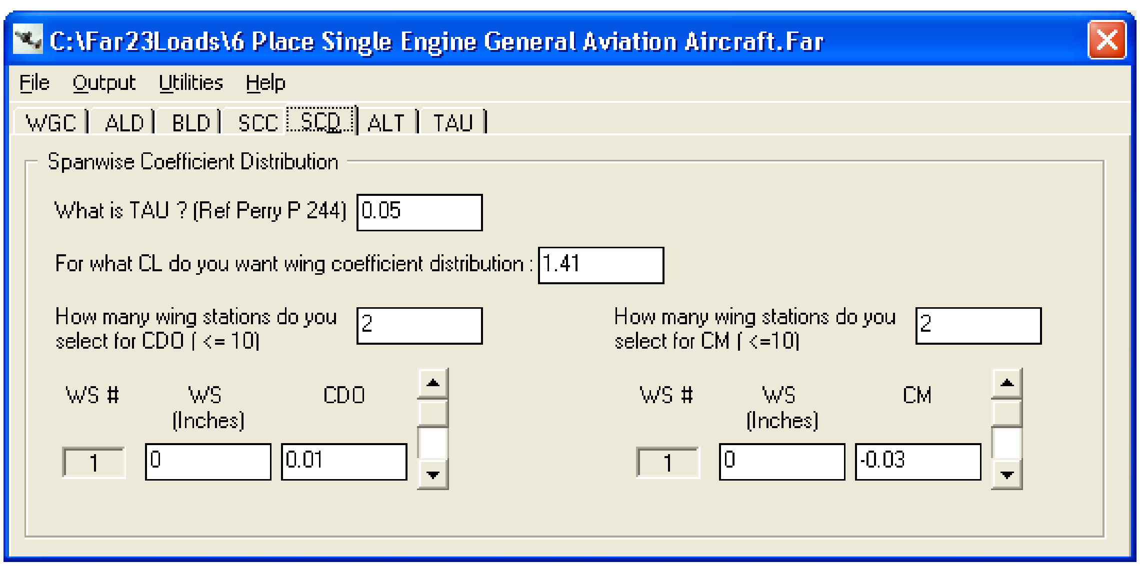


Figure 9.9 Aerodynamic Coefficients: Spanwise Coefficient Distribution

In the ALT page, additional fuselage data are entered. Figure 9.10 shows the ALT page. For the “Is landing gear extended?” prompt, landing gear information is not required if the user selects NO; otherwise, if the user selects YES, an additional page containing information about landing gear must be completed. Figure 9.11 shows the landing gear page.

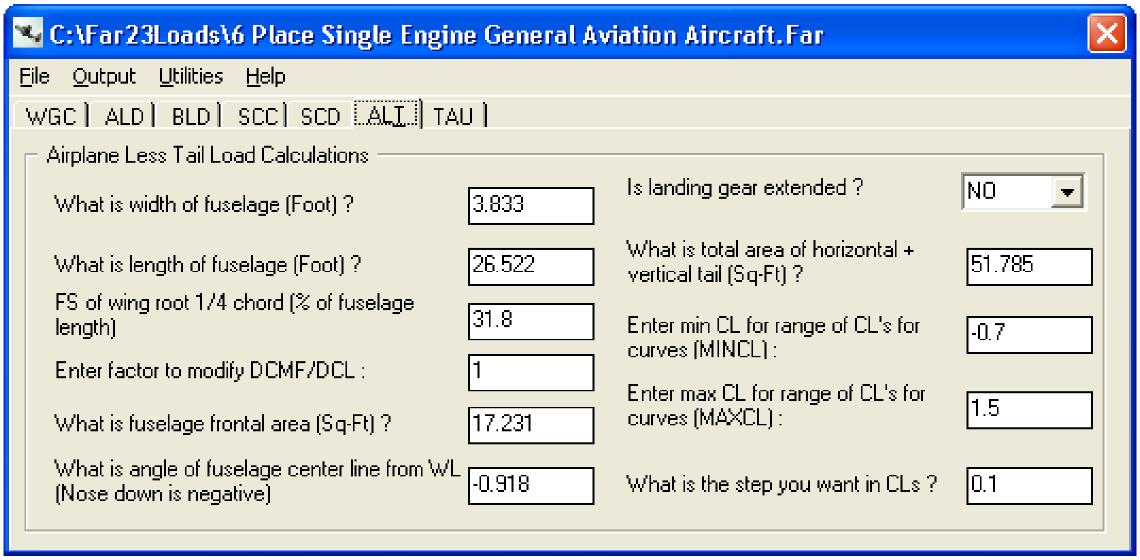


Figure 9.10 Aerodynamic Coefficients: Additional Fuselage Data

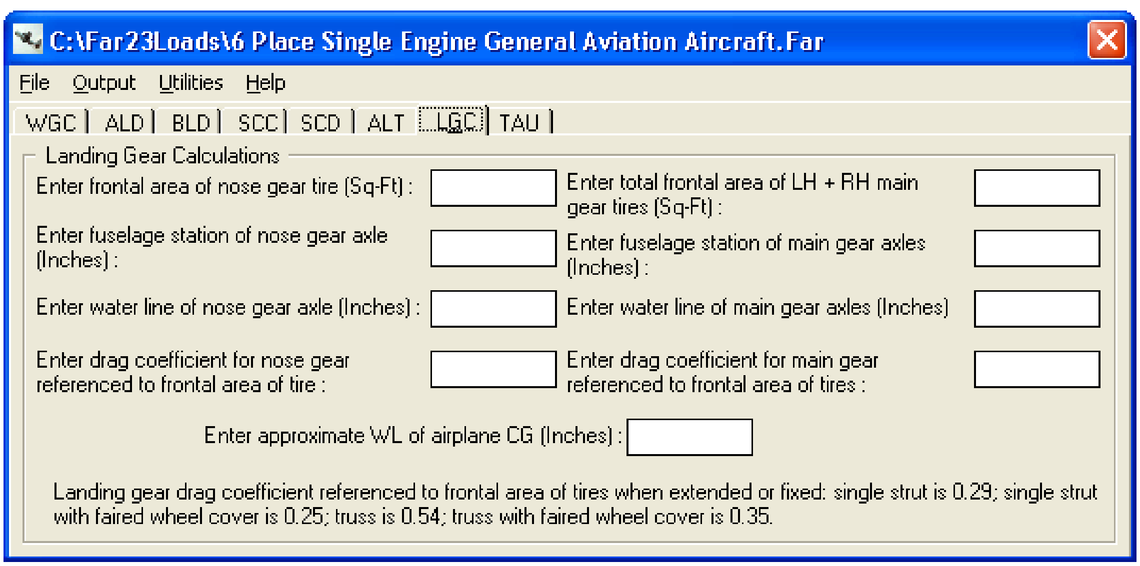


Figure 9.11 Aerodynamic Coefficients: Landing Gear Data

Data entry in the TAU page is optional. This page may only be used if the user does not have the value of tau available and desires to calculate it. The value of tau is calculated and displayed on the same page and is also updated on the SCD page. Figures 9.12 and 9.13 show the TAU calculation page and the updated tau value for the SCD page.

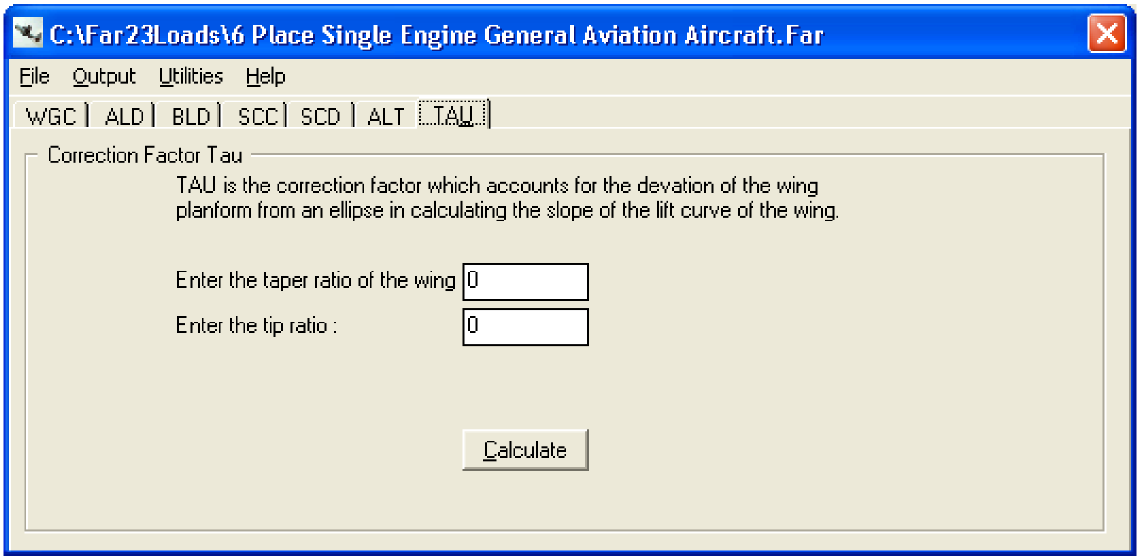


Figure 9.12 Aerodynamic Coefficients: TAU Calculation

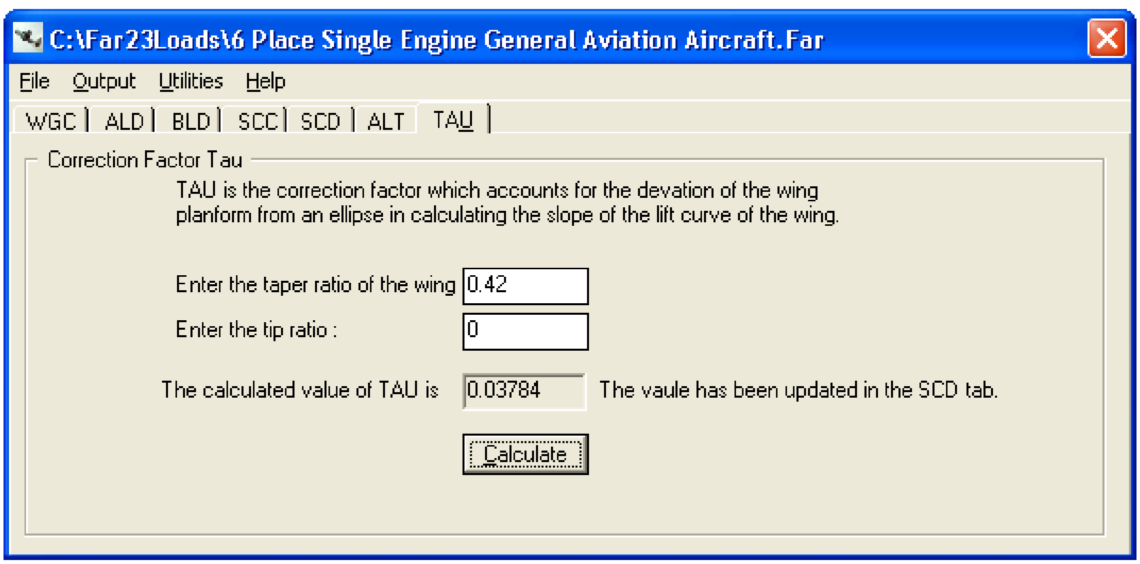


Figure 9.13 Aerodynamic Coefficients: TAU Updated in SCD Page

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the aerodynamic coefficients module.

9.3 Aerodynamic Coefficients Output

The aerodynamic coefficients module produces the following outputs:

- Wing geometry calculations
- Additive lift calculations
- Basic lift calculations
- Stall calculations
- Wing aerodynamic coefficient distributions
- Airplane-less-tail aerodynamic coefficients
- Equations for aerodynamic coefficients for the airplane-less-tail configuration

The aerodynamic coefficients for the airplane-less-tail are used in the 'Flight Loads' module.

10 Air Loads

10.1 Air Loads Description

The air loads module is used to calculate the air loads for cruise, landing and enroute conditions. This module calculates the basic and additive spanwise aerodynamic lift coefficient distributions for the wing. It combines these with the spanwise lift coefficient distribution for any specific total wing lift coefficient and then calculates the associated spanwise drag and moment coefficients for that wing C_L.

10.2 Running the Air Loads Module

To run the 'Air Loads' module, click the module from the main window. The main input window appears as shown in figure 10.1. The first window allows the selection of cruise, landing or enroute submodules. It also provides the flexibility to return to the main menu by clicking on the close button. Data input windows for Cruise, Landing and Enroute conditions are shown in figures 9.2, 9.3 and 9.4 respectively. The values in textboxes with yellow background are updated automatically from the respective modules.

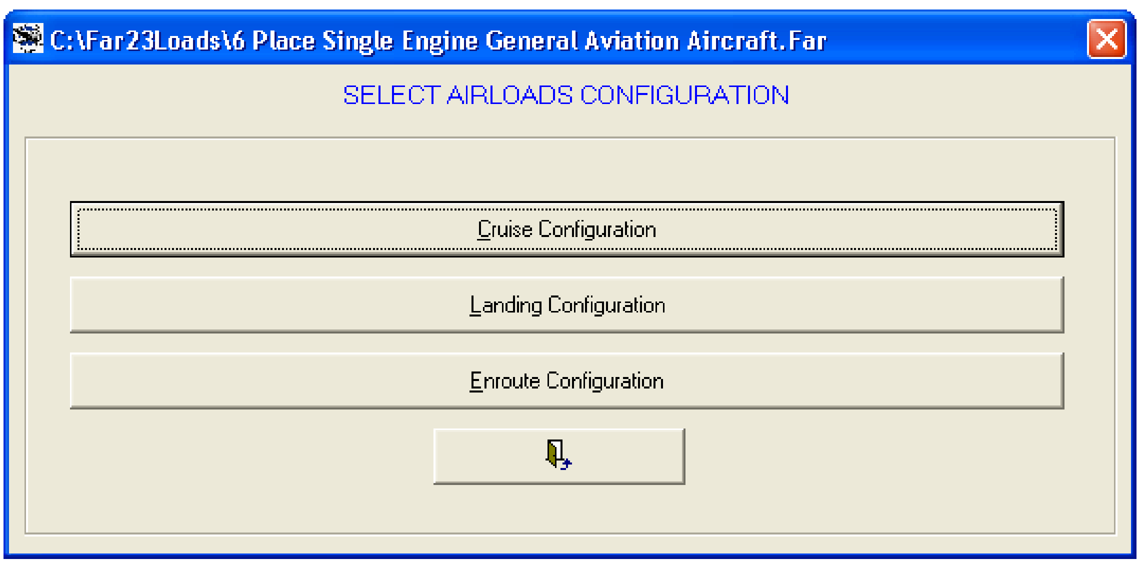


Figure 10.1 Air Loads Main Window

The data input window for any case has the general appearance and menu structure described in Section 9. Each window has 7 data entry pages:

1. WGC: Wing Geometry Calculations
2. ALD: Additive Lift Distribution

3. BLD: Basic Lift Distribution
4. SCC: Stall CL Calculations
5. SCD: Spanwise Coefficient Distribution
6. AFS: Air Loads for Specified C_L and V
7. TAU: Correction factor Tau

Note that page 6, the AFS page, is the only new entry that is not described in Section 9.

In the AFS page, air loads for a specified C_L and V are entered. Figure 10.2 shows the AFS page. The first question regarding air loads for the CL entered in the SCD tab is fixed as YES, and the user does not have option to change it. Inputs required for this module are the case number obtained from the 'Select' module, a description of the condition being analyzed, the airplane speed in KEAS, WL the wing reference plane at the plane of symmetry, and the slope of wing reference plane.

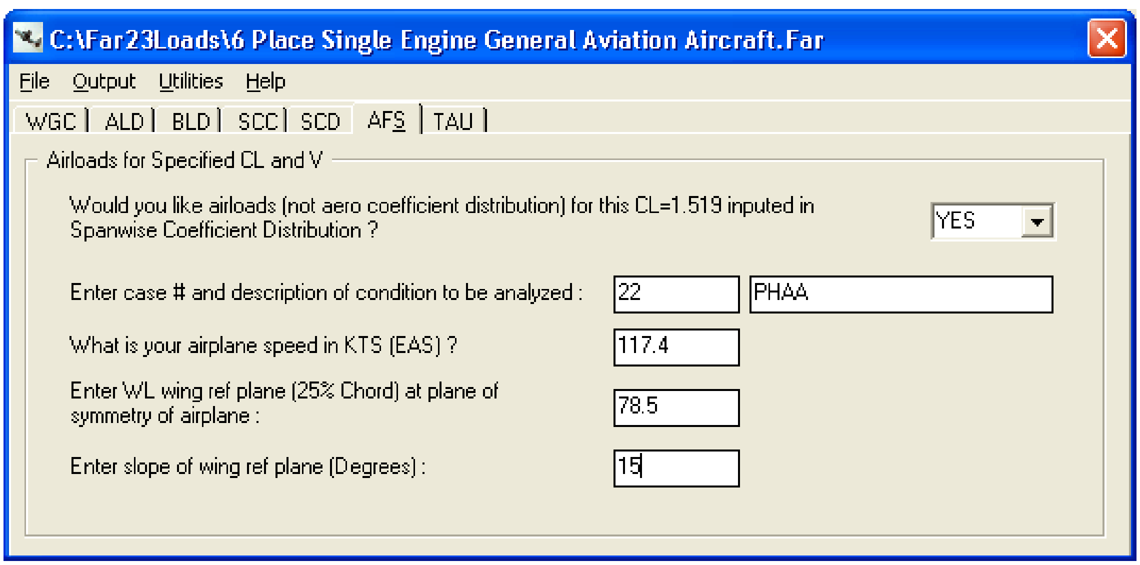


Figure 10.2 Air Loads: Air Loads for Specified C_L and V

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- File:
    - Save Data for NETLOADS Module - Enables user to save data in a format that can be read by the NETLOADS module. This module is inactive initially. After the user previews the output, this menu becomes active and the user can save the data with a desired file name.
    - Return to Air Loads Menu – Enables user to close existing menu and return to the main window of air loads module from which other submodules can be selected and run or the user can exit the air loads module.
- Help: The Help menu has only one option. It provides help on how to use the air loads module.

10.3 Air Loads Output

The air loads module produces the following outputs:

- Wing geometry calculations
- Additive lift calculations
- Basic lift calculations
- Stall calculations
- Wing airload distributions
- Equation for air loads

The air loads for airplane less tail are used in 'Flight Loads' module.

11 Flight Loads

11.1 Flight Loads Description

The ‘Flight Loads’ module calculates the loads for any combination of airspeed and load factor on and within the boundaries of the flight envelope. The data necessary to make these load calculations comes from the results of modules ‘Envelope of Loads’, ‘Wing Geometry’, ‘Structural Speed’ and ‘Air Loads’.

The flight envelope should be developed for altitudes up to the maximum operating altitude. For airplanes with a maximum operating altitude less than 20,000 feet, three altitudes are usually used: sea level, shoulder altitude, and maximum operating altitude. If the maximum operating altitude is greater than 20,000 feet, then 20,000 feet should be included since this is where the gust formulas begin to taper.

The flight envelope with flaps extended for takeoff, approach and landing needs to be determined at sea level only.

11.2 Running the Flight Loads Module

To run ‘Flight Loads’, click the module from the main window. The main input window appears as shown in figure 11.1. The values in textboxes with yellow background are updated automatically from respective modules.

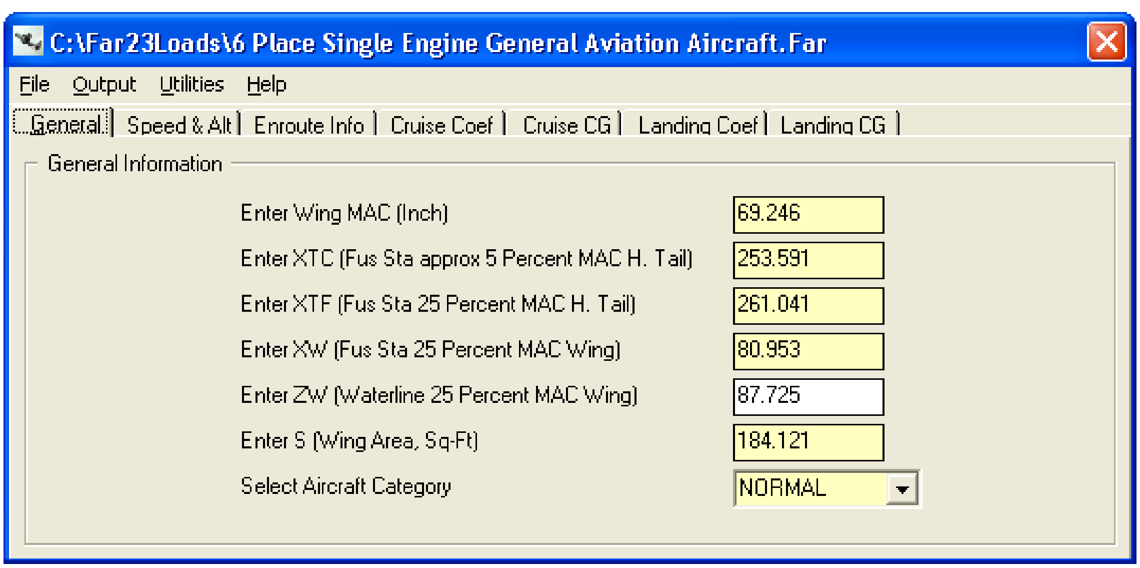


Figure 11.1 Flight Loads Main Window

The main window shows 7 or 9 pages depending upon the configuration. The pages are named as follows:

1. General: Contains general information about the airplane.
2. Speed & Alt: Contains information about speeds and altitudes.
3. Enroute Info: Contains information about enroute configuration. Based on data entered on this page, number of data entry pages varies from 7 to 9. If enroute data has to be entered, number of data entry pages is 9 else only 7.
4. Cruise Coef: Contains information about cruise coefficients.
5. Cruise CG: Contains information about cruise CG.
6. Landing Coef: Contains information about landing coefficients.
7. Landing CG: Contains information about landing CG.
8. Enroute Coef: Contains information about enroute coefficients (depends upon condition selected).
9. Enroute CG: Contains information about enroute CG (depends upon condition selected).

In the first window shown in figure 11.1, general data like category of airplane, normal, utility or acrobatic is entered. The category determines the minimum required load factor.

In the second window, data regarding speeds and altitudes are entered. Figure 11.2 shows the Speed & Alt page.

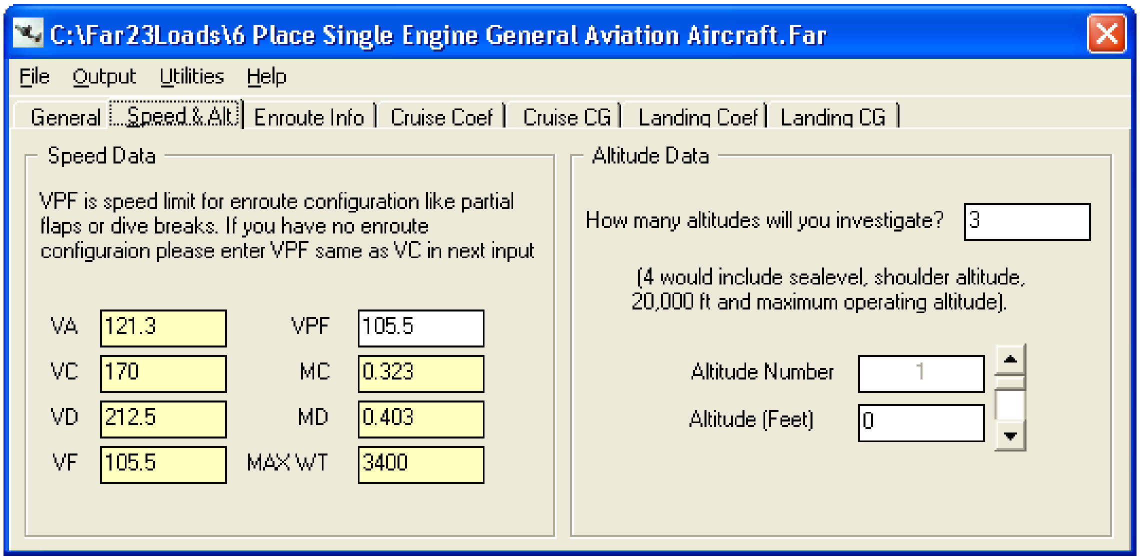


Figure 11.2 Flight Loads: Speed & Alt Page

On the Enroute Info page, the user is asked if there is an enroute configuration. If not, then the question about flaps for enroute will not appear, and the user will not see pages 8 and 9. Also, on the Speed & Alt page, if there is an enroute condition, enter a value for VPF that is the same as VC. A brief note about this can be seen in the Speed & Alt page shown above in figure 11.2. Figures 11.3 and 11.4 show the Enroute Info page.

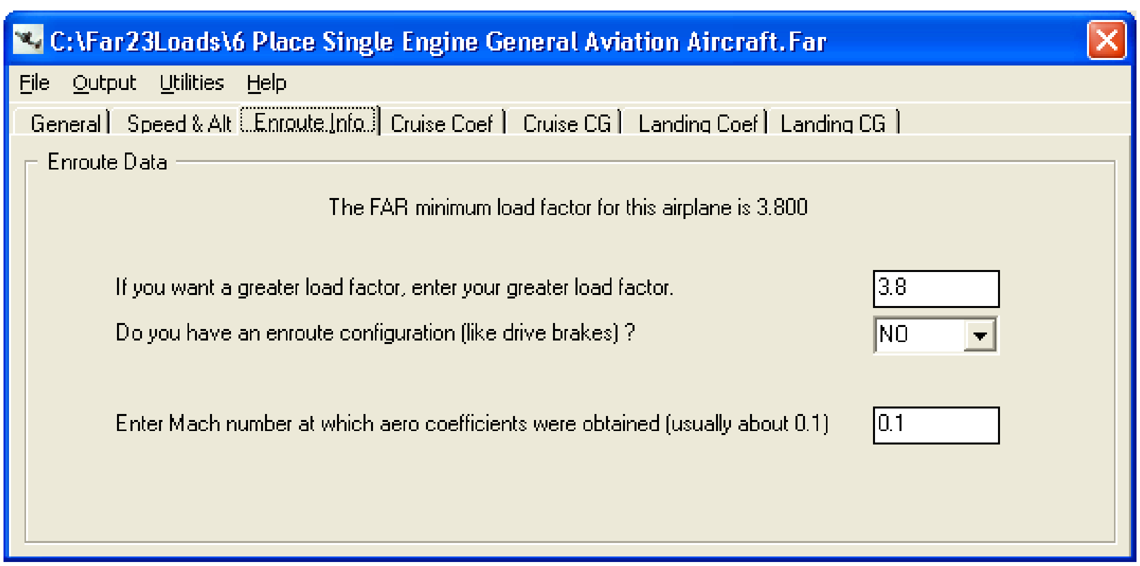


Figure 11.3 Flight Loads: Enroute Info Page with no enroute configuration

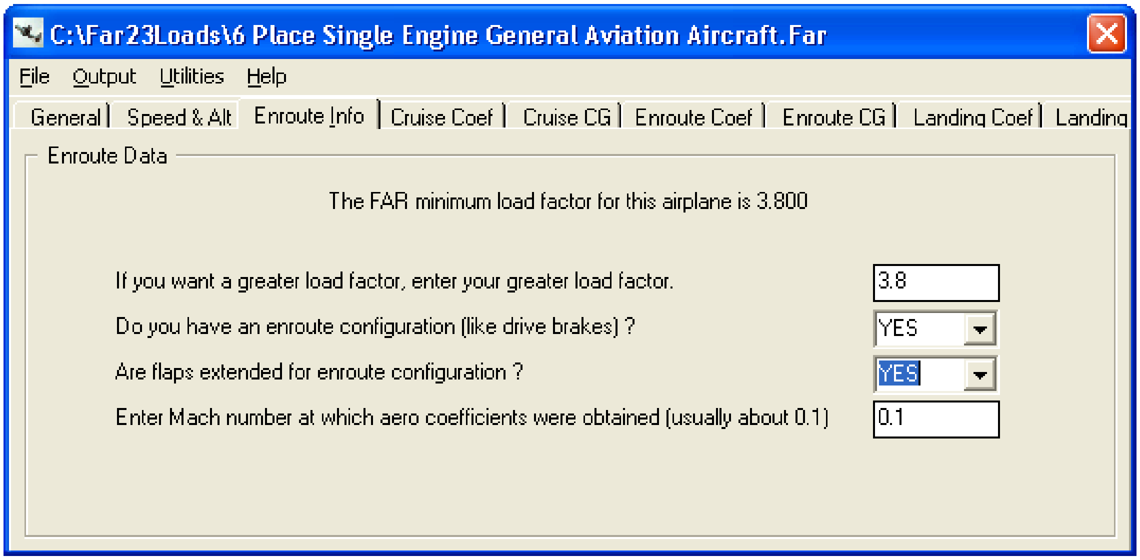


Figure 11.4 Flight Loads: Enroute Info Page with enroute configuration pages visible

The FAR minimum load factor for the airplane is calculated based on the category selected and the maximum airplane weight. The user also has the flexibility to change the value of load factor if desired.

Input pages designated by the cruise, landing and enroute coefficients and respective CGs are shown in figures 11.5 through 11.11. These pages are used to enter data for cruise, landing and enroute configurations. Each configuration requires two input pages: first where the coefficients for lift (C_0, C_1, C_2, C_3 and C_4), drag (D_0, D_1, D_2, D_3 and D_4) and pitching moment (M_0, M_1, M_2, M_3 and M_4) equations are entered, and second the CG data for each configuration.

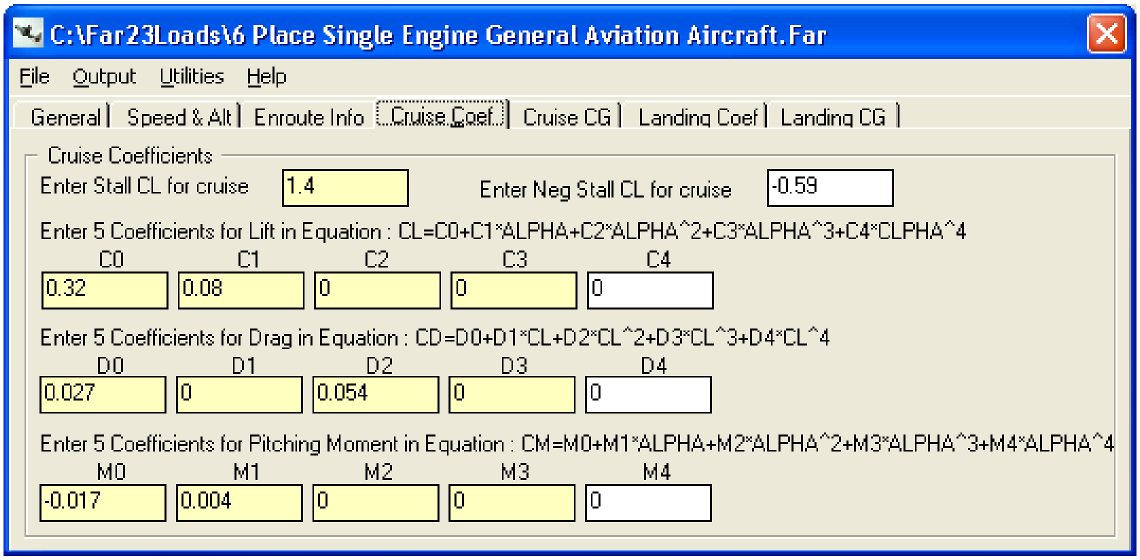


Figure 11.5 Flight Loads: Cruise Coefficients

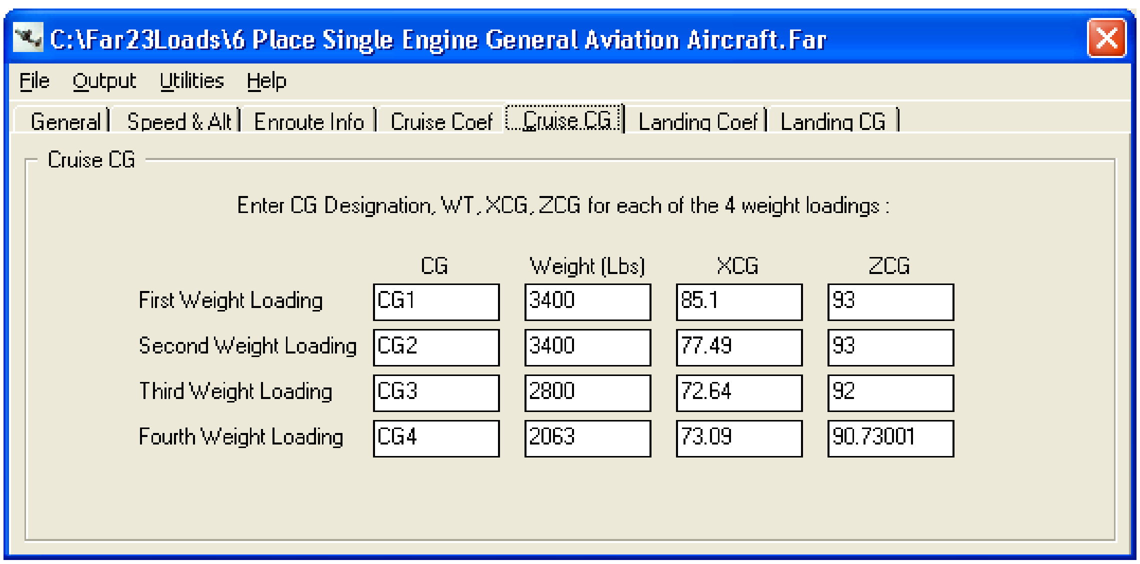


Figure 11.6 Flight Loads: Cruise CG

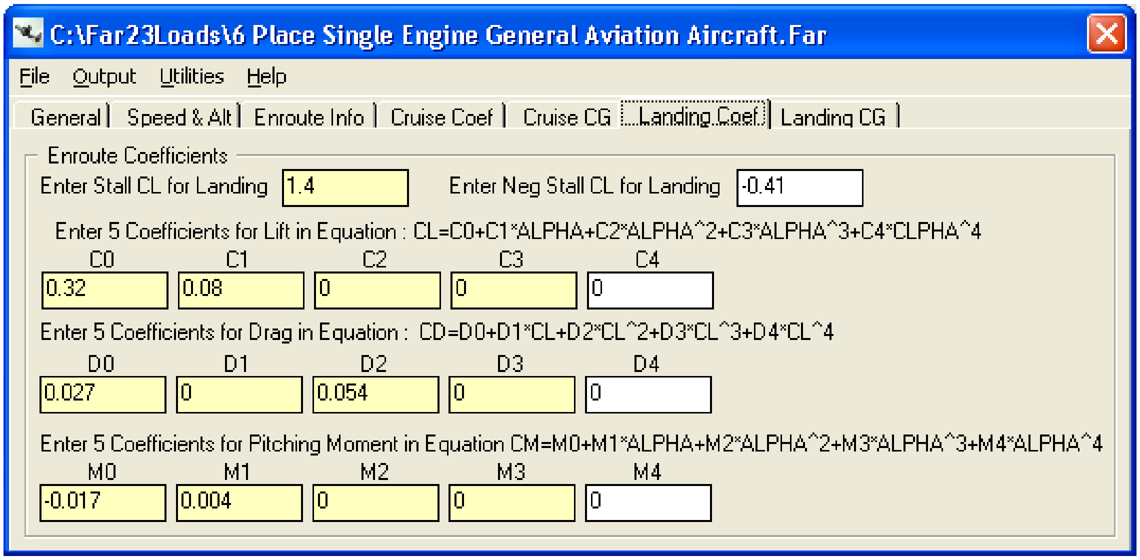


Figure 11.7 Flight Loads: Landing Coefficients

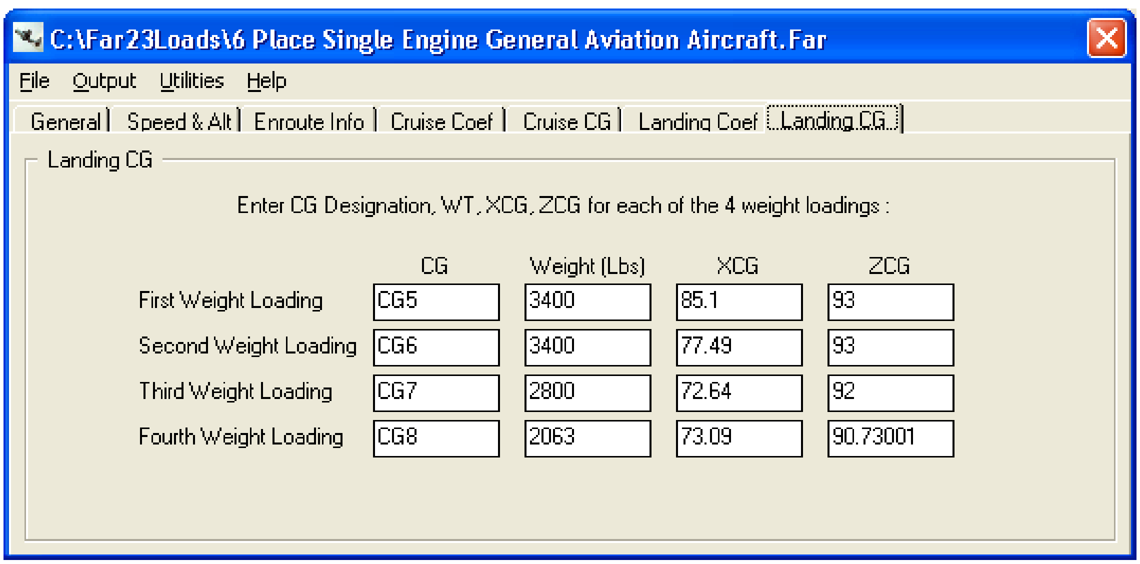


Figure 11.8 Flight Loads: Landing CG

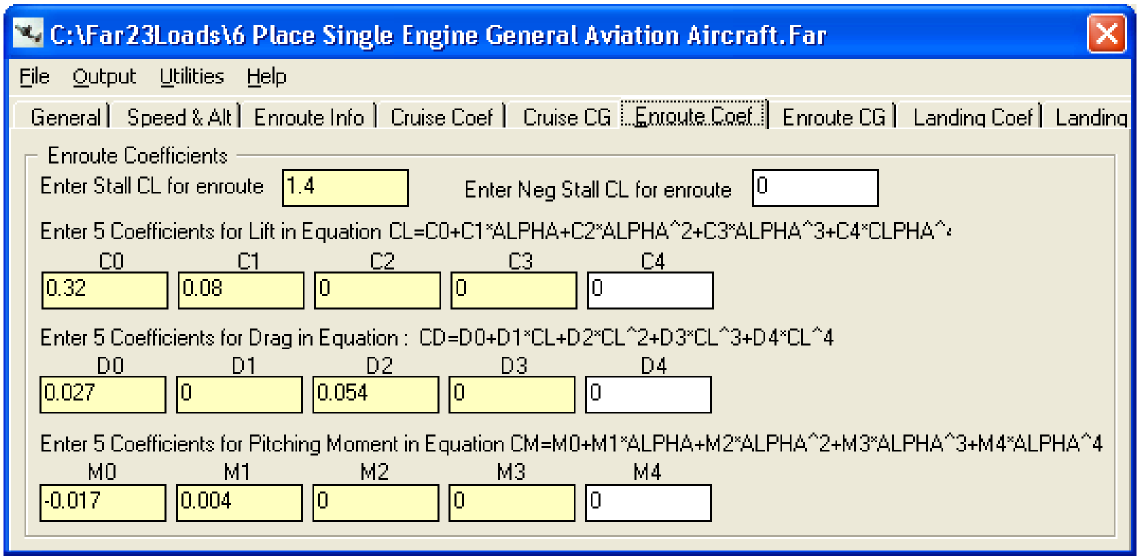


Figure 11.9 Flight Loads: Enroute Coefficients

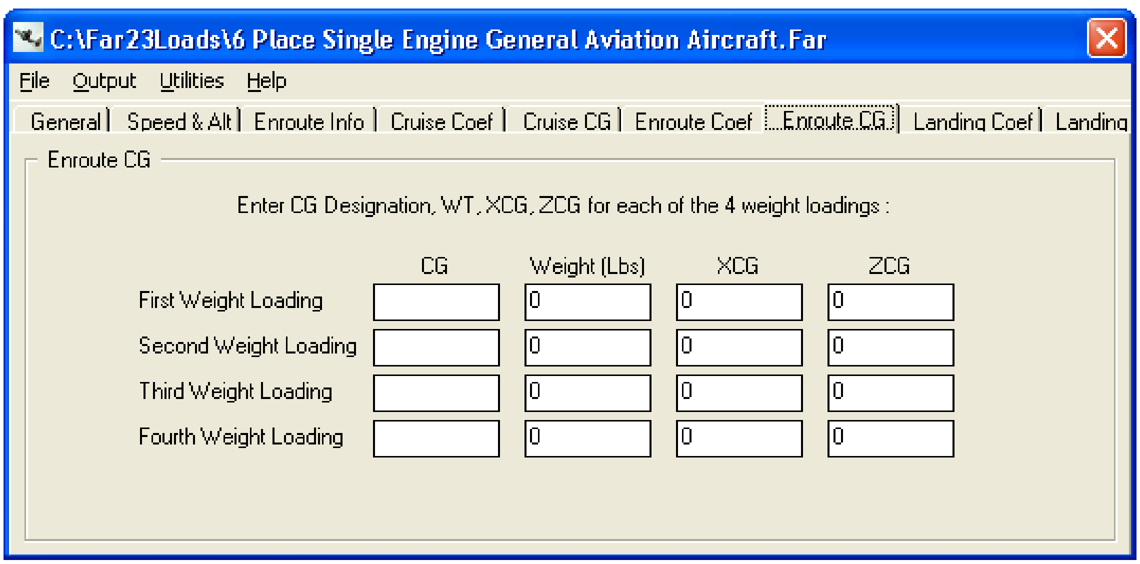


Figure 11.10 Flight Loads: Enroute CG

The data necessary to make these load calculations comes from the results of the respective modules when output data from these modules are updated.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- File:
  - Save Data for SELECT Module – Enables user to save data in a format that can be read by the SELECT module. This module is inactive initially. After the user previews the output, this menu becomes active and the user can save the data with a desired file name.
- Help: The Help menu has only one option. It provides help on how to use the flight loads module.

11.3 Flight Loads Output

The output from the 'Flight Loads' module is used in the 'Select' and 'Wing Inertia' modules. Data for each point on the flight envelope is included in the output file. The following data is included and if applicable, the variable name used in the output file is given:

- Name of the condition and case number
- Altitude (feet) and equivalent air speed (knots)
- Normal load factor n_z
- Angle of attack α (degrees)
- Compressibility factor (variable G CORR)
- Wing lift coefficient C_L
- Pitching moment of airplane less tail (variable M(W+F))
- Wing lift normal to the airplane reference line (lbs) (variable LZW)
- Tail load (lbs) (variable LT)
- Airplane drag load (lbs) (variable DX)

12 Selection of Critical Loads

12.1 Selection of Critical Loads Description

The critical flight loads are determined by the ‘Select’ module using the results of the ‘Flight Loads’ module. The output from the ‘Flight Loads’ module contains all of the balanced symmetrical flight conditions on the V-n diagram. ‘Select’ searches this file for the critical flight loads on the wing, fuselage, horizontal tail, and vertical tail. Critical loads for other structures such as ailerons, flaps, engine mounts, landing gear and tabs are determined in other modules as explained later.

In addition to the flight envelope data, additional geometry and inertia data from the ‘Weight & CG’ and ‘Envelope of Loading’ modules are required.

12.2 Running the Select Module

To run ‘Select’, click the module from the main window. The main input window appears as shown in figure 12.1. All buttons except the exit button are inactive initially. The File menu contains only one submenu:

- Read V-n Flight Loads Data – Enables user to select the flight loads file that was saved in the Flight Loads module using the Save Data For SELECT Module.

After the user selects an appropriate file (an error message appears if the file does not contain the flight loads data that can be read by this module), the Critical Wing Loads, Critical Horizontal Tail Loads, Critical Vertical Tail Loads, and Critical Fuselage Loads buttons are activated.

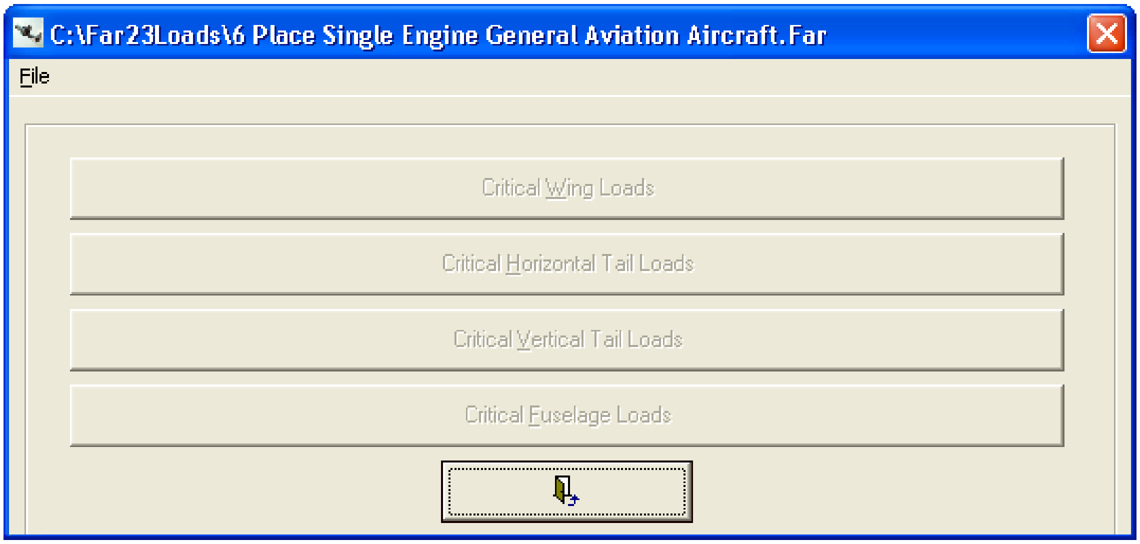


Figure 12.1 Select Critical Loads Main Window

Figure 12.2 shows the critical wing loads window. Inputs required for this window are the full aileron deflection and the airfoil section moment coefficient for no aileron. The values in textboxes with yellow background are updated automatically from respective modules.

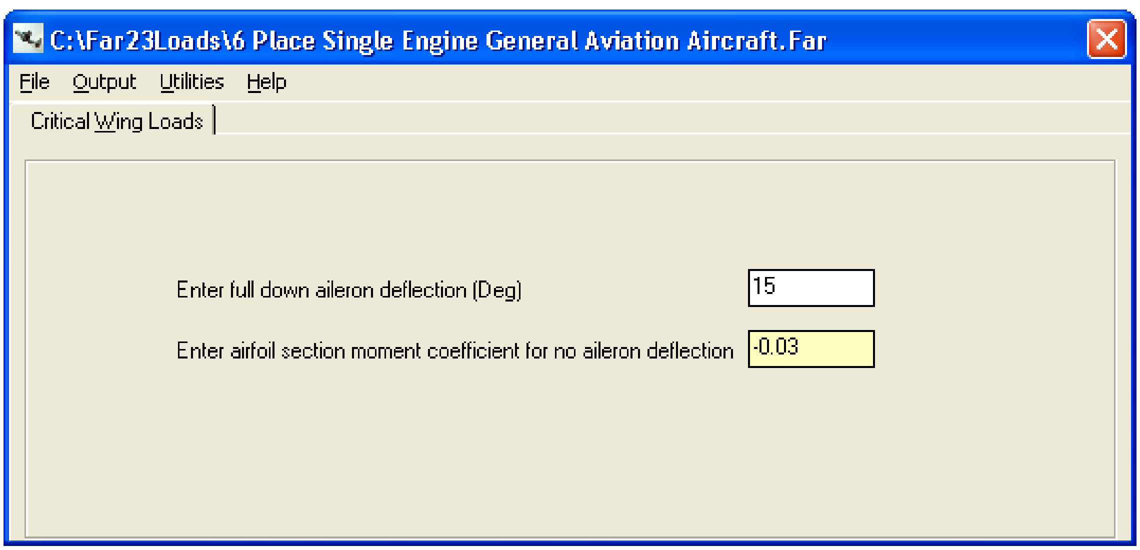


Figure 12.2 Critical Wing Loads Window

Figure 12.3 shows the critical horizontal tail loads window. The values in textboxes with yellow background are updated automatically from respective modules.

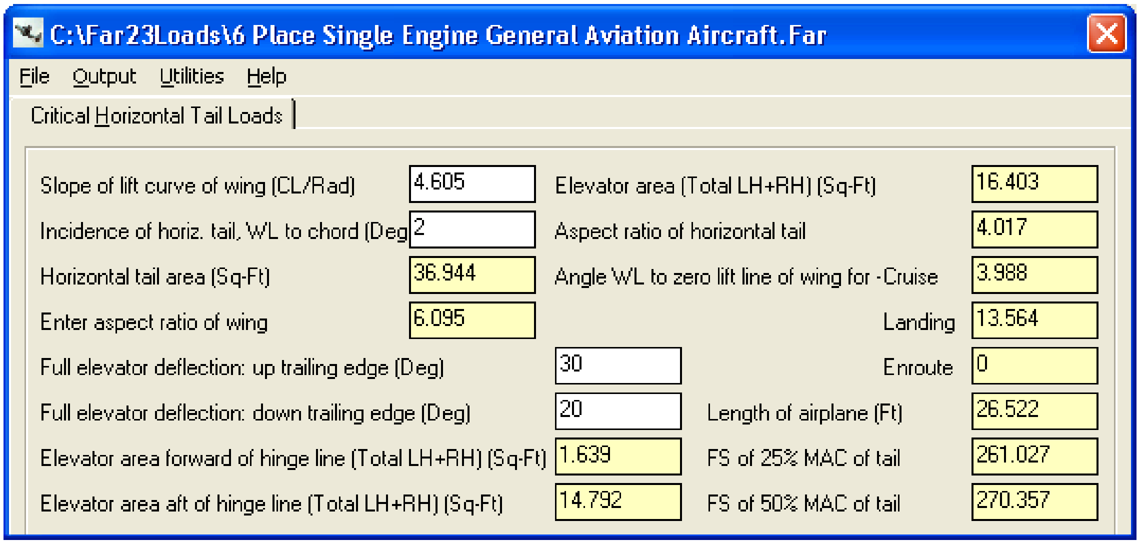


Figure 12.3 Critical Horizontal Tail Loads Window

Figure 12.4 shows the critical vertical tail loads window. The values in textboxes with yellow background are updated automatically from respective modules. The user is asked if the default moment of inertia will be used to calculate the gust on vertical tail. If the user selects YES, then default values are used; otherwise, the user is asked to input the values manually.

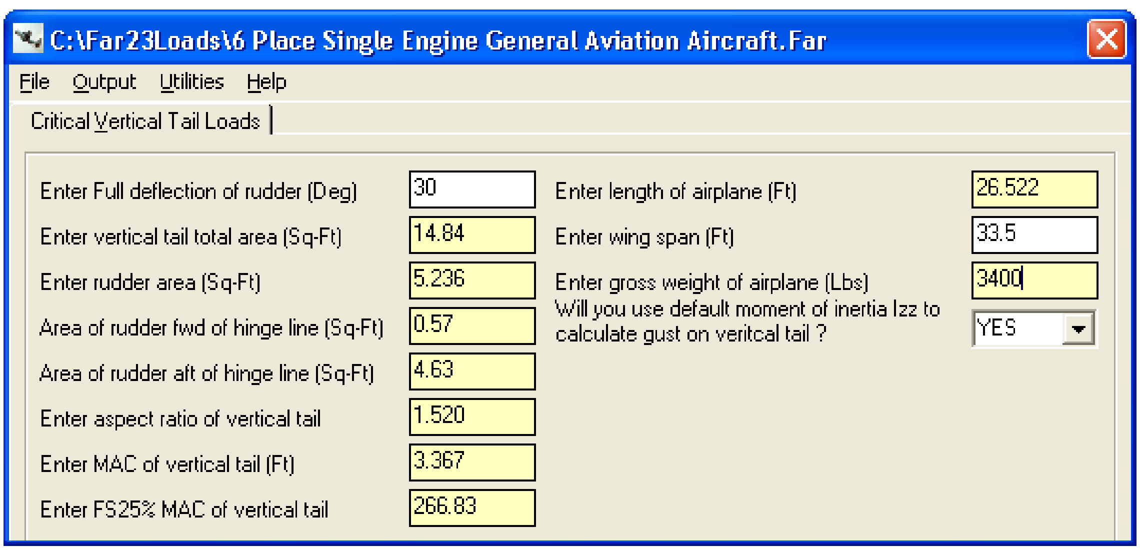


Figure 12.4 Critical Vertical Tail Loads Window

Figure 12.5 shows the critical fuselage loads window. Data required for this submodule are the location where the engine is mounted and the wing weight. Default wing weight is taken as 0.9 times the gross weight.

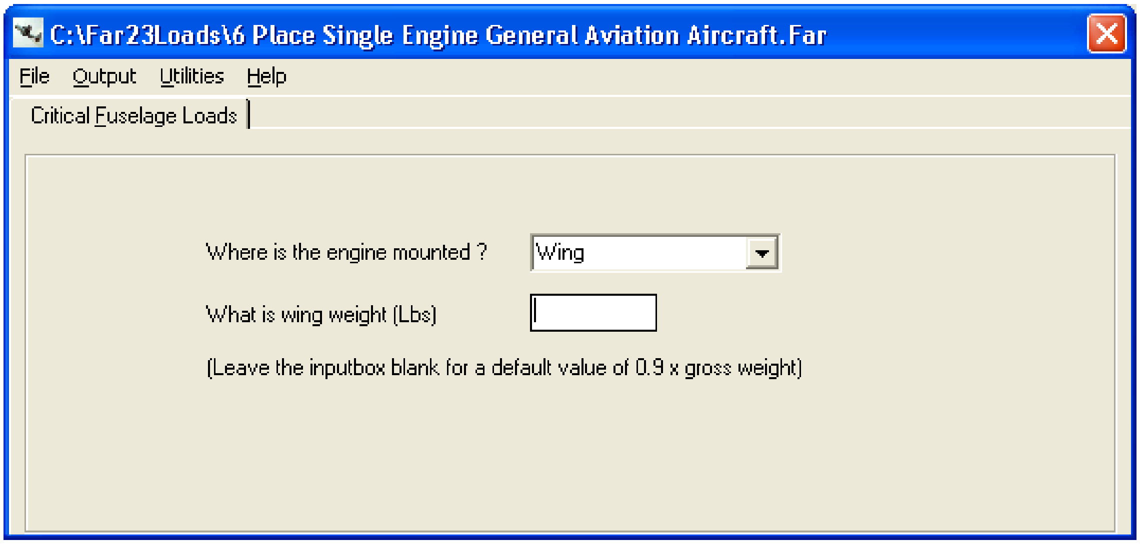


Figure 12.5 Critical Fuselage Loads Window

After entering the data for a component, the critical loads for that component must be saved before moving to the next component. Each submodule has a similar menu structure as described below.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the select module.

12.3 Select Module Output

'Select' determines the critical loads for the wing, fuselage, vertical tail, and horizontal tail. A separate output file is created for each component. The output file lists each critical condition with appropriate parameters.

The results from this module are used in the 'Air Loads', 'Wing Inertia', and 'Tail Load Distribution' modules.

13 Aileron Loads

13.1 Aileron Loads Description

The loads on the aileron are calculated in the module ‘Aileron Loads’. The deflected positions during unsymmetrical flight conditions produce the critical loads.

To calculate the aileron loads, the required data include:

- Airspeeds V_A, V_C and V_D
- Area of aileron forward and aft of hinge line
- Maximum up deflection at V_A
- Maximum down deflection at V_A

13.2 Running the Aileron Loads Module

To run ‘Aileron Loads’, click the module from the main window. The main input window appears as shown in figure 13.1. The values in textboxes with yellow background are updated automatically from respective modules.

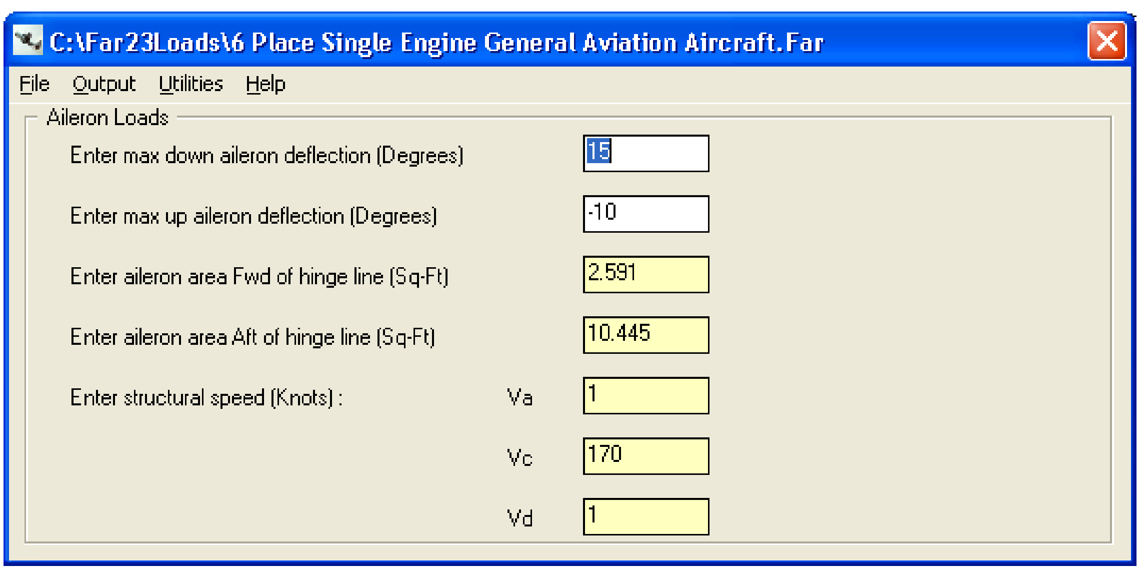


Figure 13.1 Aileron Loads Main Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the aileron loads module.

13.3 Aileron Loads Output

The output from this module includes the up and down aileron deflections at V_A, V_C and V_D. The critical load for the up and down aileron deflection and the pressure forward of the hinge line for up and down aileron is calculated.

The deflection is given in degrees, the critical load in pounds, and the pressure in lb/in^2.

14 Flap Loads

14.1 Flap Loads Description

The ‘Flap Loads’ module calculates the critical flap loads determined by calculating the lift on the flap due to wing angle of attack plus lift on the flap due to the deflection of the flap.

To calculate the flap loads, the required data include:

- Stalling speed at maximum takeoff weight with flaps retracted (knots)
- Stalling speed at maximum takeoff weight with flaps extended (knots)
- Design flap speed (knots)
- Maximum takeoff weight (lbs)
- Gust load factor with flaps extended
- Flap area on one side of airplane (ft^2)
- Total area of wing (ft^2)
- Maximum flap deflection (degrees)
- Ratio of flap chord to wing chord
- Maximum horse power of one engine
- Butt line of engine (in)
- Frontal area of nacelle (ft^2)
- Propeller diameter (in)

14.2 Running the Flap Loads Module

To run ‘Flap Loads’, click the module from the main window. The main input window appears as shown in figure 14.1. The values in textboxes with yellow background are updated automatically from respective modules.

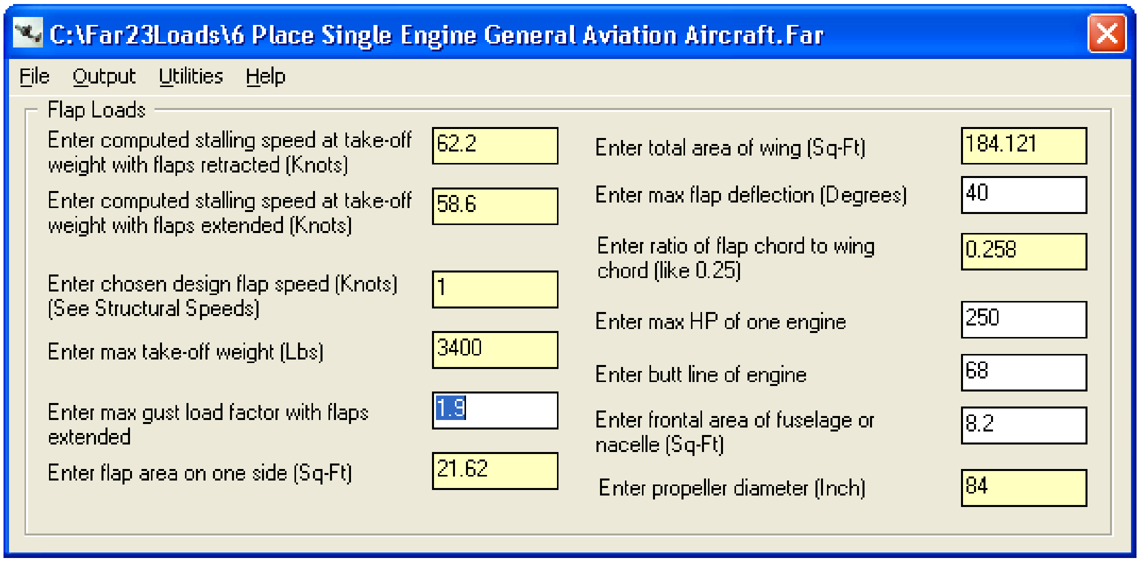


Figure 14.1 Flap Loads Main Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the aileron loads module.

14.3 Flap Loads Output

The output from this module includes the lift coefficients for the wing and flap as well as the flap load for the following conditions:

- 1 g stall
- 2 g stall
- 2 g at V_F
- 1.9 g at V_F

Additional output includes the critical flap load and the pressure at the leading edge. The butt line for the inboard and outboard edge of the slipstream velocity are given as well as the slipstream velocity at the flap. For a horizontal gust of 25 fps, the factor to increase the flap load at V_F and the critical flap load combined with the horizontal gust are given.

15 Wing Inertia

15.1 Wing Inertia Description

The ‘Wing Inertia’ module calculates the spanwise inertia shears and moments for a balanced and accelerated flight condition along the quarter chord of the wing for the critical wing conditions. Concentrated weights such as landing gear, engines, fuel tanks, and external wing stores are accounted for in the calculations.

Using the coordinates of the leading and trailing edges, the wing is divided into incremental chordwise strips. For each strip, the inertia loads, shears, and moments are calculated.

The inputs required for this module includes wing panel weight, inertia factors obtained for the selected critical wing loads, ratio of densities of the tip area to the root area, wing plan-form geometry, dihedral angle of the wing reference plane and waterline of its intersection with the center plane of symmetry at the quarter chord, weight and coordinates of the concentrated weights, wing station of inboard rib of wing panel, and the load conditions.

15.2 Running the Wing Inertia Module

To run ‘Wing Inertia’, click the module from the main window. The main input window appears as shown in figure 15.1. The values in textboxes with yellow background are updated automatically from respective modules.

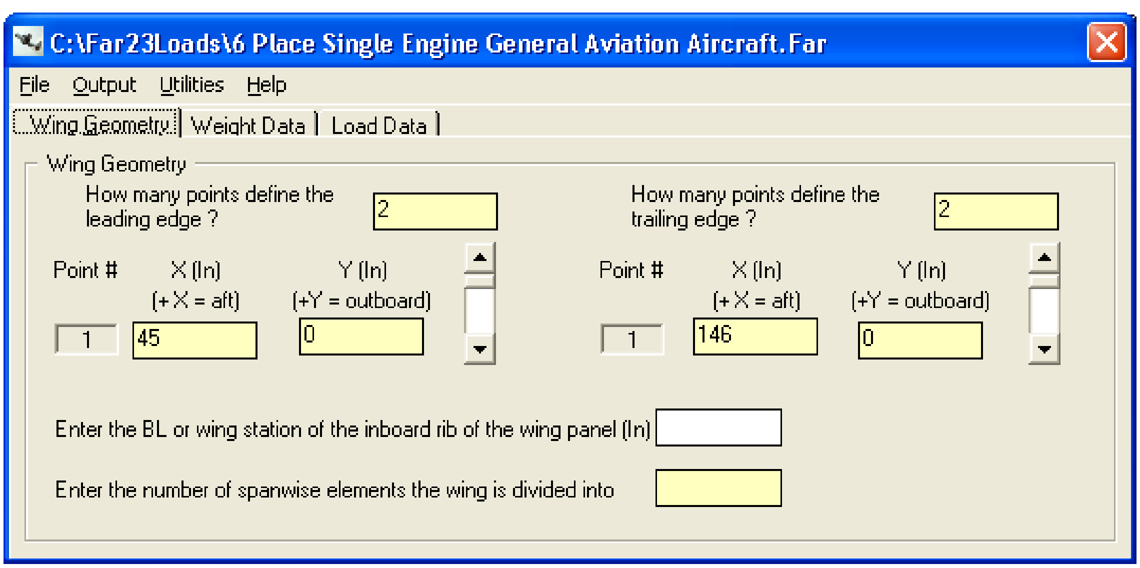


Figure 15.1 Wing Inertia Main Window

The main window has 3 data entry pages:

1. Wing Geometry: Contains information for carrying out wing geometry calculations
2. Weight Data: Contains information about different weight data
3. Load Data: Contains information about different loads and points of application

In the first window, the wing plan-form geometry (entered as leading and trailing edge coordinates), wing station of inboard rib of the wing panel, and number of wing spanwise increments (between 2 and 100) are entered.

In the second window, the wing panel weight (lb) not including concentrated weight, dihedral angle of the wing reference plane and waterline of its intersection with the center plane of symmetry at the quarter chord, and weight and coordinates for concentrated weights are entered. Figure 15.2 shows the weight data page.

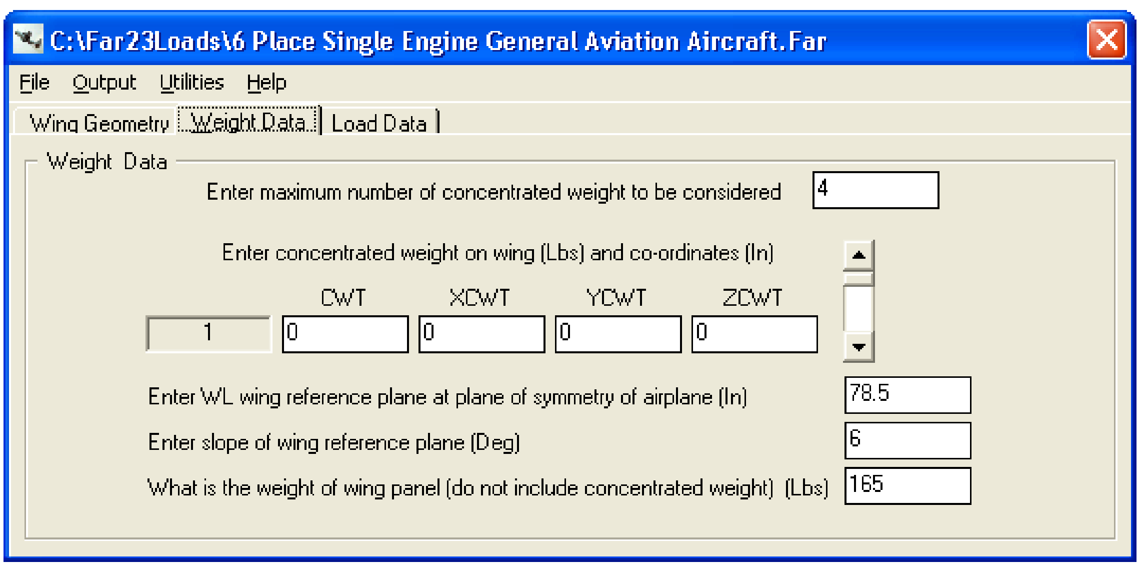


Figure 15.2 Wing Inertia: Weight Data Window

In the third window, the ratio of densities of the tip area to the root area and load conditions including case number, N_z, N_x and unbalanced moment are entered. Figure 15.3 shows the load data window.

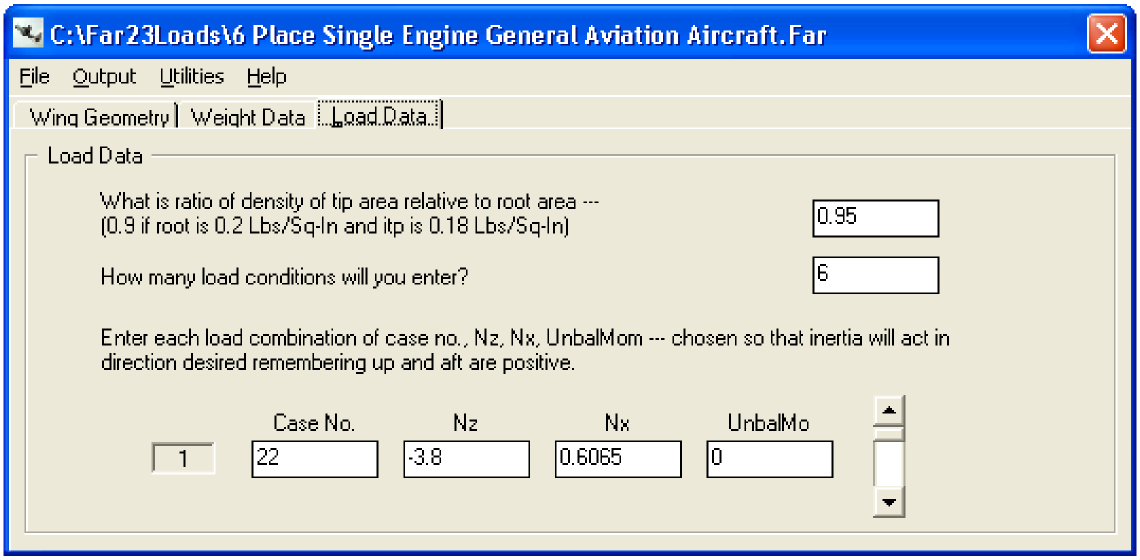


Figure 15.3 Wing Inertia: Load Data Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- File:
  - Save Data for NETLOADS Module – Enables user to save data in a format that can be read by the NETLOADS module. This module is inactive initially. After the user previews the output, this menu becomes active and the user can save the data with a desired file name.
- Help: The Help menu has only one option. It provides help on how to use the wing inertia module.

15.3 Wing Inertia Output

For each case, the output includes the load, shear, and torsion for each spanwise increment of the wing. This output is used in the 'Net Loads' module when determining the total loads.

In the output file, the data is labeled by the variable names. These variable names are defined below:

- Input case number (Case)
- Input load factors for the X and Z directions (N_X and N_Z)
- Rate of change of pitch velocity (THETADOT)

- Input value of unbalanced moment (in-lbs) (UNBAL MOM)
- Coordinates of the quarter chord for the wing increment (in) (X, Y and Z)
- Total inertia force in the X and Z directions (FX and FZ)
- Incremental torsion (in-lbs) (DMYY)
- Total drag force in X and Z directions (lbs) (SX, SZ)
- Bending moment, torsion, and yawing moment (in-lbs) (MXX, MYY and MZZ)

16 Net Loads

16.1 Net Loads Description

The module 'Net Loads' calculates the spanwise net wing shears and moments along the quarter chord of the wing. The air loads and inertia loads are algebraically added to determine the net loads.

The input data required for the calculations are the air loads and inertia loads for the selected critical wing loads. The air loads come from the 'Air Loads' module and the inertia loads come from the 'Wing Inertia' module.

16.2 Running the Net Loads Module

To run 'Net Loads', click the module from the main window. The main input window appears as shown in figure 16.1.

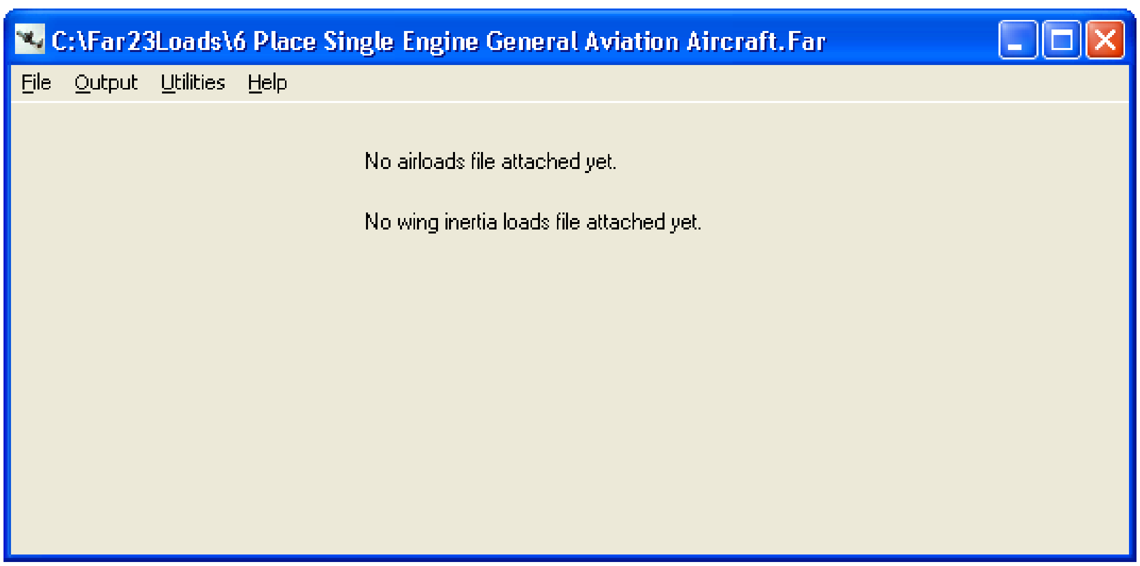


Figure 16.1 Net Loads Main Window

In this module, the loads files saved from the air loads and the wing inertia loads modules are attached and the analysis is performed. Initially, when no files are attached, the window appears as shown in figure 16.1.

From the file menu, select the submenu 'Open Air Loads File' to attach the air loads file. If this file is not the one that was saved from the 'Air Loads' module for the ‘Netloads’ module, then an error message will pop up. Figure 16.2 shows the error message that appears when the wrong air loads file is attached.

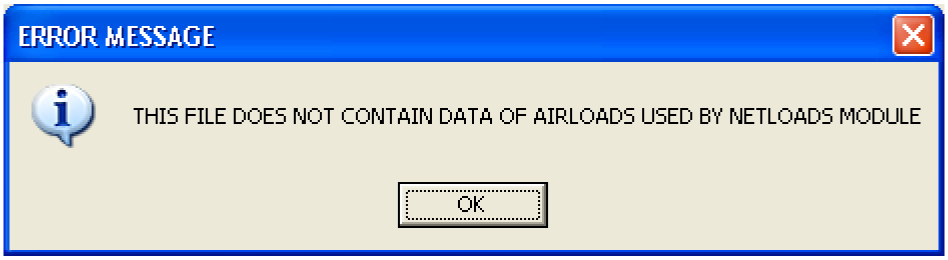


Figure 16.2 Error Message when wrong Air Loads file is Attached

From the file menu, select the submenu ‘Open Wing Inertia File’ to attach the wing inertia file. If this file is not the one that was saved from the ‘Wing Inertia’ module for the ‘Netloads’ module, then an error message will pop up. Figure 16.3 shows the error message that appears when the wrong wing inertia file is attached

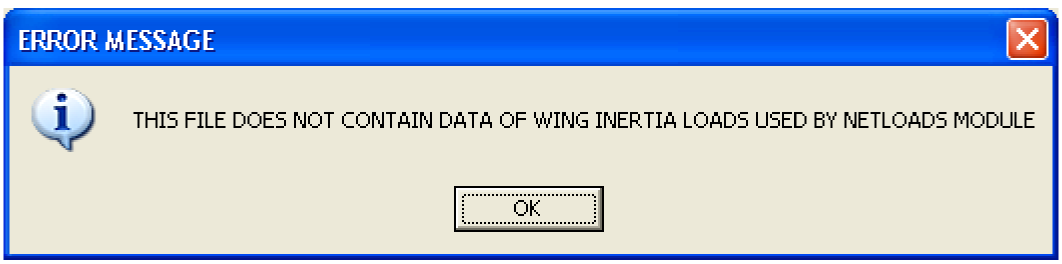


Figure 16.3 Error Message when wrong Wing Inertia file is Attached

When the correct air loads and wing inertia files are attached, the main window appears as shown in figure 16.4. Analysis can be performed once both correct files are attached. If the air loads and wing inertia files do not contain a common case number, an error message appears. Figure 16.5 shows the error message when the program fails to find a common case number.

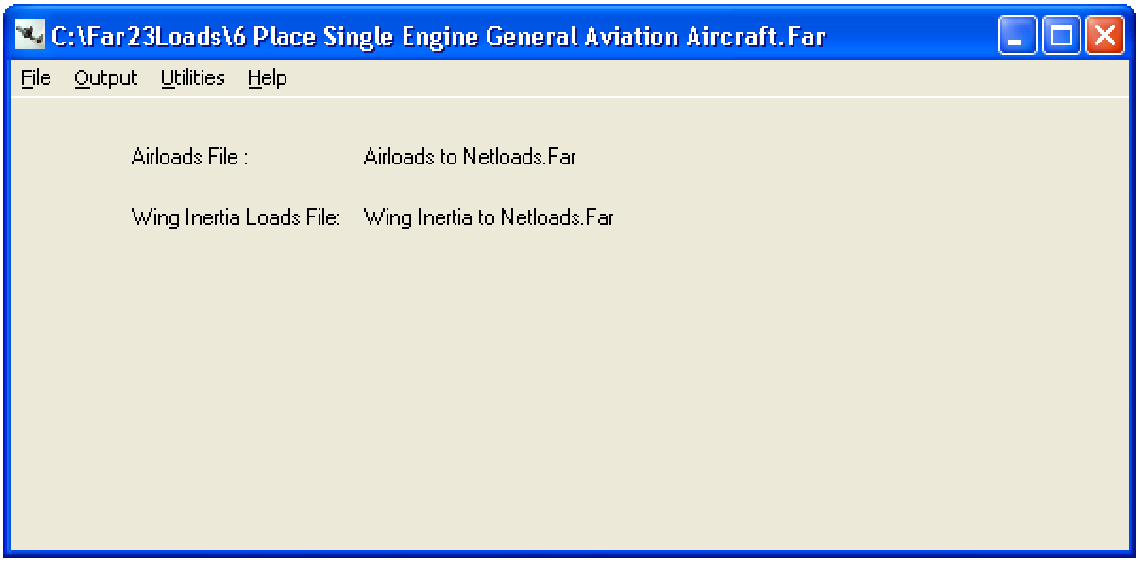


Figure 16.4 Net Loads Main Window with both Correct files Attached

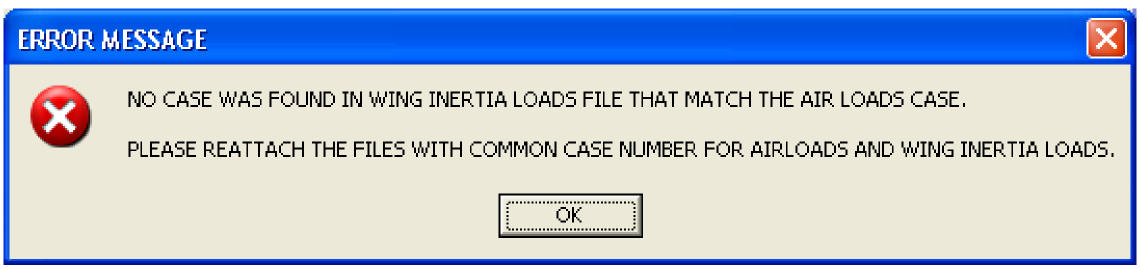


Figure 16.5 Error Message when no Common Case is Found

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- File:
  - Open Air Loads File – Enables user to open the file that had been saved in the 'Air Loads' module that saved data that can be opened in Net Loads module.
  - Open Wing Inertia File – Enables user to open the file that had been saved in the 'Wing Inertia' module that saved data that can be opened in Net Loads module.
- Help: The Help menu has only one option. It provides help on how to use the net loads module.

16.3 Net Loads Output

The output includes the spanwise net wing loads, shears, and moments along the quarter chord of the wing. In the output file, the data is labeled by the variable names. These variable names are defined below (positive direction is up and aft):

- Coordinates of the quarter chord (X, Y and Z)
- Normal force in the X and Z directions (lb) (FX and FZ)
- Drag and shear loads (lb) (SX, SZ)
- Moments (in-lb) (MX, MY, MZ)

17 Engine Mount Loads

17.1 Engine Mount Loads Description

The ‘Engine Mount Loads’ module calculates the loads on the engine mount and its supporting structure. These loads include those resulting from engine torque loads, vertical inertia loads, and side inertia loads. For turbine engines, the torque due to sudden stoppage is also included. For turbine-powered airplanes, the gyroscopic and aerodynamic loads resulting from the combination of yaw velocity, pitching velocity, normal inertia loads, and propeller thrust must be considered.

17.2 Running the Engine Mount Loads Module

To run ‘Engine Mount Loads’, click the module from the main window. The main input window appears as shown in figure 17.1. The values in textboxes with yellow background are updated automatically from respective modules.

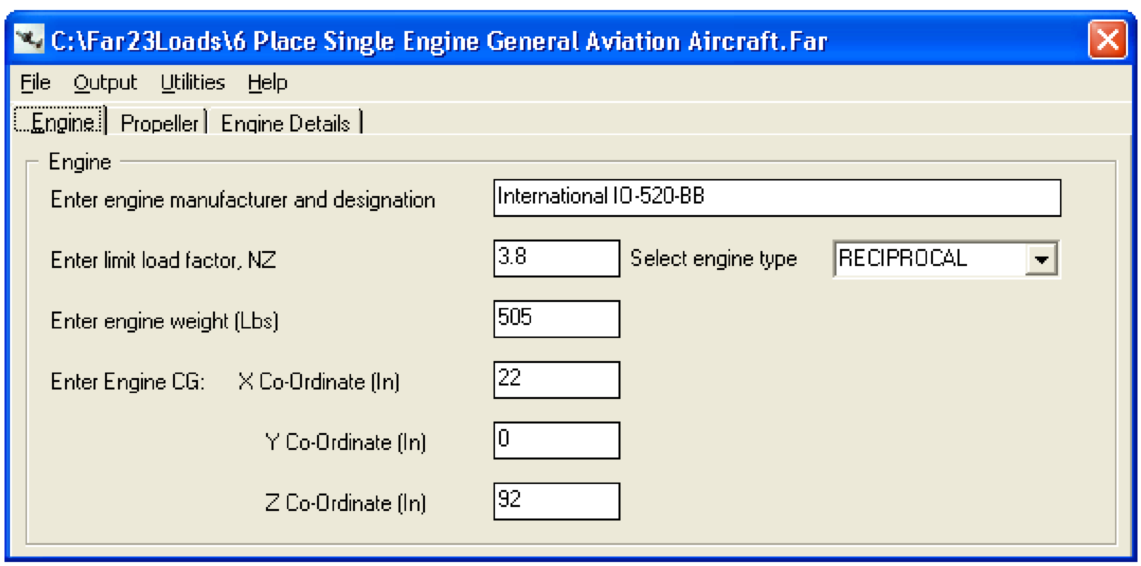


Figure 17.1 Engine Mount Loads Main Window

The main window has 3 data entry pages:

4. Engine: Contains general engine data
5. Propeller: Contains propeller data
6. Engine Details: Contains engine details

In the first window, general engine data such as manufacturer, type, limit load factor, weight, and CG coordinates are entered.

In the second window, shown in figure 17.2, propeller data is entered. This data includes manufacturer, weight, takeoff RPM, max continuous RPM, diameter, number of blades, and CG coordinates.

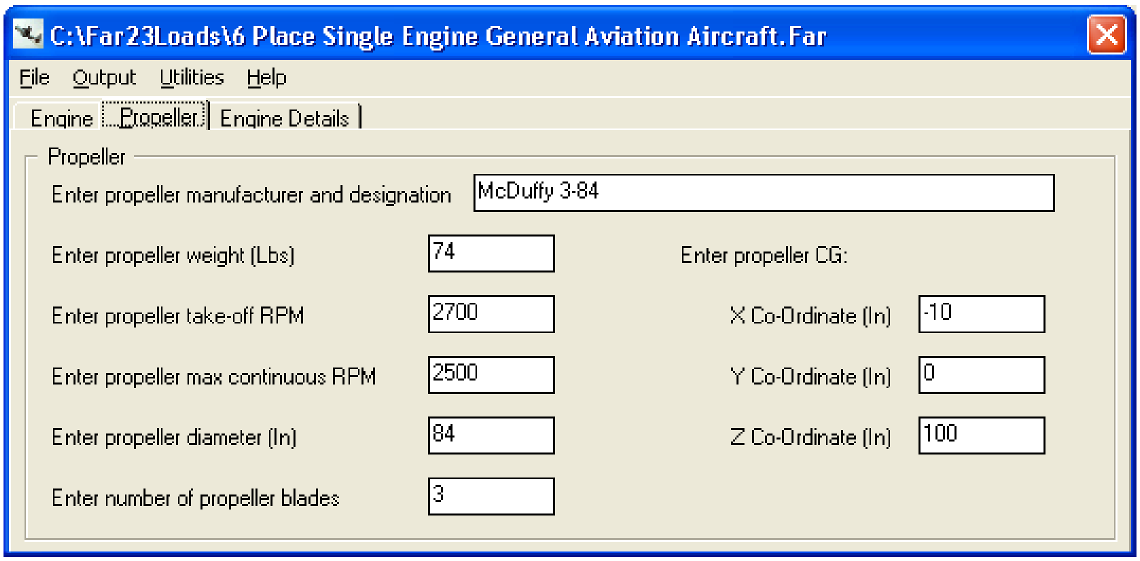


Figure 17.2 Engine Mount Loads Propeller Window

In the third window, shown in figure 17.3, further engine details such as takeoff HP, max continuous HP, and number of cylinders are entered. Depending on the type of engine selected on the Engine page, the required data changes. If the user selects ‘Reciprocal’ engine, engine details are as shown in figure 17.3; if the user selects ‘Turboprop’, engine details are as shown in figure 17.4.

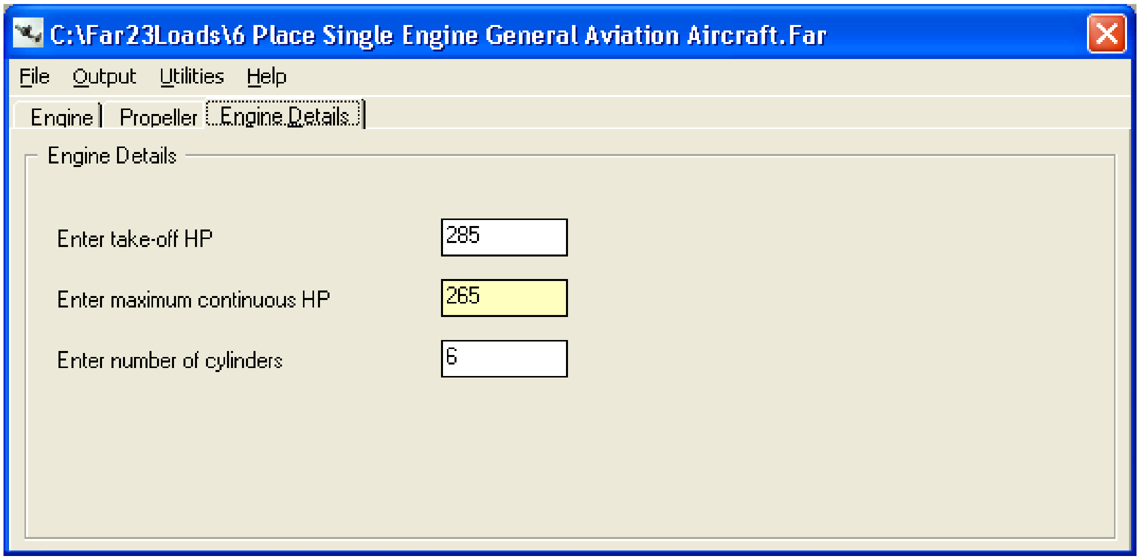


Figure 17.3 Engine Mount Loads Engine Details Window (Reciprocal Engine)

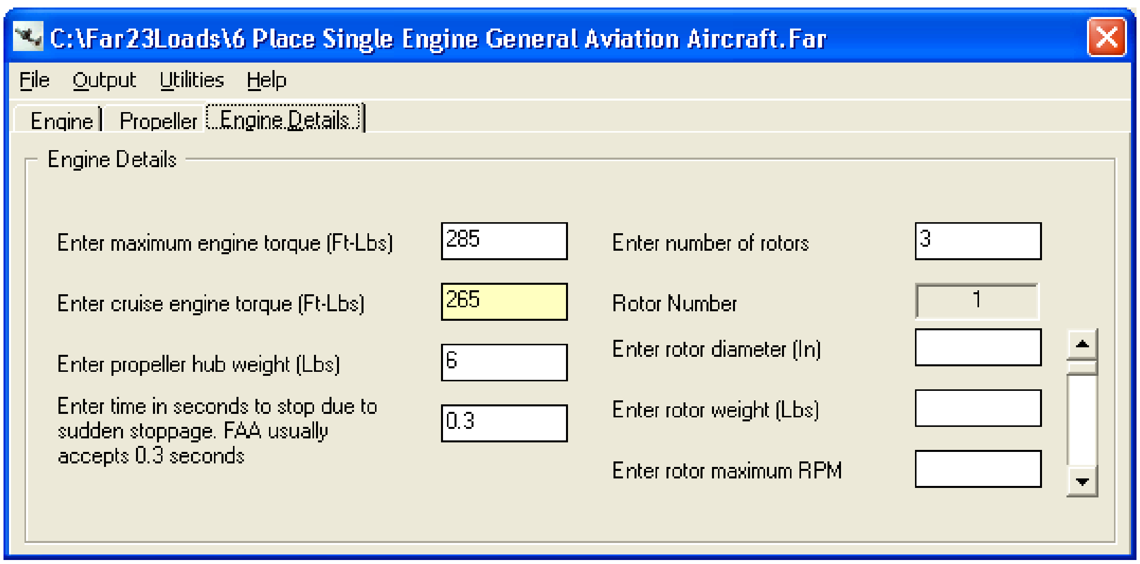


Figure 17.4 Engine Mount Loads Engine Details Window (Turboprop Engine)

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the engine mount loads module.

17.3 Engine Mount Loads Output

For reciprocating and turboprop engines, the output includes results for the following conditions:

- Limit takeoff torque with 75% limit maneuver vertical load factor
- A factor times the maximum continuous torque with 100% limit maneuver vertical load factor
- Side load independent of other flight loads

The results include the vertical load factor and load, the coordinates where the load acts, and the engine torque. For the side load, the vertical and side load factors are given, as well as the coordinates where the load acts. For each condition, the applicable FAR requirement is specified.

For turboprop engines, additional results are included in the output file. These include the loads for turboprop propeller malfunction, torque for sudden stoppage due to malfunction, and gyroscopic loads.

18 Landing Loads

18.1 Landing Loads Description

The ‘Landing Loads’ module calculates the landing loads for a tricycle landing gear with spring or oleo struts. The main and nose gears do not have to be the same type of gear.

The inputs required for calculation of the landing loads include the landing weight, landing gear load factor, assumed lift factor during landing, the station and waterline of the axles for the static position, rolling radius of the tires, distance between main wheels, tail down bump angle, and the weight and CG for the structural limits.

18.2 Running the Landing Loads Module

To run ‘Landing Loads’, click the module from the main window. The main input window appears as shown in figure 18.1. The values in textboxes with yellow background are updated automatically from respective modules.

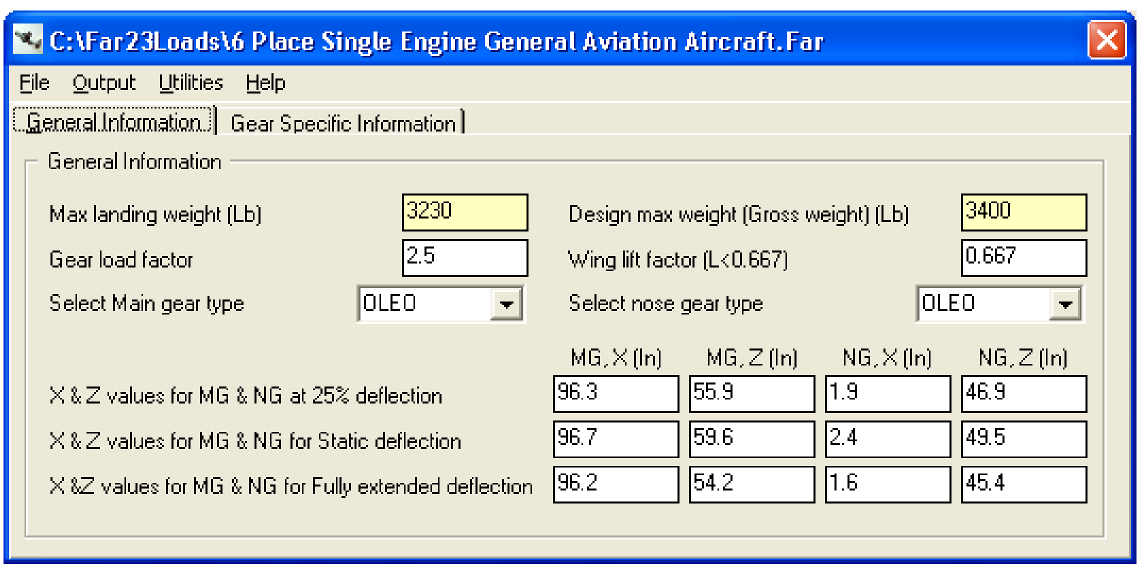


Figure 18.1 Landing Loads Main Window (General Information)

The main window has 2 data entry pages:

1. General Information: Contains general information about the landing gear

2. Gear Specific Information: Contains specific information about the landing gear

In the first window, shown in figure 18.1, maximum landing weight, design maximum (gross) weight, landing gear load factor, lift factor during landing, type of struts (oleo or spring), the wing station and waterline of the gear axles for the static deflection and fully extended deflection, for oleo struts, the wing station and waterline of the gear axles for the 25% deflection and for spring struts, and the wing station and waterline of the gear axles for the 25% deflection are entered.

In the second input window, shown in figure 18.2, rolling radius of the tires, distance between main wheels (tread), tail-down bump angle, and weight and CG for the structural limits are entered.

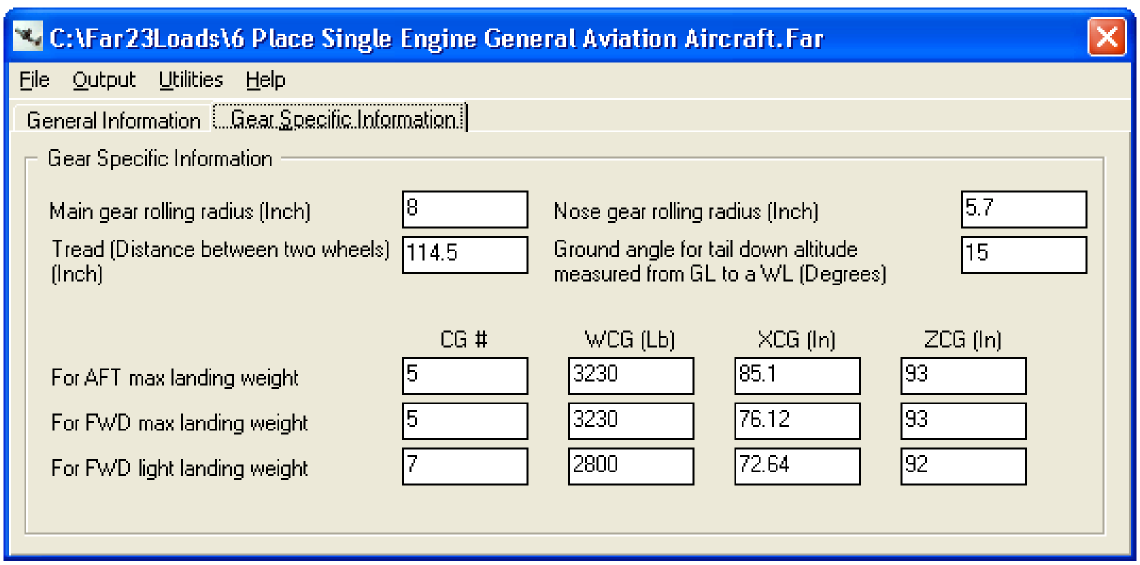


Figure 18.2 Landing Loads Main Window (Gear Specific Information)

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the landing loads module.

18.3 Landing Loads Output

In the output file of this module, the input data is listed. The type of landing gear is not given explicitly. For oleo struts the X and Z coordinates are given for the 25% deflection, while for the spring struts the coordinates are given for the 100% deflection.

The 'Landing Loads' module calculates the ground reactions and load factors as well as unbalanced moments. The ground reactions and load factors are given relative to the ground line and with respect to the airplane datum.

19 Landing Load Factor

19.1 Landing Load Factor Description

When test data is not available, the landing load factor is estimated in this module. This load factor is used in the calculation of landing loads. After test data is available, the load factor should be revised.

19.2 Running the Landing Load Factor Module

To run 'Landing Load Factor', click the module from the main window. The main input window appears as shown in figure 19.1. The values in textboxes with yellow background are updated automatically from respective modules.

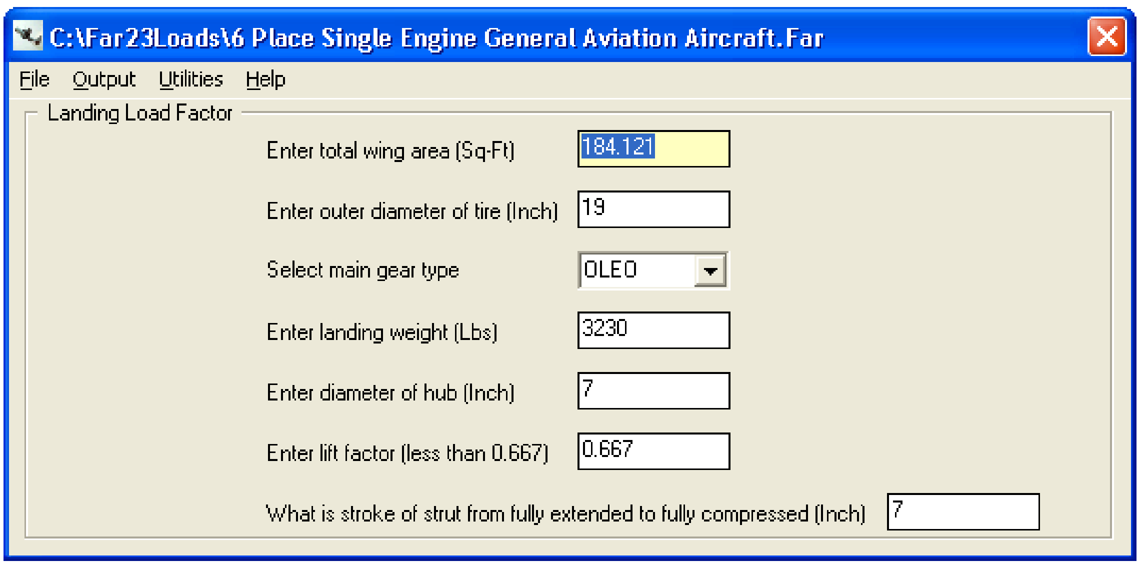


Figure 19.1 Landing Load Factor Main Input Window

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the landing load factor module.

19.3 Landing Load Factor Output

This module produces the following output:

- Sink rate
- Airplane load factor
- Landing gear load factor

The landing gear load factor is used in the ‘Landing Loads’ module when test data is not available.

20 Tail Loads Distribution

20.1 Tail Loads Description

The ‘Tail Loads Distribution’ module calculates chordwise distributions on the average chord for critical horizontal and vertical tail loads. It also calculates the chordwise distribution for any tail station for any of the critical loads. There are thirteen critical horizontal tail loads and four critical vertical tail loads.

20.2 Running the Tail Loads Module

To run the ‘Tail Loads Distribution’ module, click the module from the main window. The main input window appears as shown in figure 20.1. The first window allows the selection of 13 critical horizontal tail loads on average chord, 4 vertical tail loads on average chord, and 1 critical horizontal tail load or critical vertical tail load distributed on N station chords. It also provides the flexibility to return to the main menu by clicking on the close button.

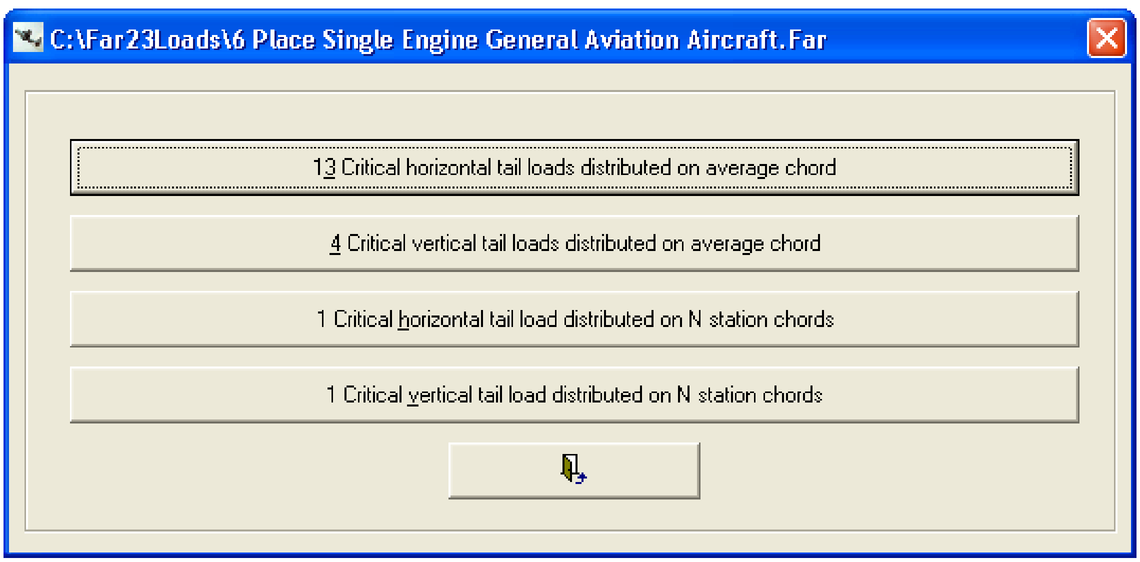


Figure 20.1 Tail Loads Distribution Main Window

Clicking on the first button opens the 13 critical horizontal tail loads submodule shown in figure 20.2. This submodule has 3 pages:

- Area: Contains information about different areas required to calculate critical loads as shown in figure 20.2

- Load Cases 1-6: Contains loads due to angle of attack at 25% MAC and due to camber at 50% MAC for cases 1 through 6 as shown in figure 20.3.
- Load Cases 7-13: Contains loads due to angle of attack at 25% MAC and due to camber at 50% MAC for cases 7 through 13 as shown in figure 20.4.

In the first window, the required input includes the area of the horizontal tail, area of the elevator, area of the elevator forward and aft of the hinge line, and the semispan of the tail.

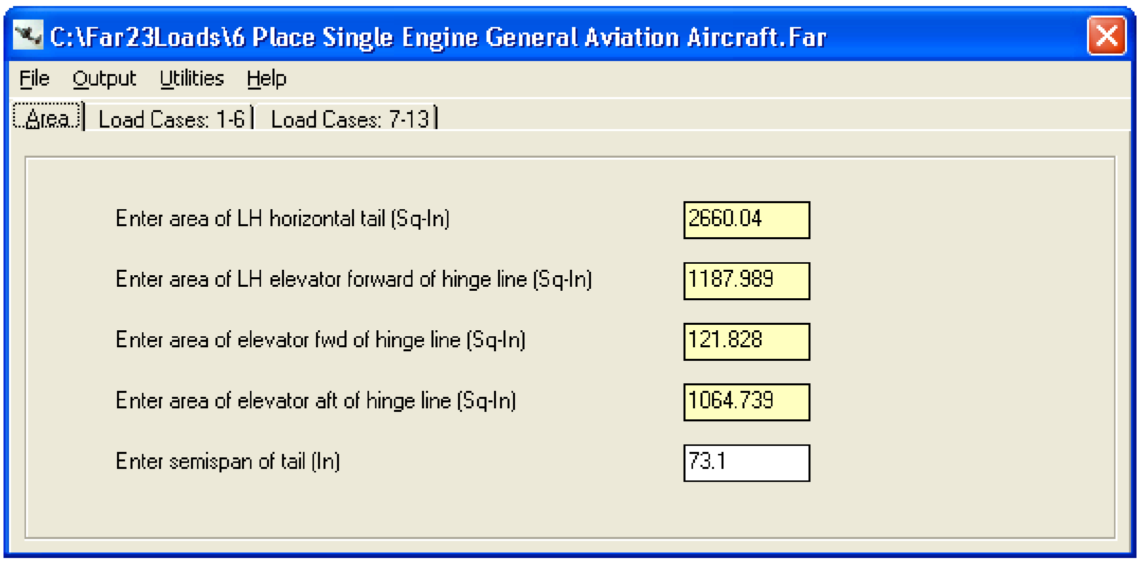


Figure 20.2 13 Critical Horizontal Tail Loads Window (Area Page)

In the second and third windows, the loads due to angle of attack at the 25% MAC and due to camber at the 50% MAC are entered for thirteen conditions. These conditions are:

- Up and down balancing tail loads with flaps retracted
- Up and down balancing tail loads with flaps extended
- Up and down unchecked maneuver tail load
- Up and down checked maneuver tail load
- Up and down gust tail load with flaps retracted
- Up and down gust tail load with flaps extended
- Unsymmetrical tail load

The loads data come from the critical horizontal tail loads output from the 'Select' module. For the gust tail loads, both the incremental and total loads are given in the 'Select' module. The total load is entered automatically. For the unsymmetrical tail load, the 'Select' file does not list the loads, but it does identify which case has the same distribution. The condition is not updated automatically.

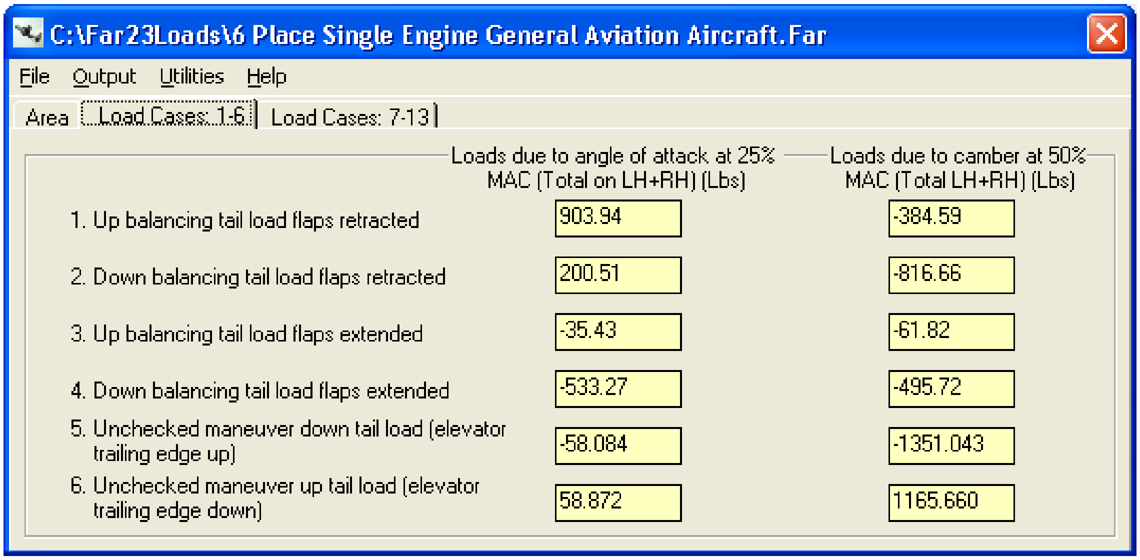


Figure 20.3 13 Critical Horizontal Tail Loads Window (Load Cases 1-6 Page)

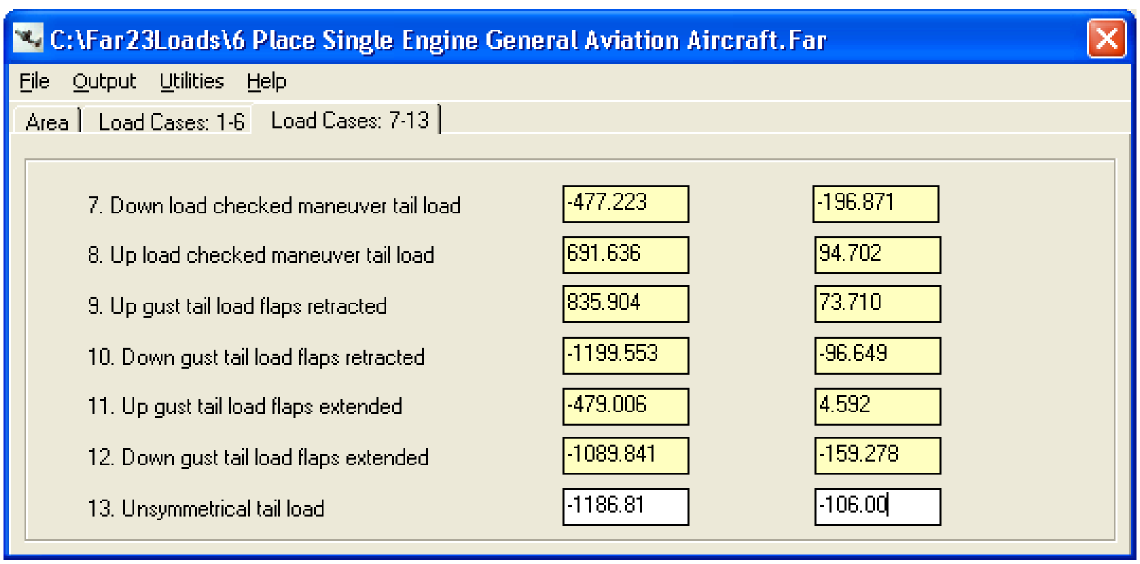


Figure 20.4 13 Critical Horizontal Tail Loads Window (Load Cases 7-13 Page)

Clicking on the second button in figure 20.1 opens the 4 critical vertical tail loads submodule shown in figure 20.5. This submodule has 2 pages:

- Area: Contains information about different areas required to calculate critical loads as shown in figure 20.5
- Loads: Contains loads due to angle of attack at 25% MAC and due to camber at 50% MAC as shown in figure 20.6

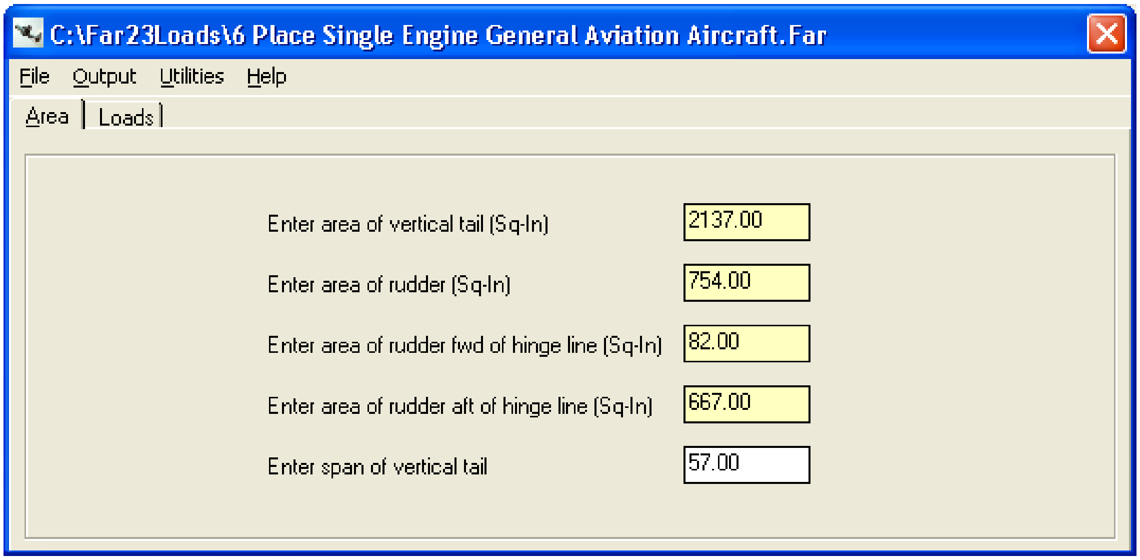


Figure 20.5 4 Critical Vertical Tail Loads Window (Area Page)

In the first window, the required input includes the area of the vertical tail, area of the rudder, area of the rudder forward and aft of the hinge line, and the span of vertical tail.

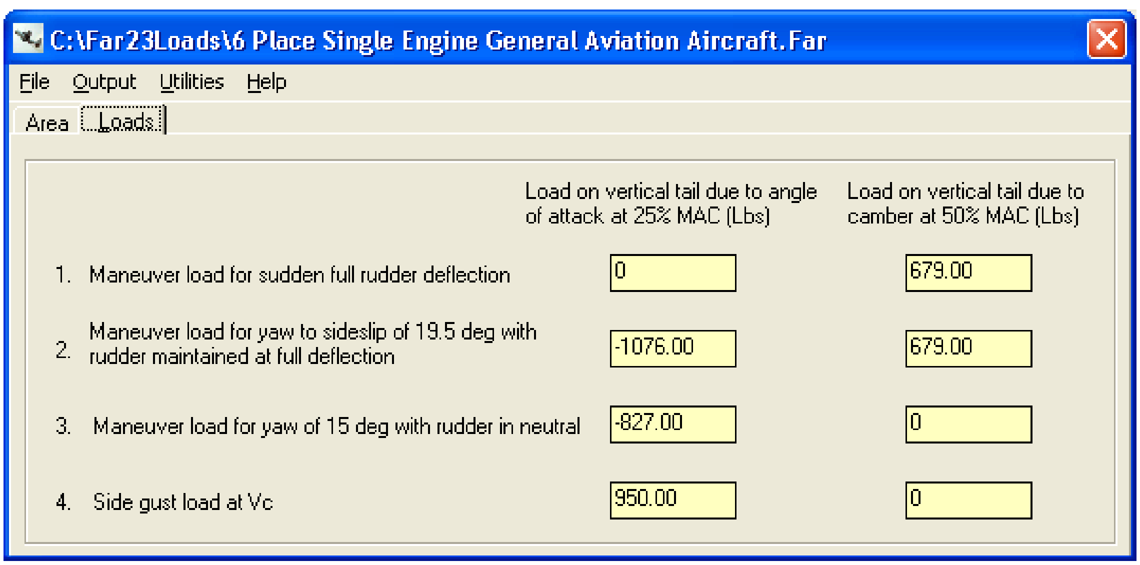


Figure 20.6 4 Critical Vertical Tail Loads Window (Loads Page)

In the second window, the loads due to angle of attack at the 25% MAC and due to camber at the 50% MAC are entered for four conditions. These conditions are:

- Maneuver loads for sudden full rudder deflection
- Maneuver loads for yaw to sideslip of 19.5° with rudder maintained at full deflection
- Maneuver load for yaw of 15° with rudder in neutral
- Side gust load at V_C

The loads data are transferred automatically from the critical vertical tail loads output file from the 'Select' module. Some of the loads may be zero. For zero loads a numerical value of 0 should be entered.

Clicking on the third button in figure 20.1 opens the 1 critical horizontal tail loads submodule shown in figure 20.7. This submodule has 2 pages:

- General: Contains information about the critical horizontal tail load to be analyzed, different respective loads, and areas as shown in figure 20.7
- Loads: Contains data for each butt line station and respective loads data as shown in figure 20.8

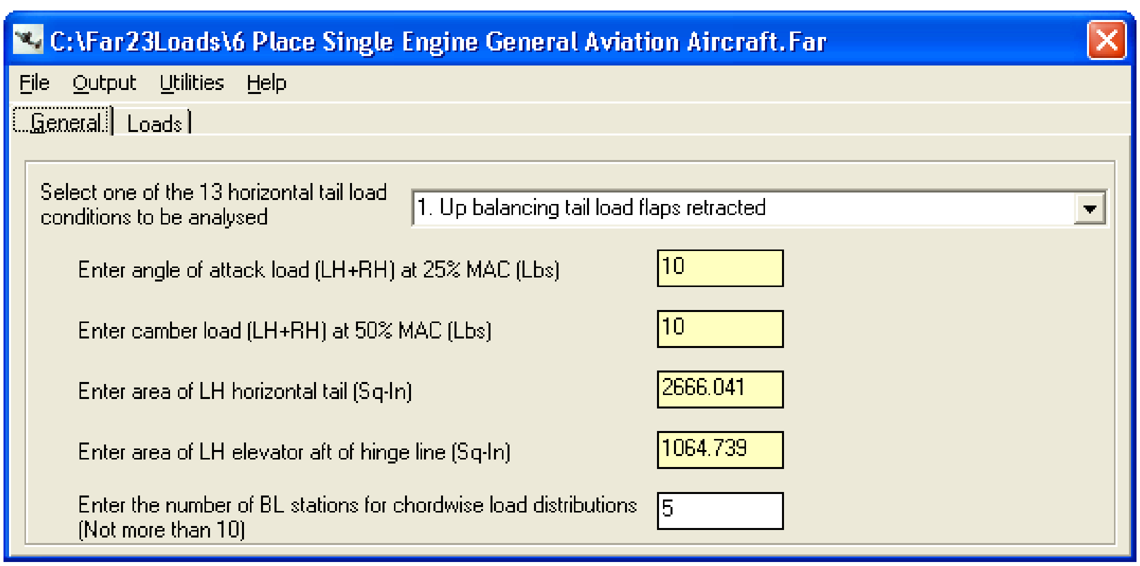


Figure 20.7 1 Critical Horizontal Tail Load Window (General Page)

In the first window, the name of the load condition, the angle of attack load at 25% MAC, the camber load at 50% MAC, area of one side of the horizontal tail, area of the LH elevator aft of hinge line, and the number of butt line stations for chordwise distribution are the required inputs. The values in textboxes with yellow background are updated automatically when the load condition is selected or changed by the user.

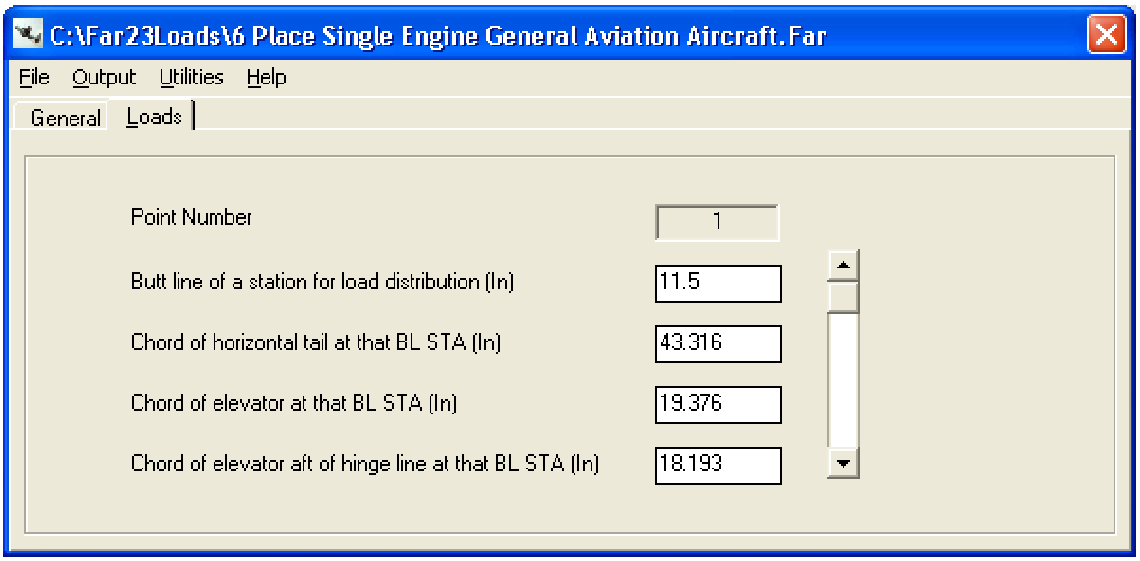


Figure 20.8 1 Critical Horizontal Tail Load Window (Loads Page)

In the second window, data for each butt line station is entered in inches. The data required is as follows:

- Butt line of the station
- Chord of horizontal tail at the butt line station
- Chord of elevator at the butt line station
- Chord of elevator aft of the hinge line at the butt line station

The number of inputs for the above mentioned data depends upon the number of BL stations entered in the ‘General’ page. The maximum value allowed is 10.

Clicking on the fourth button in figure 20.1 opens the 1 critical vertical tail loads submodule shown in figure 20.9. This submodule has 2 pages:

- General: Contains information about the critical vertical load to be analyzed, different respective loads, and areas as shown in figure 20.9
- Loads: Contains data for each water line station and respective loads data as shown in figure 20.10

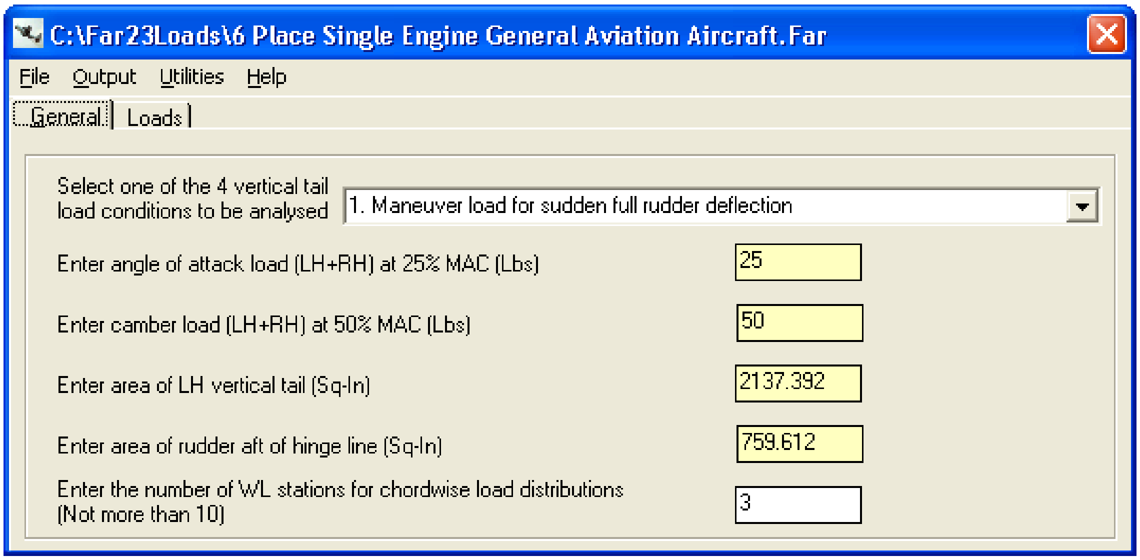


Figure 20.9 1 Critical Vertical Tail Load Window (General Page)

In the first window, the name of the load condition, the angle of attack load at 25% MAC, the camber load at 50% MAC, area of one side of the vertical tail, area of the rudder aft of hinge line, and the number of water line stations for chordwise distribution are the required inputs. The values in textboxes with yellow background are updated automatically when the load condition is selected or changed by the user.

In the second window, data for each water line station is entered in inches. The data required is as follows:

- Water line of the station
- Chord of vertical tail at the water line station
- Chord of rudder at the water line station
- Chord of rudder aft of the hinge line at the water line station

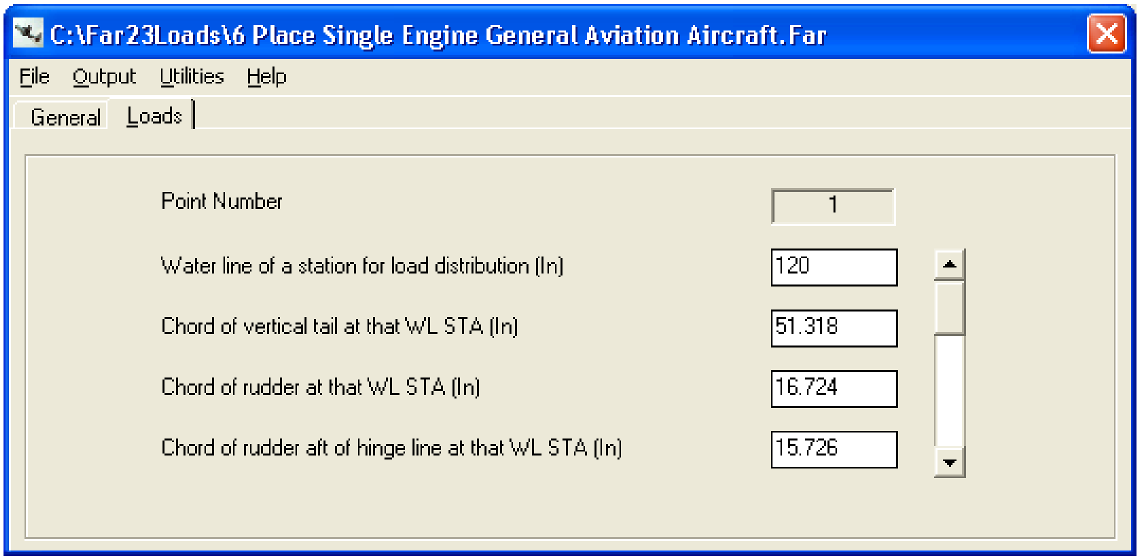


Figure 20.10 1 Critical Vertical Tail Load Window (Loads Page)

The number of inputs for the above mentioned data depends upon the number of WL stations entered in the 'General' page. The maximum value allowed is 10.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- File:
    - Return to Tail Loads Menu – Enables user to close existing menu and return to the main window of tail loads module from which other submodules can be selected and run or the user can exit the tail loads distribution module.
- Output: The Output menu is used to preview, save and print output and also to update input / output data to other modules. The Output menu has 5 sub menus in 13 critical horizontal tail loads and 4 critical vertical tail loads and has 4 sub menus in 1 critical horizontal tail loads and 1 critical vertical tail load submodule. Details are as follows:
    - Update Output To Other Modules (available in 13 critical horizontal and 4 critical vertical tail loads submodules only) – Updates output to respective modules. Refer to table 2.1 for details of data flow between modules. After the data is updated, a message box displaying the names of modules to which data has been updated appears.
- Help: The Help menu has only one option. It provides help on how to use the tail loads module.

20.3 Tail Loads Output

The results of this module are the chordwise distribution of critical loads for each of the critical load conditions. For the horizontal tail, there are 13 critical conditions; for the vertical tail, there are 4 critical conditions. The load distribution is given at five chordwise stations. In the output, LT25 refers to the load due to angle of attack at the 25% MAC and the LT50 refers to the load due to camber at the 50% MAC. This is an echo of the input.

21 Tab Loads

21.1 Tab Loads Description

The tab loads for the wing, horizontal tail, and vertical tail are determined in the 'Tab Loads' module. The loads are calculated for full deflection at VC at the shoulder on the structural limit diagram. This is where V_C is the highest in equivalent airspeed (KEAS) with the greatest Mach number.

21.2 Running the Tab Loads Module

To run 'Tab Loads', click the module from the main window. The main input window appears as shown in figure 21.1. The values in textboxes with yellow background are updated automatically from respective modules.

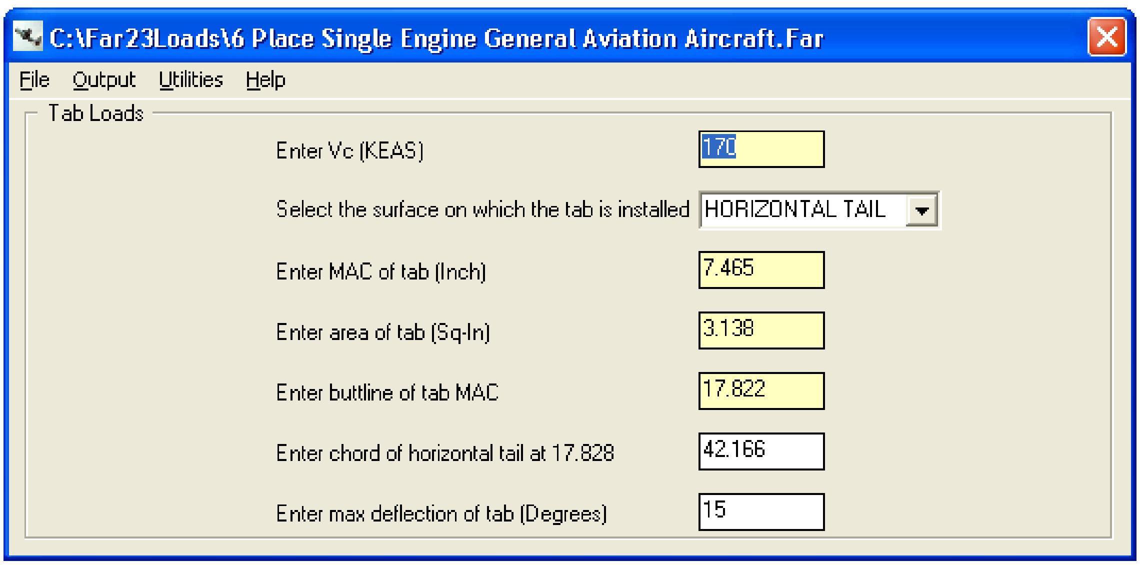


Figure 21.1 Tab Loads Main Window

The inputs required for tab loads calculations include:

- Cruise speed VC (KEAS)
- Surface of interest (wing, horizontal tail or vertical tail)
- The mean aerodynamic chord (MAC) for the tab (inches)
- The area of the tab (in^2)
- Butt line of the MAC of the tab (inches)

- Chord of wind (inches)
- Maximum deflection of the tab (degrees)

Depending upon the surface selected, the prompts on the input window will change.

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the tab loads module.

21.3 Tab Loads Output

The output from this module includes the ratio of the tab to airfoil chord, the tab load in pounds, and the tab pressure at the leading and trailing edges.

22 One Engine Out Loads

22.1 One Engine Out Description

The loads for one engine out are calculated using this module. When one of the engines fail, the primary force acting on the airplane is an unbalanced moment about the vertical axis at the center of gravity of the airplane. The rotational acceleration of the airplane about the vertical axis at the CG is resisted by the mass moment of inertia of the airplane. As the airplane rotates, the vertical tail provides an aerodynamic force that also resists the unbalanced moment.

22.2 Running the One Engine Out Loads Module

To run the 'One Engine Out Loads' module, click the module from the main window. The main input window appears as shown in figure 22.1. The values in textboxes with yellow background are updated automatically from respective modules.

The input for this module is entered on two pages as shown in figures 22.1 and 22.2. On page 1, the following data is entered:

- Aspect ratio of vertical tail
- Area of vertical tail (in^2)
- Area of rudder (in^2)
- Maximum deflection of rudder (degrees)
- Fuselage station of the CG (inches)
- Fuselage station for the 25% MAC of the vertical tail (inches)
- Fuselage station for the 50% MAC of the vertical tail (inches)
- Speed (KEAS)
- Altitude (feet)
- Moment of inertia of the airplane about vertical axis, I_{ZZ} (slug-ft^2)

On page 2, the following data is entered:

- Butt line of the engine (inches)
- Maximum horsepower of one engine
- Propeller diameter (ft)
- Time at which thrust decays to zero (seconds)

- Time at which windmill drag builds to maximum (seconds)
- Time to develop full rudder deflection (seconds)
- Incremental time step (seconds) (suggested as 0.05)

Separate analyses are required for each speed. These include V_C, V_D and V_S.

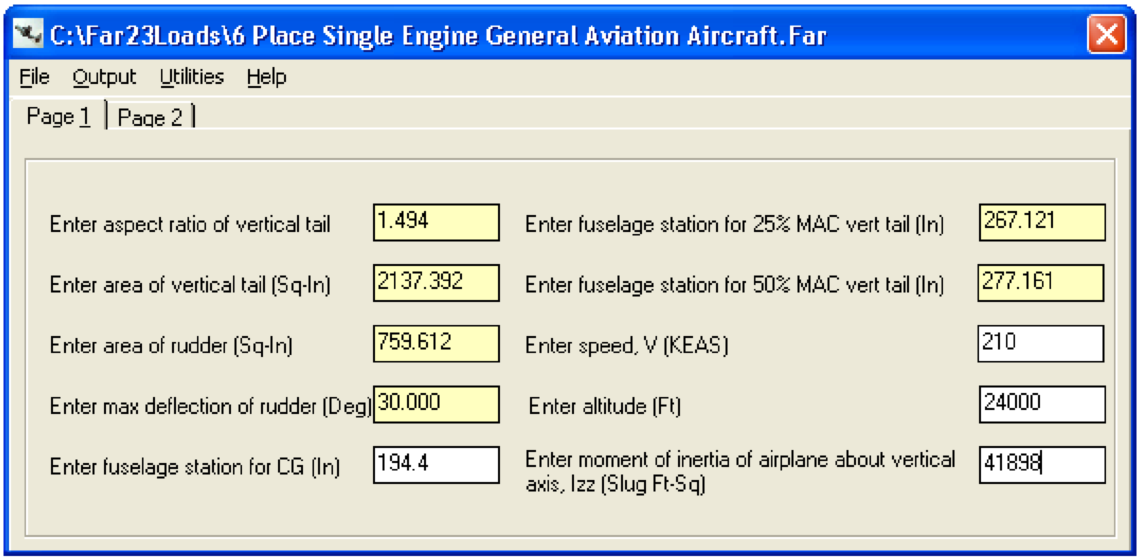


Figure 22.1 One Engine Out Loads Main Window (Page 1)

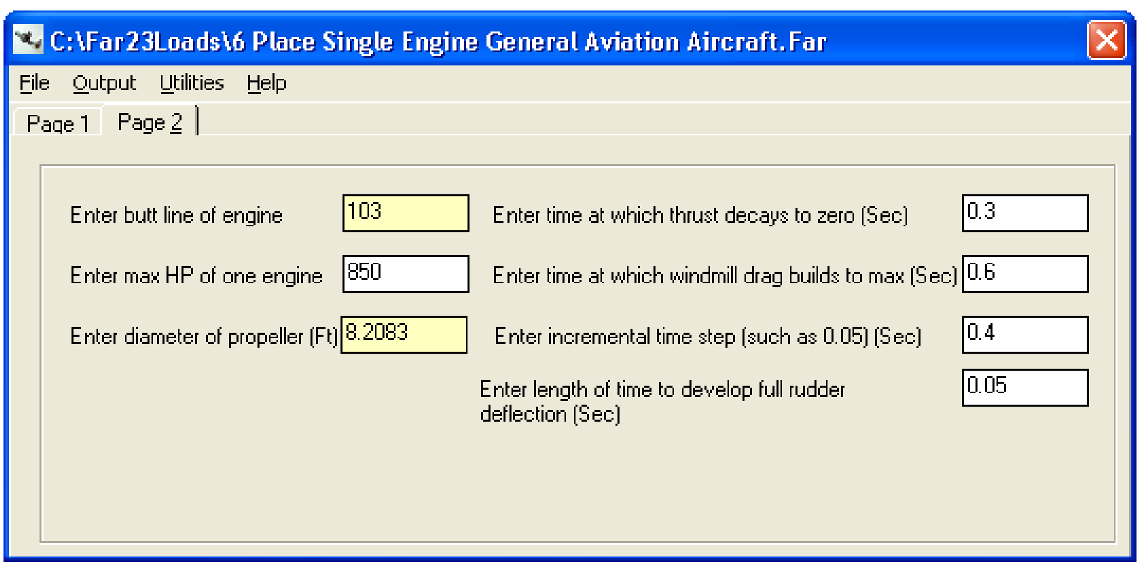


Figure 22.2 One Engine Out Loads Main Window (Page 2)

The input window is displayed when the module starts and is used to specify the parameters for the analysis. The window includes 4 visible and 1 invisible menu option(s). Details of the visible menus are described in Section 2 under the headings File, Output, Utilities, and Input. Additional module menu items include:

- Help: The Help menu has only one option. It provides help on how to use the one engine out loads module.

22.3 One Engine Out Loads Output

The output of this module includes the engine thrust in pounds, the windmill drag in pounds, the maximum yawing velocity, and maximum tail load. The time history provides the data from time zero until recovery is complete at the specified increments. At each time increment, several parameters are printed. These parameters along with the variable names are listed below:

- Time (TIME)
- Yaw angle (THETA)
- Angular velocity (THETADOT)
- Angular acceleration about the CG (THETA2DOT)
- Load at the 25% MAC (LT25)
- Load at the 50% MAC (LT50)
- Total Load (LT)
- Rudder deflection (degrees) (RUD DEFL)
- Moment (MOMENT)

In the time history, it is indicated when the engine thrust has decayed to zero, when the windmill drag has built up to the maximum, when corrective action is initiated, and when recovery is complete.

23 References

1. McMaster, Hal C., “FAR 23 Loads,” Aero Science Software, Wichita, KS, 1996.
2. Code of Federal Regulations, Title 14, Parts 1 to 59, Aeronautics Chapter I – Federal Aviation Administration, Subchapter C – Aircraft, Part 23 – Airworthiness Standards: Normal, Utility, Acrobatic and Commuter Category Airplanes, Subpart C- Structures, Revised as of January 1, 1994.
3. Pope, Alan, “Basic Wing and Airfoil Theory,” McGraw-Hill Book Company, 1951.
4. User’s Guide for FAR23 Loads Program, Final Report, March 1997.

Appendix A Input/Output Data for a 6 Place General Aviation Airplane

A.1 Weight Estimation

```
ESTIMATED WEIGHT DATA FOR 6 PLACE SINGLE ENGINE GENERAL AVIATION AIRPLANE

INPUT

MAX CONTINUOUS HP          265
NUMBER OF ENGINES          1
NUMBER OF SEATS            6
HOURS AT CRUISE POWER      3
MAX BAGGAGE WEIGHT         0
UNPRESSURIZED
RECIPROCAL 4 CYCLE ENGINE

OUTPUT

MAX TAKE OFF WT            3468
USEFUL LOAD                1318
EMPTY WEIGHT               2150
W(EMPTY)/W(TO)             .62

WING                       359
FUSELAGE                   340
TAIL                       81
NACELLE                    50
LANDING GEAR               198
CONTROLS                   52
TOTAL STRUCTURE.....       1081

ENGINE INSTALLED           490
(INCLUDES PROPELLER(S))   ( 83)
FUEL SYSTEM                52
EXHAUST                    72
OTHER ENGINE DETAILS       86
TOTAL POWRPLANT.....       700

INSTRUMENTS & NAV EQUIP    15
PNEUMATICS                 3
ELECTRICAL                 83
ELECTRONICS                0
FURNISHINGS & EQUIPMENT    152
ENVIRONMENTAL & ANTI-ICE   10
MISC OTHER SYSTEM WT       0
TOTAL SYSTEMS WEIGHT..... 268

OPTIONS & MISCELLANEOUS    99


EMPTY WEIGHT..........     2150

PILOT                      170
PASSENGER NO.  2           170
PASSENGER NO.  3           170
PASSENGER NO.  4           170
PASSENGER NO.  5           170
PASSENGER NO.  6           170
BAGGAGE                    0
FUEL                       298
USEFUL LOAD                1318
```

A.2 Weight and CG

AIRPLANE WEIGHT AND INERTIA

ITEM	COMPONENT	WEIGHT	X	Y	Z	IXX	IYY	IZZ
1	WING OUTBOARD	330.000	97.870	0.000	87.730	4444110.000	133485.000	4444110.000
2	HORIZONTAL TAIL	42.000	270.360	0.000	111.000	0.000	0.000	0.000
3	VERTICAL TAIL	23.000	276.930	0.000	137.760	0.000	0.000	0.000
4	MAIN GEAR WHEEL	45.000	97.000	0.000	69.000	0.000	0.000	0.000
5	MAIN GEAR STRUT	110.000	97.000	0.000	78.000	0.000	0.000	0.000
6	NOSE GEAR WHEEL	9.000	1.000	0.000	52.000	0.000	0.000	0.000
7	NOSE GEAR STRUT	40.000	1.000	0.000	65.000	0.000	0.000	0.000
8	FLIGHT CONTROL	57.000	123.000	0.000	105.000	0.000	0.000	0.000
9	NACELLE	62.000	21.000	0.000	92.000	0.000	0.000	0.000
10	ENGINE INSTALL	505.000	22.000	0.000	92.000	52604.000	81473.000	81473.000
11	PROPELLER	74.000	-10.000	0.000	100.000	0.000	0.000	0.000
12	SYSTEMS	88.000	60.000	0.000	100.000	0.000	0.000	0.000
13	FURINSHINGS	175.000	105.000	0.000	100.000	0.000	0.000	0.000
14	UNUSABLE FUEL	12.000	73.000	0.000	80.000	0.000	0.000	0.000
15	FUSELAGE STRUC	250.000	99.000	0.000	80.000	0.000	1131020.000	1131020.000
	---------- END OF EMPTY WEIGHT ITEMS ----------							
51	PILOT	170.000	75.000	0.000	100.000	28730.000	24480.000	4250.000
52	30 MIN FUEL	71.000	70.000	0.000	82.000	86975.000	0.000	86975.000
	---------- END OF MINIMUM WEIGHT ITEMS ----------							
61	COPILOT	170.000	75.000	0.000	100.000	28730.000	24480.000	4250.000
62	3RD PERSON	170.000	111.000	0.000	100.000	28730.000	24480.000	4250.000
63	4TH PERSON	170.000	111.000	0.000	100.000	28730.000	24480.000	4250.000
64	5TH PERSON	170.000	150.000	0.000	100.000	28730.000	24480.000	4250.000
65	6TH PERSON	170.000	150.000	0.000	100.000	28730.000	24480.000	4250.000
67	GUEL TO GR WT	409.000	70.000	0.000	87.000	501025.000	0.000	501025.000
68	BALLAST	78.000	103.700	0.000	90.000	0.000	0.000	0.000
	---------- END OF DISCRETIONARY WEIGHT ITEMS ----------							

CENTER OF GRAVITY, WEIGHT & INERTIA

CENTER OF GRAVITY AND WEIGHT

XBAR (FUS STA) = 84.999

ZBAR (WATERLINE) = 92.579

WEIGHT (POUNDS) = 3400.000

INERTIAS WITH RESPECT TO AIRPLANE COORDINATES

IXX	IYY	IZZ	IXZ	UNITS
1201.530	2058.218	3022.772	134.403	SLUG FEET SQUARED
5566065.738	9534655.502	14002928.764	622619.791	LBS INCHES SQUARED

INERTIAS WITH RESPECT TO PRINCIPAL AXES

IX(P)	IY(P)	IZ(P)	UNITS
1191.665	2058.218	3032.637	SLUG FT SQUARED
5520365.471	9534655.502	14048629.031	LBS INCHES SQUARED

THETA = 4.198 (DEGREES, MEASURED UP FROM WL & AFT FROM CG)

A.3 Envelope of Loadings

AIRPLANE ENVELOPE OF WEIGHT VS CENTER OF GRAVITY FOR DISCRETIONARY LOAD

WEIGHT DATA FOR 7

ITEM	COMPONENT	WEIGHT	X	Y	Z
1	WING OUTBOARD	330.000	97.870	0.000	87.730
2	HORIZONTAL TAIL	42.000	270.360	0.000	111.000
3	VERTICAL TAIL	23.000	276.930	0.000	137.760
4	MAIN GEAR WHEEL	45.000	97.000	0.000	69.000
5	MAIN GEAR STRUT	110.000	97.000	0.000	78.000
6	NOSE GEAR WHEEL	9.000	1.000	0.000	52.000
7	NOSE GEAR STRUT	40.000	1.000	0.000	65.000
8	FLIGHT CONTROL	57.000	123.000	0.000	105.000
9	NACELLE	62.000	21.000	0.000	92.000
10	ENGINE INSTALL	505.000	22.000	0.000	92.000
11	PROPELLER	74.000	-10.000	0.000	100.000
12	SYSTEMS	88.000	60.000	0.000	100.000
13	FURINSHINGS	175.000	105.000	0.000	100.000
14	UNUSABLE FUEL	12.000	73.000	0.000	80.000
15	FUSELAGE STRUC	250.000	99.000	0.000	80.000
--- END OF EMPTY WEIGHT ITEMS---					
51	PILOT	170.000	75.000	0.000	100.000
52	30 MIN FUEL	71.000	70.000	0.000	82.000
--- END OF MINIMUM WEIGHT ITEMS---					
61	COPILOT	170.000	75.000	0.000	100.000
62	3RD PERSON	170.000	111.000	0.000	100.000
63	4TH PERSON	170.000	111.000	0.000	100.000
64	5TH PERSON	170.000	150.000	0.000	100.000
65	6TH PERSON	170.000	150.000	0.000	100.000
66	BAGGAGE	120.000	180.000	0.000	110.000
67	FUEL TO FULL	409.000	70.000	0.000	87.000
--- END OF DISCRETIONARY WEIGHT ITEMS ---					

ENVELOPE OF DISCRETIONARY LOAD FOR 7

ADDED	XBAR	ZBAR	WEIGHT
MINIMUM WEIGHT	73.092	90.725	2063.000
Fuel to Full	72.581	90.109	2472.000
Copilot	72.736	90.745	2642.000
4th Person	75.050	91.305	2812.000
3rd Person	77.099	91.800	2982.000
6th Person	81.031	92.243	3152.000
5th Person	84.560	92.639	3322.000
Baggage	87.888	93.245	3442.000
MINIMUM WEIGHT	73.092	90.725	2063.000
Baggage	78.969	91.784	2183.000
5th Person	84.101	92.378	2353.000
6th Person	88.541	92.892	2523.000
3rd Person	89.959	93.340	2693.000
4th Person	91.208	93.736	2863.000
Copilot	90.300	94.087	3033.000
Fuel to Full	87.888	93.245	3442.000

A.4 Wing Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

WING GEOMETRY
SYMMETRICAL ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	45.000	0.000
2	64.313	46.500
3	72.000	201.000

COORDINATES OF TRAILING EDGE

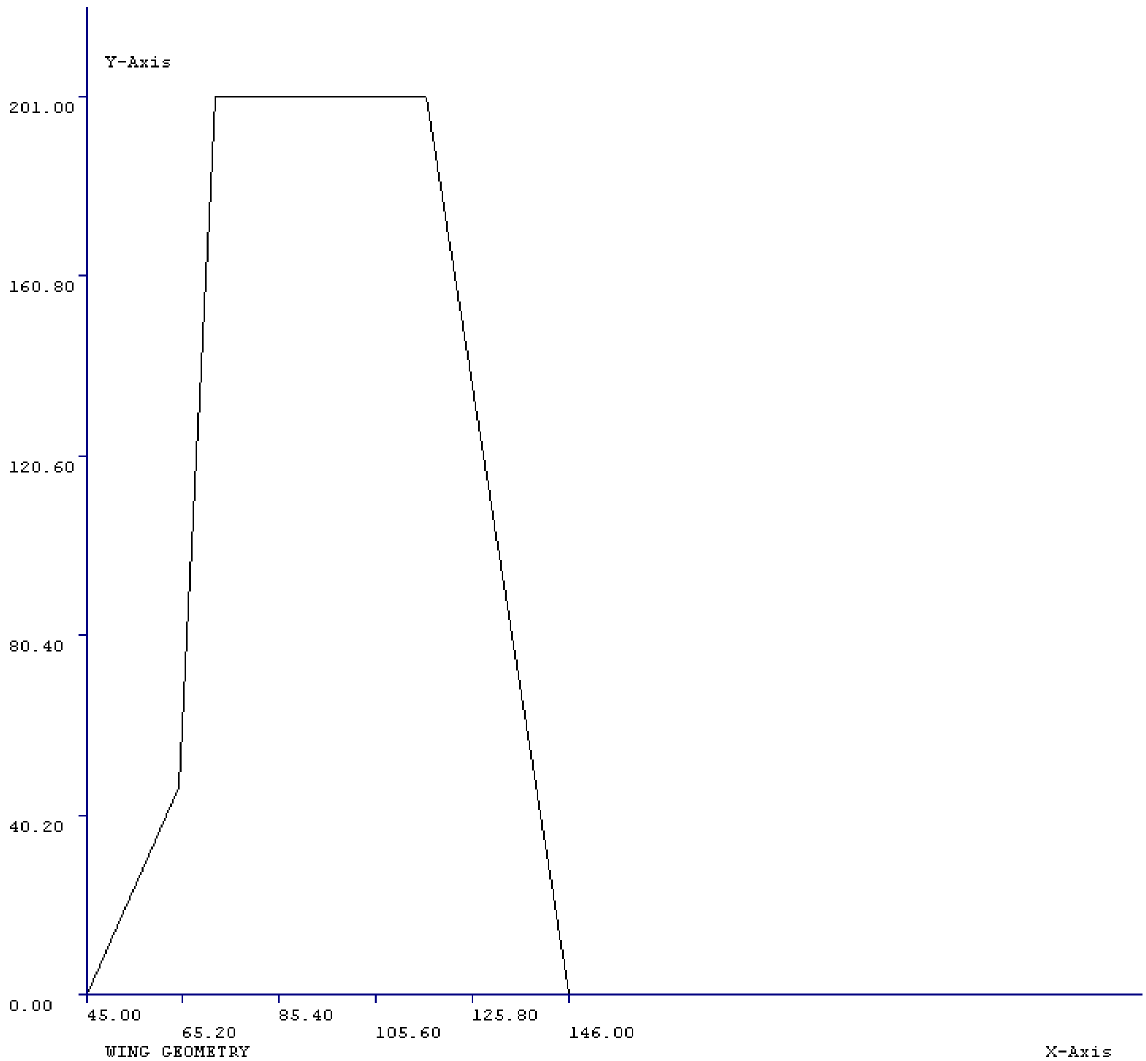
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	146.000	0.000
2	116.000	201.000

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
13256.722	69.246	87.854	63.641	6.095

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	47.087	145.250	5.025	98.163	986.538
2	51.261	143.750	15.075	92.489	929.513
3	55.435	142.250	25.125	86.815	872.488
4	59.609	140.750	35.175	81.141	815.463
5	63.783	139.250	45.225	75.467	758.439
6	64.750	137.750	55.275	73.000	733.654
7	65.250	136.250	65.325	71.000	713.554
8	65.750	134.750	75.375	69.000	693.453
9	66.250	133.250	85.425	67.000	673.353
10	66.750	131.750	95.475	65.000	653.253
11	67.250	130.250	105.525	63.000	633.153
12	67.750	128.750	115.575	61.000	613.052
13	68.250	127.250	125.625	59.000	592.952
14	68.750	125.750	135.675	57.000	572.852
15	69.250	124.250	145.725	55.000	552.752
16	69.750	122.750	155.775	53.000	532.651
17	70.250	121.250	165.825	51.000	512.551
18	70.750	119.750	175.875	49.000	492.451
19	71.250	118.250	185.925	47.000	472.350
20	71.750	116.750	195.975	45.000	452.250



Y-Axis
201.00
160.80
120.60
80.40
40.20
0.00
45.00
65.20
85.40
105.60
125.80
146.00
WING GEOMETRY
X-Axis

A.5 Aileron Geometry

```
AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

AILERON GEOMETRY
NOT SYM ABOUT CL

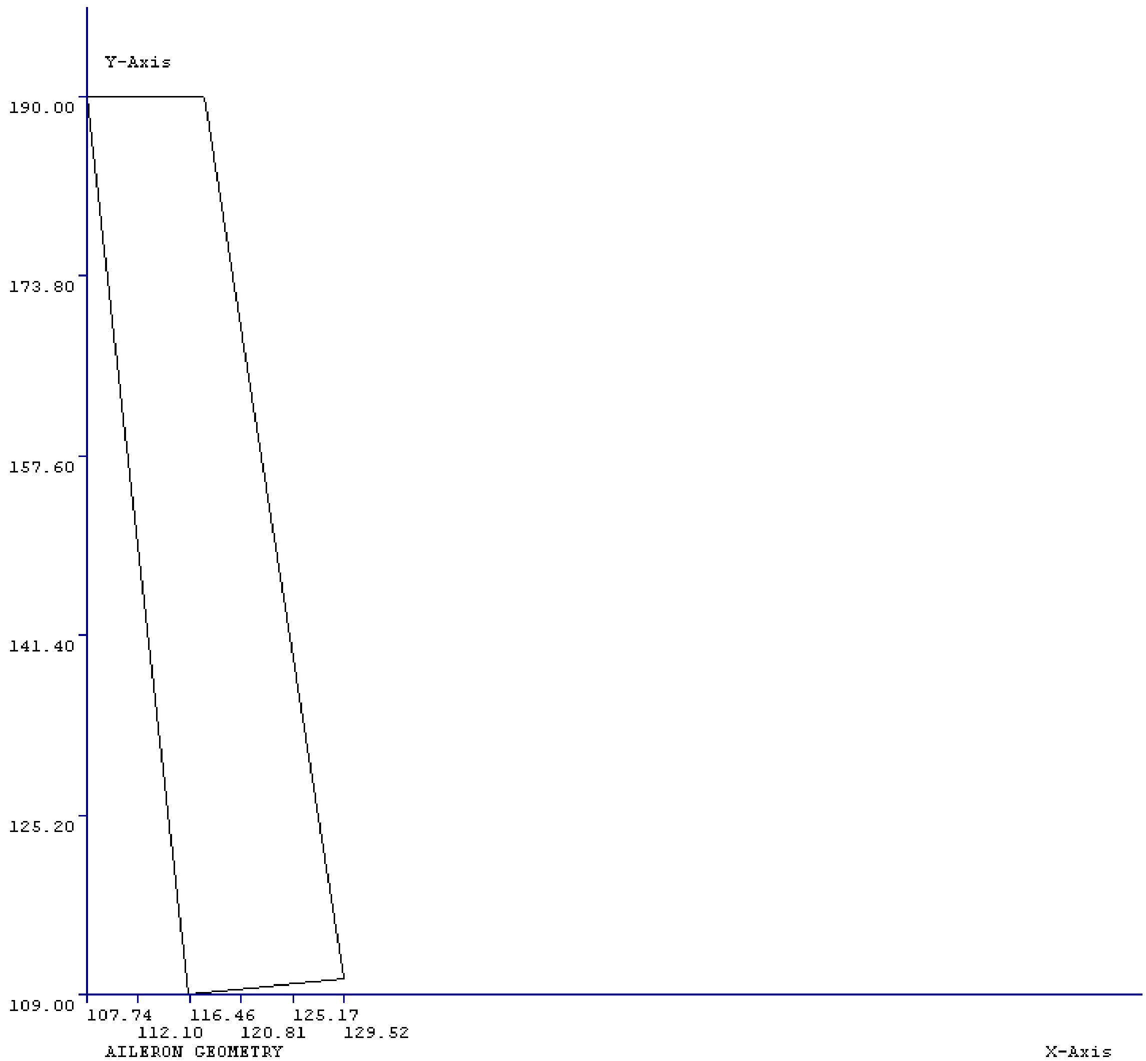
COORDINATES OF LEADING EDGE
POINT NO   FUS STA (XLE)   WING STA (YLE)
 1            116.380         109.000
 2            107.744         190.000

COORDINATES OF TRAILING EDGE
POINT NO   FUS STA (XTE)   WING STA (YTE)
 1            116.380         109.000
 2            129.522         110.401
 3            117.642         190.000

THE SURFACE IS DIVIDED INTO  20 INCREMENTS OF DY

AREA/SIDE         MAC    YLE(MAC)   XLE(MAC)   ASPECT RATIO
   941.589     11.710    147.500    112.275      6.968

ELEMENT DATA
ELEM       XLE        XTE         Y          C         AREA
   1    116.164    129.429    111.025     13.265     53.722
   2    115.732    128.824    115.075     13.092     53.023
   3    115.301    128.220    119.125     12.920     52.324
   4    114.869    127.616    123.175     12.747     51.625
   5    114.437    127.011    127.225     12.574     50.926
   6    114.005    126.407    131.275     12.402     50.226
   7    113.573    125.802    135.325     12.229     49.527
   8    113.142    125.198    139.375     12.056     48.828
   9    112.710    124.593    143.425     11.884     48.128
  10    112.278    123.989    147.475     11.711     47.429
  11    111.846    123.384    151.525     11.538     46.730
  12    111.414    122.780    155.575     11.366     46.031
  13    110.983    122.175    159.625     11.193     45.331
  14    110.551    121.571    163.675     11.020     44.632
  15    110.119    120.967    167.725     10.848     43.933
  16    109.687    120.362    171.775     10.675     43.233
  17    109.255    119.758    175.825     10.502     42.534
  18    108.824    119.153    179.875     10.330     41.835
  19    108.392    118.549    183.925     10.157     41.136
  20    107.960    117.944    187.975      9.984     40.436
```


Y-Axis
190.00
173.80
157.60
141.40
125.20
109.00
107.74
112.10
116.46
120.81
125.17
129.52
AILERON GEOMETRY
X-Axis

A.6 Aileron Forward of Hinge Line Geometry

```
AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

AILERON FORWARD OF HINGE GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE
POINT NO   FUS STA (XLE)   WING STA (YLE)
 1            116.380         109.000
 2            107.744         190.000

COORDINATES OF TRAILING EDGE
POINT NO   FUS STA (XTE)   WING STA (YTE)
 1            116.380         109.000
 2            118.993         109.279
 3            109.705         190.000

THE SURFACE IS DIVIDED INTO  20 INCREMENTS OF DY

AREA/SIDE         MAC    YLE(MAC)   XLE(MAC)   ASPECT RATIO
   186.547      2.320    147.500    112.275     35.171

ELEMENT DATA
ELEM       XLE        XTE        Y          C         AREA
   1    116.164    118.792    111.025      2.628     10.643
   2    115.732    118.326    115.075      2.594     10.505
   3    115.301    117.860    119.125      2.560     10.366
   4    114.869    117.394    123.175      2.525     10.228
   5    114.437    116.928    127.225      2.491     10.089
   6    114.005    116.462    131.275      2.457      9.951
   7    113.573    115.996    135.325      2.423      9.812
   8    113.142    115.530    139.375      2.389      9.674
   9    112.710    115.064    143.425      2.354      9.535
  10    112.278    114.598    147.475      2.320      9.397
  11    111.846    114.132    151.525      2.286      9.258
  12    111.414    113.666    155.575      2.252      9.119
  13    110.983    113.200    159.625      2.218      8.981
  14    110.551    112.734    163.675      2.183      8.842
  15    110.119    112.268    167.725      2.149      8.704
  16    109.687    111.802    171.775      2.115      8.565
  17    109.255    111.336    175.825      2.081      8.427
  18    108.824    110.870    179.875      2.047      8.288
  19    108.392    110.404    183.925      2.012      8.150
  20    107.960    109.938    187.975      1.978      8.011
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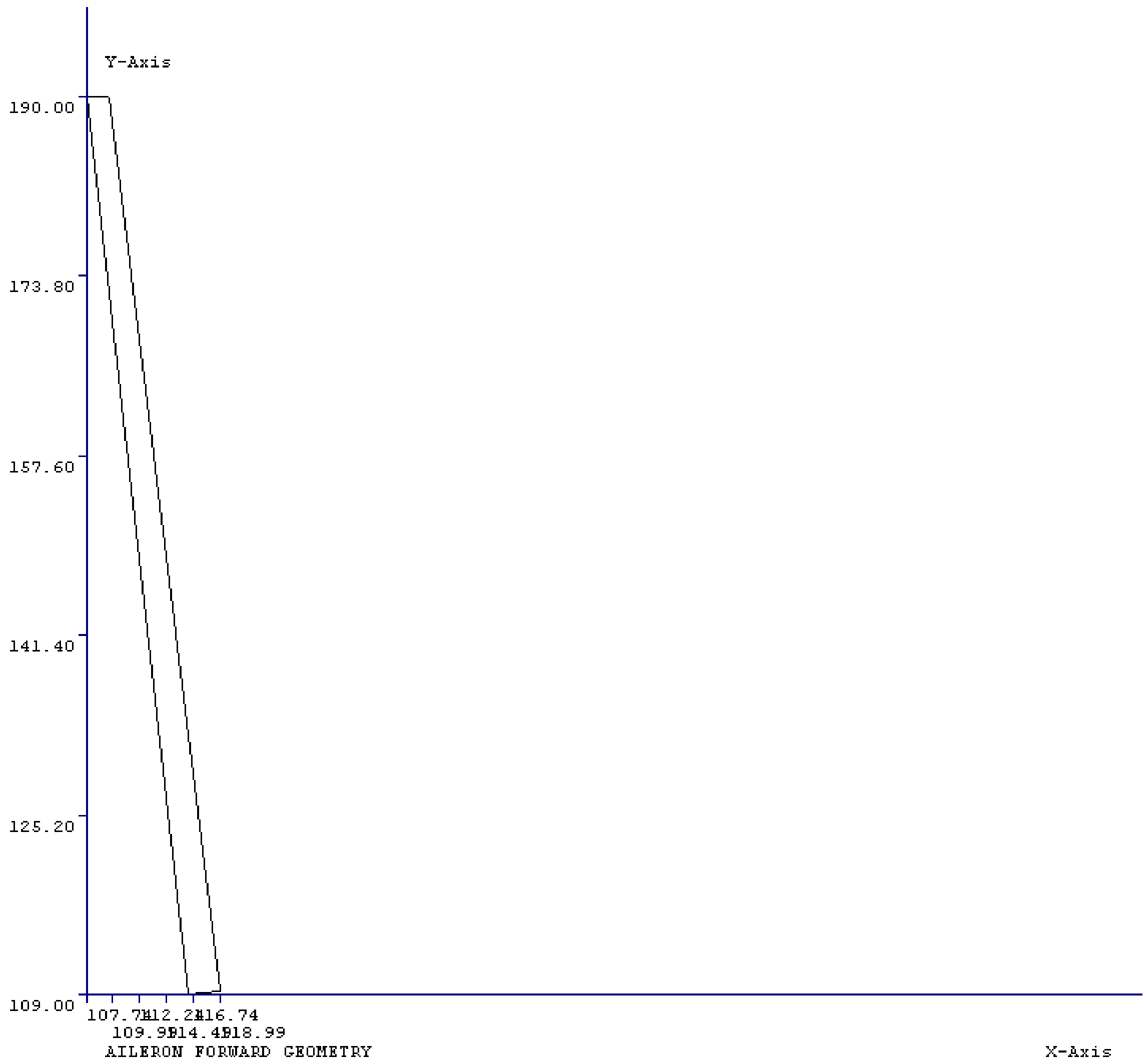
Y-Axis
190.00
173.80
157.60
141.40
125.20
109.00
107.74
109.99
112.24
114.49
116.74
118.99
AILERON FORWARD GEOMETRY
X-Axis

A.7 Aileron Aft of Hinge Line Geometry

```
AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

AILERON AFT OF HINGE GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE
POINT NO   FUS STA (XLE)   WING STA (YLE)
 1            118.993         109.279
 2            109.705         190.000

COORDINATES OF TRAILING EDGE
POINT NO   FUS STA (XTE)   WING STA (YTE)
 1            118.993         109.279
 2            129.522         110.401
 3            117.642         190.000

THE SURFACE IS DIVIDED INTO  20 INCREMENTS OF DY

AREA/SIDE         MAC    YLE(MAC)   XLE(MAC)   ASPECT RATIO
   752.056      9.385    147.652    114.578      8.664

ELEMENT DATA
ELEM       XLE        XTE         Y          C         AREA
   1    118.761    129.388    111.297     10.627     42.893
   2    118.296    128.786    115.333     10.489     42.336
   3    117.832    128.184    119.369     10.352     41.779
   4    117.368    127.581    123.405     10.214     41.222
   5    116.903    126.979    127.441     10.076     40.666
   6    116.439    126.376    131.477      9.938     40.109
   7    115.974    125.774    135.513      9.800     39.552
   8    115.510    125.172    139.549      9.662     38.995
   9    115.046    124.569    143.585      9.524     38.438
  10    114.581    123.967    147.621      9.386     37.881
  11    114.117    123.365    151.658      9.248     37.324
  12    113.652    122.762    155.694      9.110     36.767
  13    113.188    122.160    159.730      8.972     36.211
  14    112.724    121.557    163.766      8.834     35.654
  15    112.259    120.955    167.802      8.696     35.097
  16    111.795    120.353    171.838      8.558     34.540
  17    111.330    119.750    175.874      8.420     33.983
  18    110.866    119.148    179.910      8.282     33.426
  19    110.402    118.546    183.946      8.144     32.869
  20    109.937    117.943    187.982      8.006     32.313
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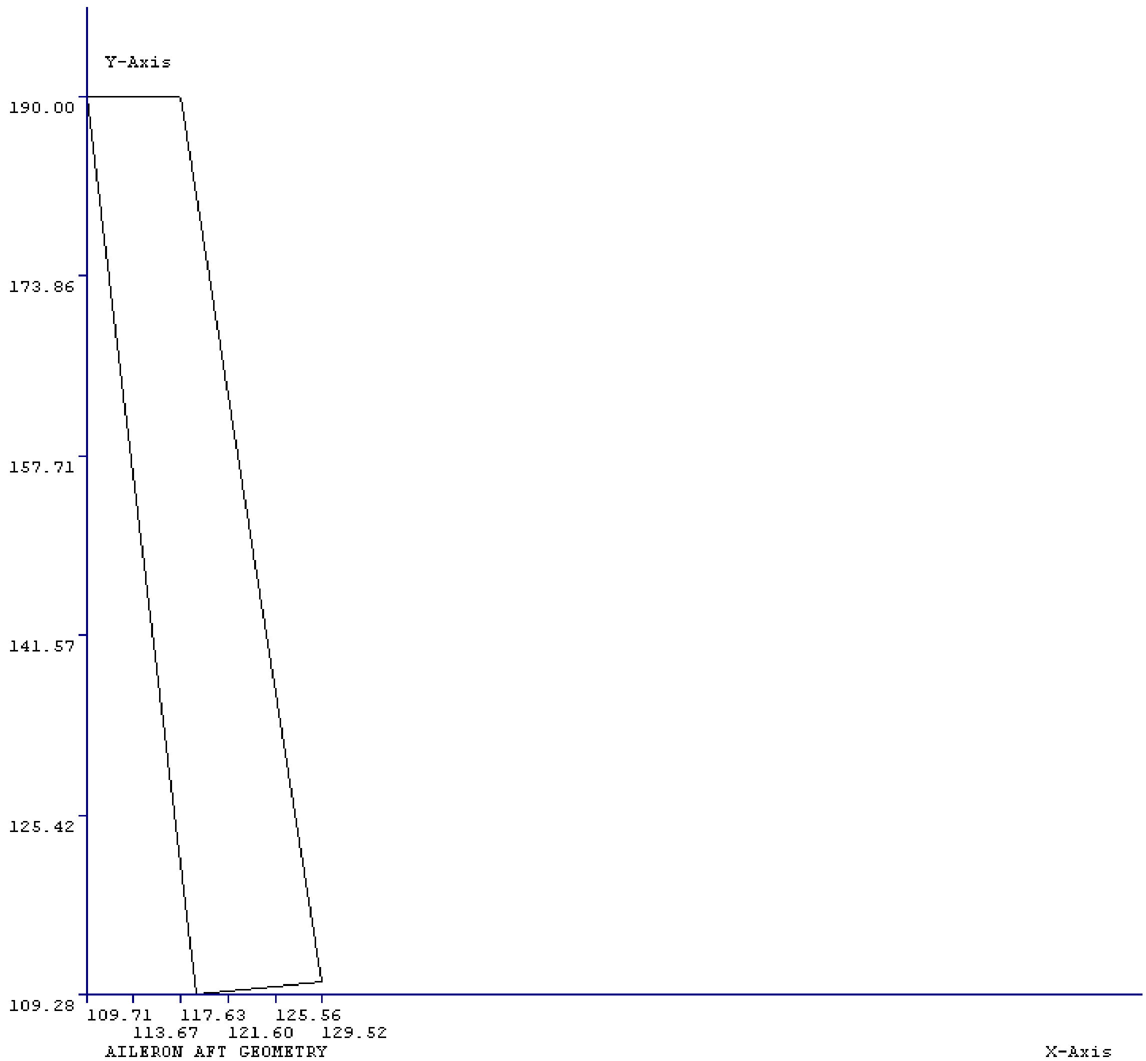

Y-Axis
190.00
173.86
157.71
141.57
125.42
109.28
109.71
113.67
117.63
121.60
125.56
129.52
AILERON AFT GEOMETRY
X-Axis

A.8 Flap Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

FLAP GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	122.567	23.000
2	114.065	108.753
3	129.522	110.401

COORDINATES OF TRAILING EDGE

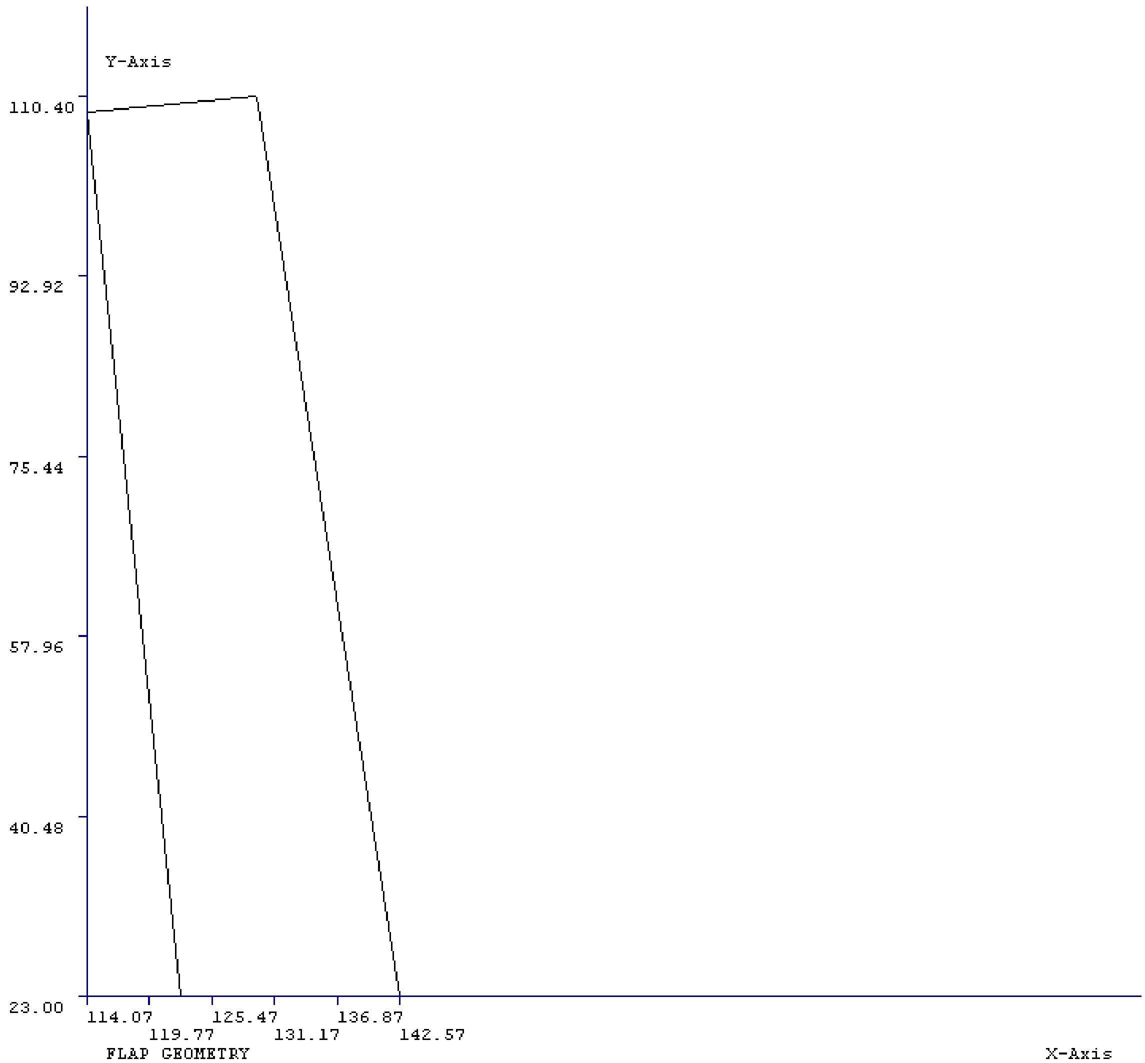
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	142.567	23.000
2	129.522	110.401

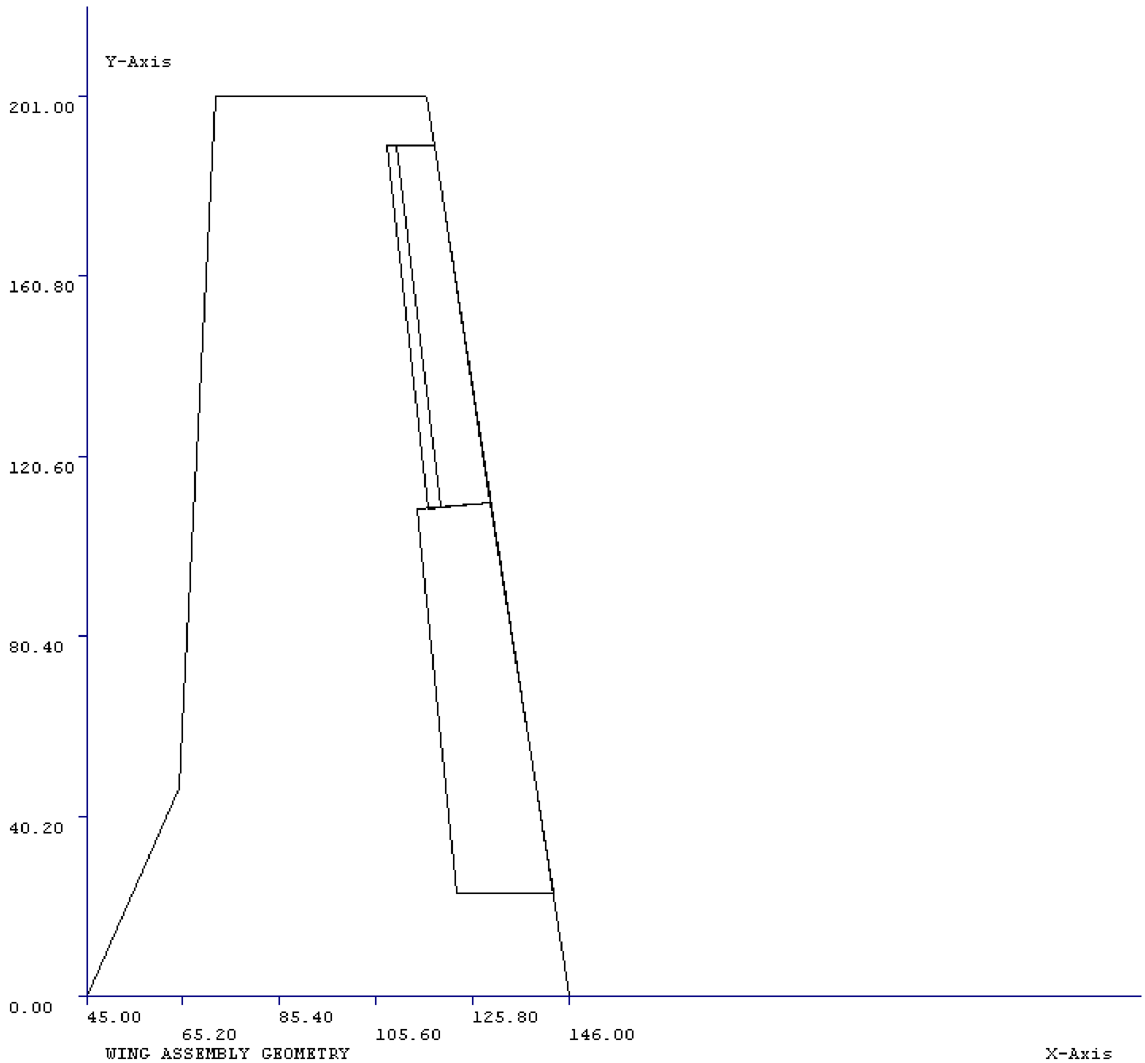
THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
1556.629	17.900	64.914	118.411	4.907

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	122.350	142.241	25.185	19.891	86.922
2	121.917	141.589	29.555	19.672	85.965
3	121.484	140.936	33.925	19.453	85.009
4	121.051	140.284	38.295	19.234	84.051
5	120.617	139.632	42.665	19.015	83.095
6	120.184	138.980	47.035	18.796	82.138
7	119.751	138.327	51.405	18.577	81.181
8	119.318	137.675	55.775	18.358	80.224
9	118.884	137.023	60.145	18.139	79.267
10	118.451	136.371	64.515	17.920	78.310
11	118.018	135.718	68.886	17.701	77.353
12	117.584	135.066	73.256	17.482	76.396
13	117.151	134.414	77.626	17.263	75.439
14	116.718	133.762	81.996	17.044	74.482
15	116.285	133.109	86.366	16.825	73.525
16	115.851	132.457	90.736	16.606	72.568
17	115.418	131.805	95.106	16.387	71.611
18	114.985	131.153	99.476	16.168	70.654
19	114.552	130.500	103.846	15.949	69.697
20	114.118	129.848	108.216	15.730	68.740


Y-Axis
110.40
92.92
75.44
57.96
40.48
23.00
114.07
119.77
125.47
131.17
136.87
142.57
FLAP GEOMETRY
X-Axis


Y-Axis
201.00
160.80
120.60
80.40
40.20
0.00
45.00
65.20
85.40
105.60
125.80
146.00
WING ASSEMBLY GEOMETRY
X-Axis

A.9 Vertical Tail Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

VERTICAL TAIL GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	292.917	111.500
2	240.912	117.000
3	277.000	168.000

COORDINATES OF TRAILING EDGE

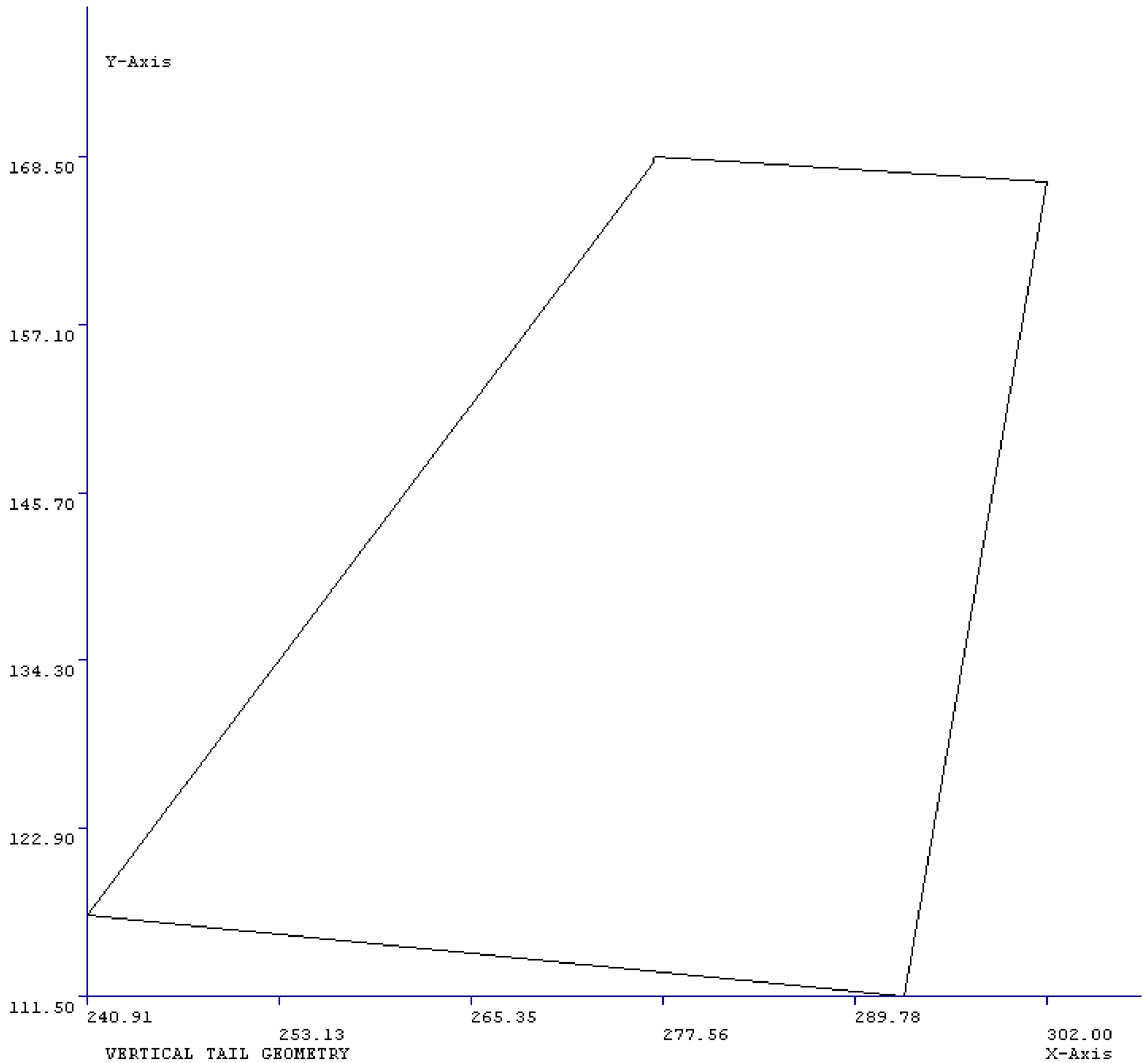
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	292.917	111.500
2	302.000	166.794
3	277.000	168.500

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
2137.330	40.159	137.819	257.081	1.494

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	279.561	293.149	112.913	13.588	38.386
2	252.850	293.613	115.738	40.764	115.157
3	242.018	294.077	118.563	52.060	147.068
4	244.017	294.541	121.388	50.525	142.732
5	246.016	295.005	124.213	48.990	138.396
6	248.015	295.469	127.038	47.455	134.060
7	250.014	295.933	129.863	45.920	129.723
8	252.013	296.397	132.688	44.385	125.387
9	254.012	296.862	135.513	42.850	121.051
10	256.011	297.326	138.338	41.315	116.715
11	258.010	297.790	141.163	39.780	112.379
12	260.009	298.254	143.988	38.245	108.042
13	262.008	298.718	146.813	36.710	103.706
14	264.007	299.182	149.638	35.175	99.370
15	266.006	299.646	152.463	33.640	95.034
16	268.005	300.110	155.288	32.105	90.698
17	270.004	300.574	158.113	30.570	86.361
18	272.003	301.038	160.938	29.036	82.025
19	274.002	301.502	163.763	27.501	77.689
20	276.001	301.966	166.588	25.966	73.353


Y-Axis
168.50
157.10
145.70
134.30
122.90
111.50
240.91
253.13
265.35
277.56
289.78
302.00
VERTICAL TAIL GEOMETRY
X-Axis

A.10 Vertical Stabilizer Geometry

```
AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

VERTICAL STABILIZER GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE
POINT NO   FUS STA (XLE)   WING STA (YLE)
 1            275.346         113.358
 2            240.912         117.000
 3            277.000         168.500

COORDINATES OF TRAILING EDGE
POINT NO   FUS STA (XTE)   WING STA (YTE)
 1            275.346         113.358
 2            292.443         163.539
 3            280.098         164.773
 4            282.000         168.000
 5            277.000         168.500

THE SURFACE IS DIVIDED INTO  20 INCREMENTS OF DY

AREA/SIDE         MAC    YLE(MAC)   XLE(MAC)   ASPECT RATIO
  1371.616     27.386    137.437    255.857      2.217

ELEMENT DATA
ELEM        XLE        XTE         Y          C         AREA
   1     262.313    275.816    114.737     13.503     37.229
   2     241.258    276.755    117.494     35.497     97.868
   3     243.190    277.694    120.251     34.504     95.132
   4     245.122    278.634    123.008     33.512     92.395
   5     247.054    279.573    125.765     32.519     89.658
   6     248.986    280.513    128.522     31.526     86.921
   7     250.918    281.452    131.279     30.534     84.184
   8     252.850    282.391    134.036     29.541     81.448
   9     254.782    283.331    136.793     28.549     78.711
  10     256.714    284.270    139.550     27.556     75.974
  11     258.646    285.209    142.308     26.563     73.237
  12     260.578    286.149    145.065     25.571     70.500
  13     262.510    287.088    147.822     24.578     67.764
  14     264.442    288.027    150.579     23.585     65.027
  15     266.374    288.967    153.336     22.593     62.290
  16     268.306    289.906    156.093     21.600     59.554
  17     270.238    290.846    158.850     20.608     56.817
  18     272.170    291.785    161.607     19.615     54.080
  19     274.102    284.187    164.364     10.085     27.805
  20     276.034    281.482    167.121      5.448     15.021
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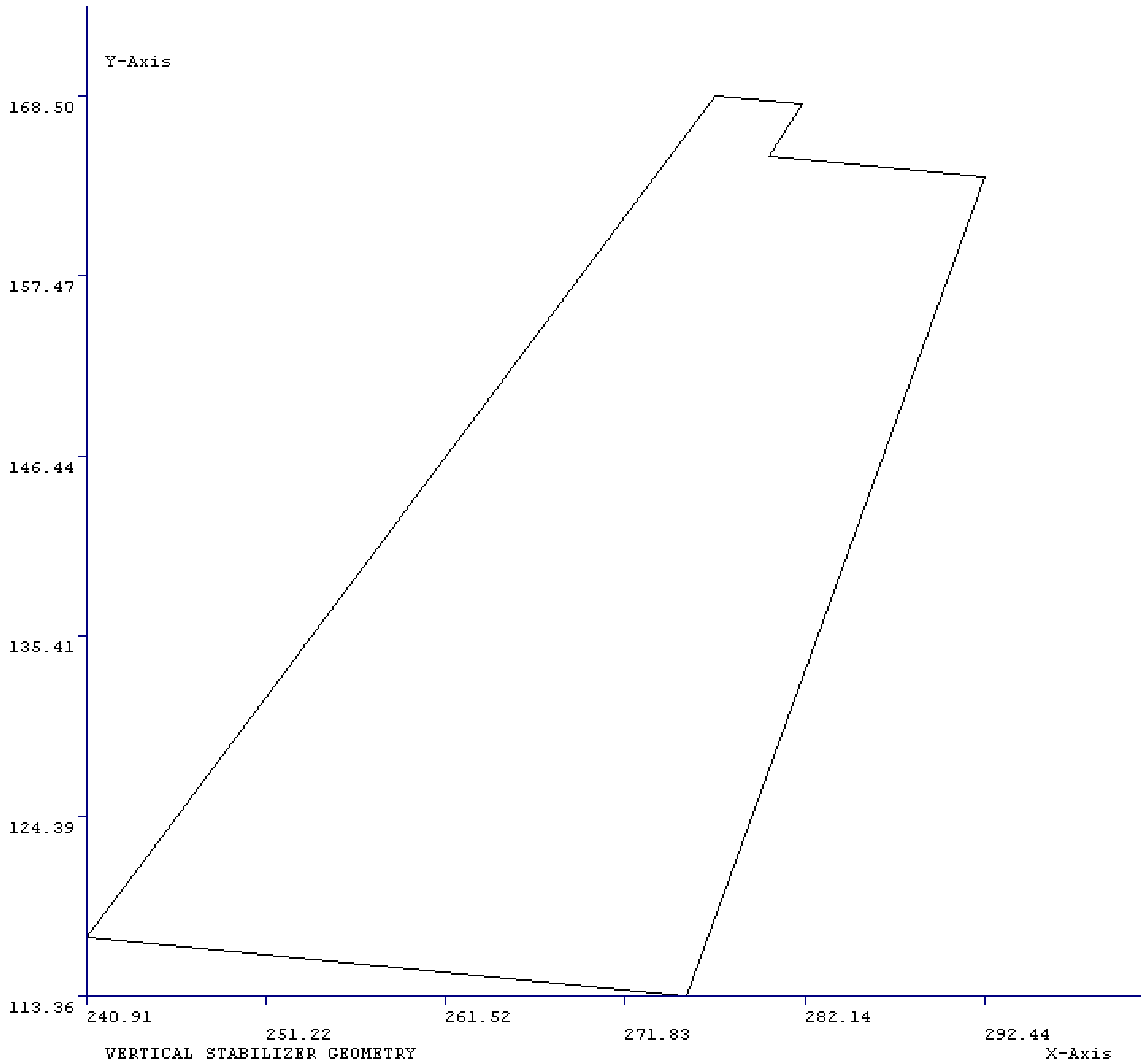


Y-Axis
168.50
157.47
146.44
135.41
124.39
113.36
240.91
251.22
261.52
271.83
282.14
292.44
VERTICAL STABILIZER GEOMETRY
X-Axis

A.11 Rudder Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

RUDDER GEOMETRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	292.917	111.500
2	275.798	114.685
3	292.443	163.539
4	280.098	164.773
5	282.000	168.000

COORDINATES OF TRAILING EDGE

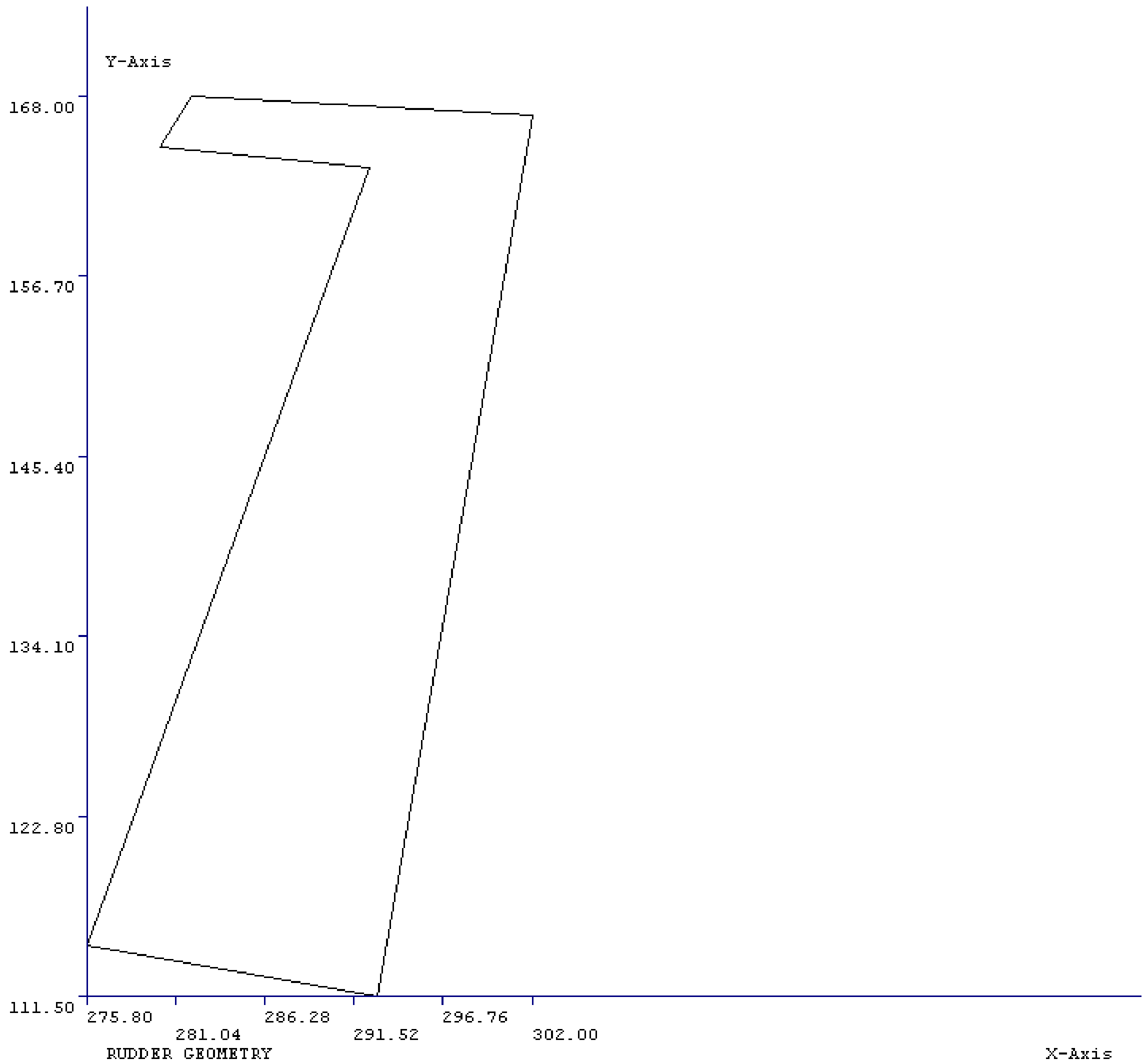
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	292.917	111.500
2	302.000	166.794
3	282.000	168.000

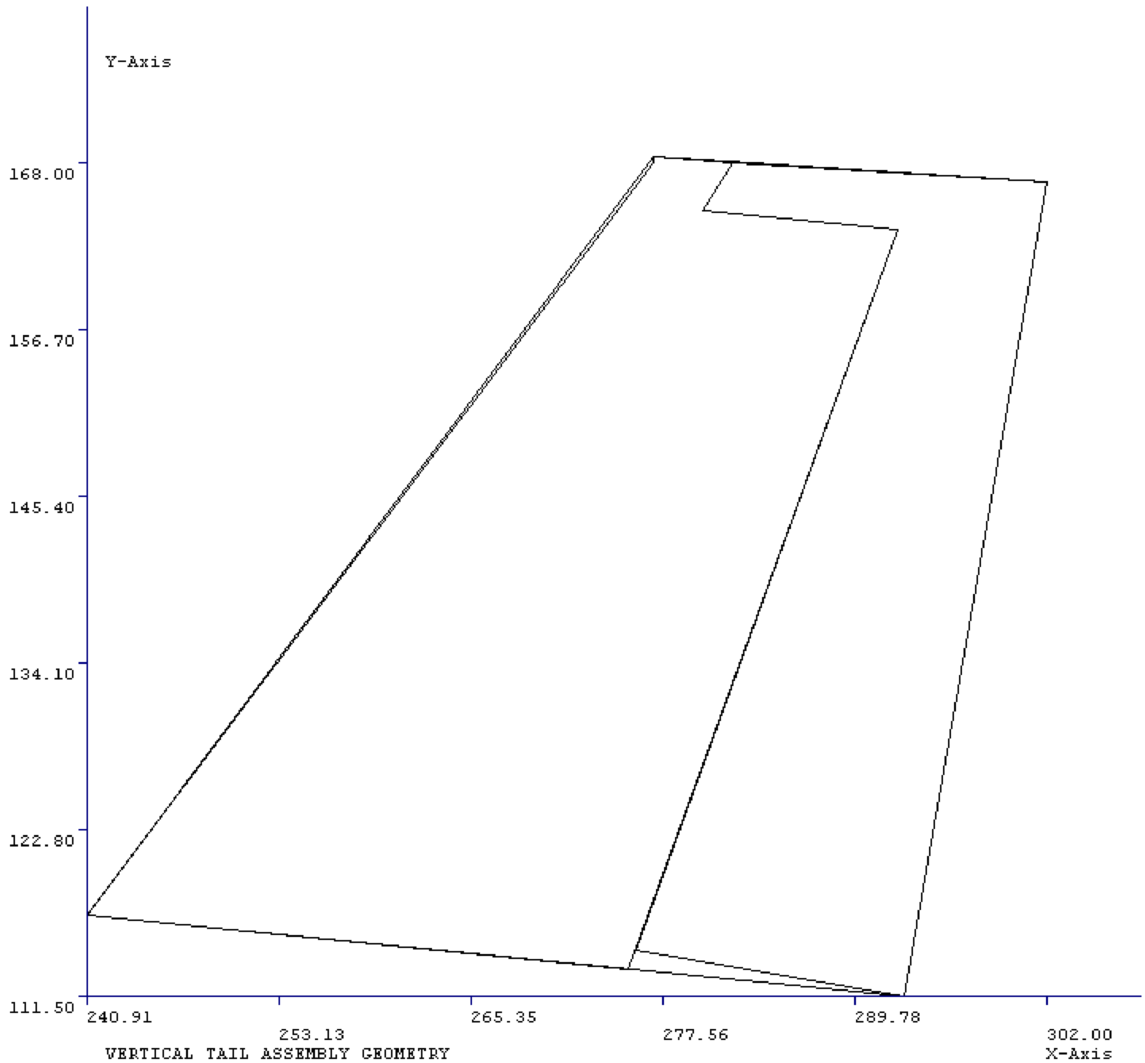
THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
759.612	14.157	138.714	283.230	4.202

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	285.325	293.149	112.913	7.824	22.103
2	276.157	293.613	115.738	17.457	49.315
3	277.119	294.077	118.563	16.958	47.906
4	278.082	294.541	121.388	16.460	46.498
5	279.044	295.005	124.213	15.961	45.090
6	280.007	295.469	127.038	15.463	43.682
7	280.969	295.933	129.863	14.964	42.274
8	281.932	296.397	132.688	14.466	40.866
9	282.894	296.862	135.513	13.967	39.458
10	283.857	297.326	138.338	13.469	38.050
11	284.819	297.790	141.163	12.971	36.642
12	285.782	298.254	143.988	12.472	35.233
13	286.744	298.718	146.813	11.974	33.825
14	287.707	299.182	149.638	11.475	32.417
15	288.669	299.646	152.463	10.977	31.009
16	289.632	300.110	155.288	10.478	29.601
17	290.594	300.574	158.113	9.980	28.193
18	291.557	301.038	160.938	9.481	26.785
19	290.207	301.502	163.763	11.295	31.908
20	281.168	301.966	166.588	20.799	58.756


Y-Axis
168.00
156.70
145.40
134.10
122.80
111.50
275.80
281.04
286.28
291.52
296.76
302.00
RUDDER GEOMETRY
X-Axis


Y-Axis
168.00
156.70
145.40
134.10
122.80
111.50
240.91
253.13
265.35
277.56
289.78
302.00
VERTICAL TAIL ASSEMBLY GEOMETRY
X-Axis

A.12 Horizontal Tail Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

HORIZONTAL TAIL GEOMETRY
SYMMETRICAL ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	247.600	0.000
2	255.207	63.685
3	256.117	70.100
4	261.000	72.100
5	268.100	73.100

COORDINATES OF TRAILING EDGE

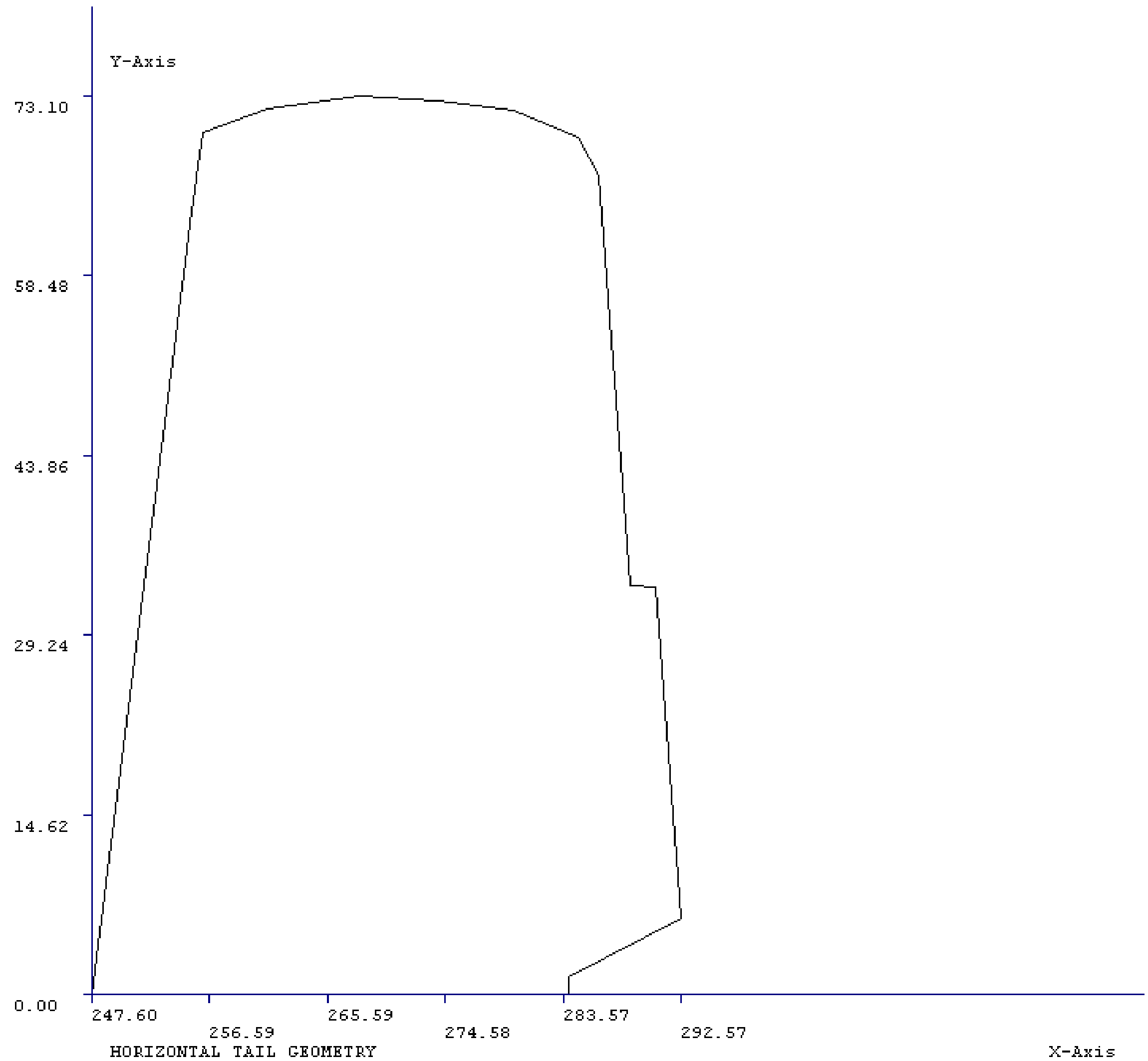
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	284.000	0.000
2	284.000	1.445
3	292.567	6.200
4	290.662	33.200
5	288.657	33.201
6	286.300	66.600
7	284.700	69.700
8	279.800	71.900
9	274.000	72.700
10	268.100	73.100

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
2666.041	37.249	33.799	251.728	4.009

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	247.818	284.689	1.828	36.871	134.763
2	248.255	291.274	5.483	43.019	157.236
3	248.691	292.360	9.138	43.668	159.608
4	249.128	292.102	12.793	42.974	157.070
5	249.565	291.844	16.448	42.279	154.531
6	250.001	291.586	20.103	41.585	151.993
7	250.438	291.328	23.758	40.890	149.454
8	250.874	291.070	27.413	40.196	146.916
9	251.311	290.813	31.068	39.502	144.378
10	251.748	288.550	34.723	36.802	134.512
11	252.184	288.292	38.378	36.108	131.973
12	252.621	288.034	42.033	35.413	129.435
13	253.057	287.776	45.688	34.719	126.896
14	253.494	287.518	49.343	34.024	124.358
15	253.930	287.260	52.998	33.330	121.819
16	254.367	287.002	56.653	32.635	119.281
17	254.804	286.744	60.308	31.941	116.743
18	255.246	286.486	63.963	31.240	114.181
19	255.765	285.775	67.618	30.010	109.687
20	258.980	281.198	71.273	22.218	81.207


Y-Axis
73.10
58.48
43.86
29.24
14.62
0.00
247.60
256.59
265.59
274.58
283.57
292.57
HORIZONTAL TAIL GEOMETRY
X-Axis

A.13 Horizontal Stabilizer Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

HORIZONTAL STABILIZER GEOMETRY
SYMMETRICAL ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	247.600	0.000
2	255.207	63.685
3	256.117	70.100
4	261.000	72.100

COORDINATES OF TRAILING EDGE

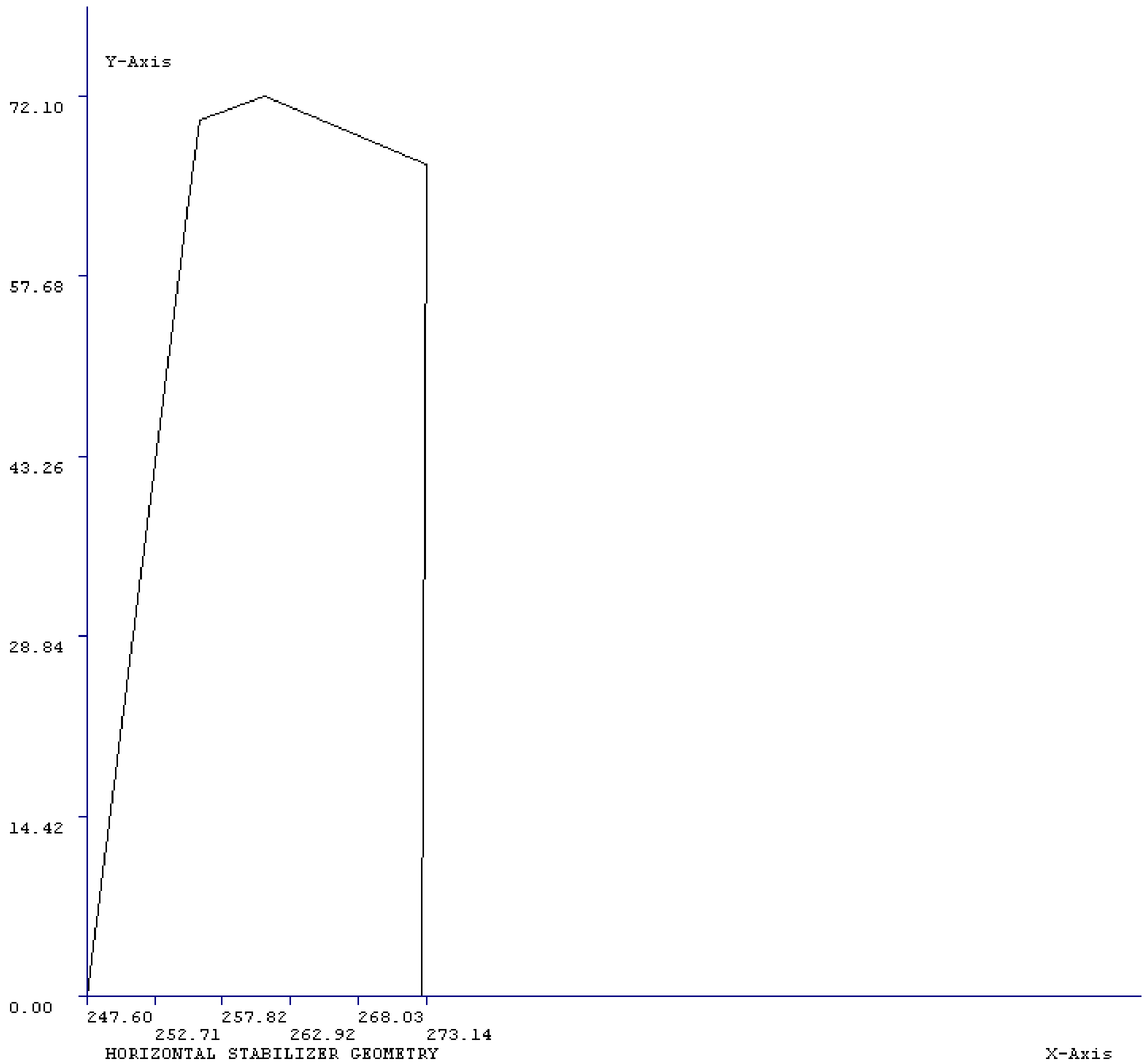
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	272.750	0.000
2	273.138	66.600
3	261.000	72.100

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
1485.342	21.220	32.916	251.547	7.000

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	247.815	272.761	1.803	24.945	89.927
2	248.246	272.782	5.408	24.536	88.451
3	248.677	272.803	9.013	24.126	86.974
4	249.107	272.824	12.618	23.716	85.498
5	249.538	272.845	16.223	23.307	84.021
6	249.968	272.866	19.828	22.897	82.544
7	250.399	272.887	23.433	22.488	81.068
8	250.830	272.908	27.038	22.078	79.591
9	251.260	272.929	30.643	21.668	78.114
10	251.691	272.950	34.248	21.259	76.638
11	252.121	272.971	37.853	20.849	75.161
12	252.552	272.992	41.458	20.440	73.684
13	252.983	273.013	45.063	20.030	72.208
14	253.413	273.034	48.668	19.620	70.731
15	253.844	273.055	52.273	19.211	69.255
16	254.274	273.076	55.878	18.801	67.778
17	254.705	273.097	59.483	18.392	66.301
18	255.136	273.118	63.088	17.982	64.825
19	255.634	272.934	66.693	17.300	62.368
20	256.599	264.978	70.298	8.379	30.205


Y-Axis
72.10
57.68
43.26
28.84
14.42
0.00
247.60
252.71
257.82
262.92
268.03
273.14
HORIZONTAL STABILIZER GEOMETRY
X-Axis

A.14 Elevator Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

ELEVATOR GEOMETRY
SYMMETRICAL ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	272.750	0.000
2	273.138	66.600
3	261.000	72.100
4	268.100	73.100

COORDINATES OF TRAILING EDGE

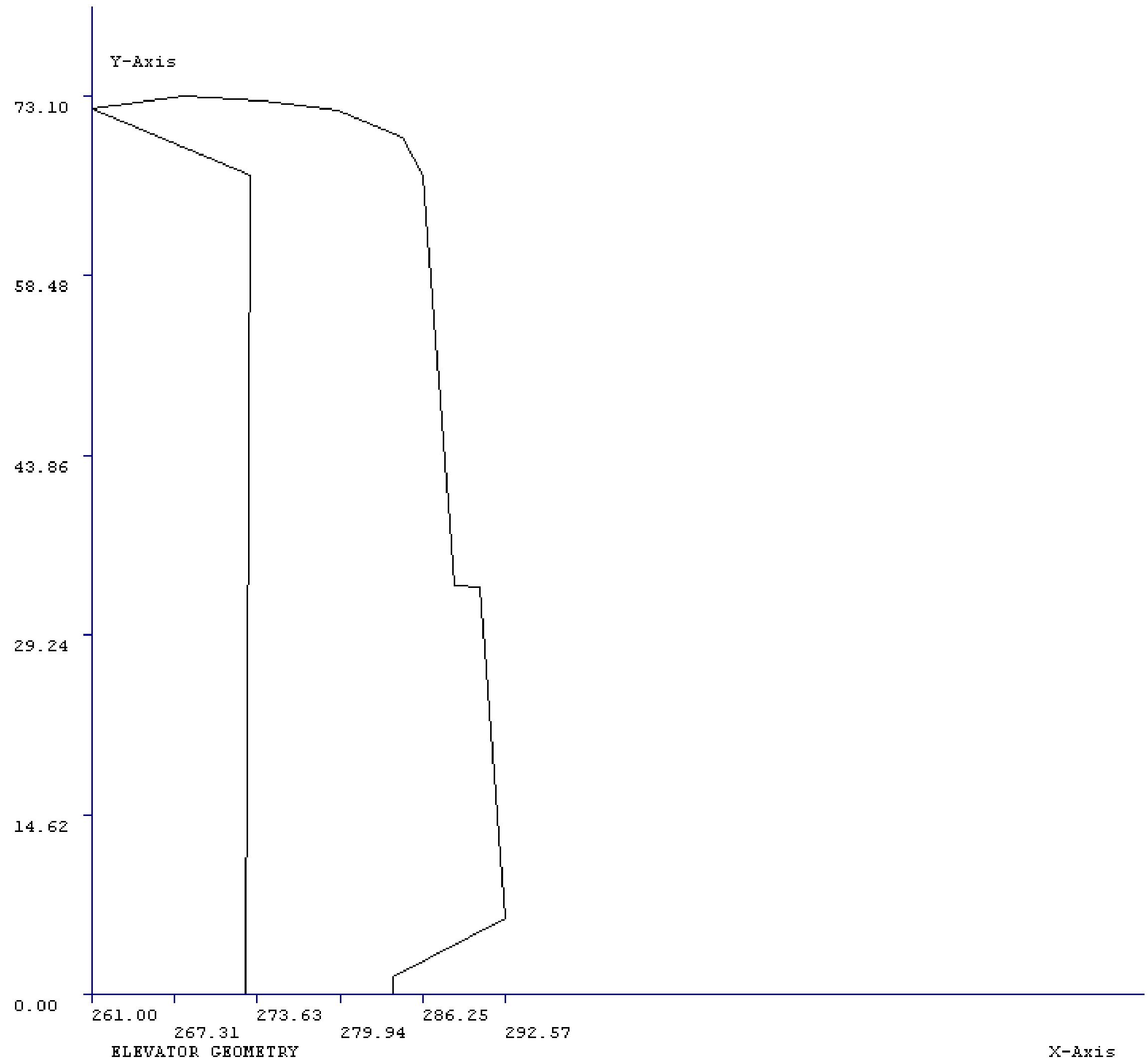
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	284.000	0.000
2	284.000	1.445
3	292.567	6.200
4	290.662	33.200
5	288.657	33.201
6	286.300	66.600
7	284.700	69.700
8	279.800	71.900
9	274.000	72.700
10	268.100	73.100

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
1187.989	16.583	35.116	272.267	8.996

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	272.761	284.689	1.828	11.929	43.599
2	272.782	291.274	5.483	18.492	67.590
3	272.803	292.360	9.138	19.557	71.479
4	272.825	292.102	12.793	19.277	70.459
5	272.846	291.844	16.448	18.998	69.438
6	272.867	291.586	20.103	18.719	68.418
7	272.888	291.328	23.758	18.440	67.397
8	272.910	291.070	27.413	18.161	66.377
9	272.931	290.813	31.068	17.882	65.357
10	272.952	288.550	34.723	15.597	57.008
11	272.974	288.292	38.378	15.318	55.988
12	272.995	288.034	42.033	15.039	54.967
13	273.016	287.776	45.688	14.760	53.946
14	273.038	287.518	49.343	14.480	52.926
15	273.059	287.260	52.998	14.201	51.905
16	273.080	287.002	56.653	13.922	50.885
17	273.101	286.744	60.308	13.643	49.864
18	273.123	286.486	63.963	13.364	48.844
19	270.893	285.775	67.618	14.882	54.395
20	262.826	281.198	71.273	18.371	67.147


Y-Axis
73.10
58.48
43.86
29.24
14.62
0.00
261.00
267.31
273.63
279.94
286.25
292.57
ELEVATOR GEOMETRY
X-Axis

A.15 Elevator Forward of Hinge Line Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

ELEVATOR FORWARD OF HINGE GEOMETRY
SYMMETRICAL ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	272.750	0.000
2	273.138	66.600
3	261.000	72.100
4	268.100	73.100

COORDINATES OF TRAILING EDGE

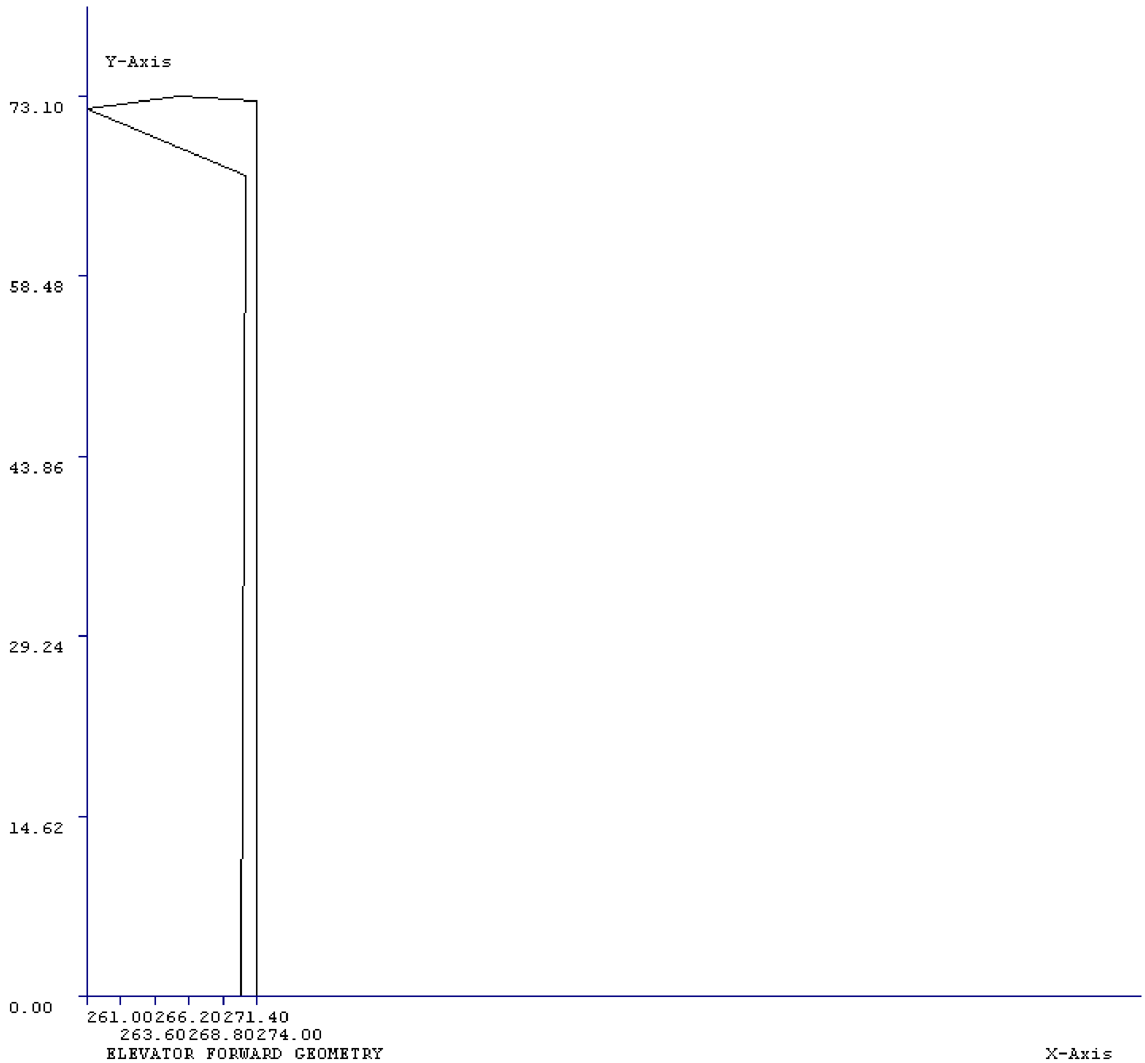
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	274.000	0.000
2	274.000	72.700
3	268.100	73.100

THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
121.828	4.647	47.866	269.353	87.724

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	272.761	274.000	1.828	1.239	4.530
2	272.782	274.000	5.483	1.218	4.452
3	272.803	274.000	9.138	1.197	4.374
4	272.825	274.000	12.793	1.176	4.296
5	272.846	274.000	16.448	1.154	4.219
6	272.867	274.000	20.103	1.133	4.141
7	272.888	274.000	23.758	1.112	4.063
8	272.910	274.000	27.413	1.090	3.985
9	272.931	274.000	31.068	1.069	3.907
10	272.952	274.000	34.723	1.048	3.829
11	272.974	274.000	38.378	1.026	3.751
12	272.995	274.000	42.033	1.005	3.674
13	273.016	274.000	45.688	0.984	3.596
14	273.038	274.000	49.343	0.963	3.518
15	273.059	274.000	52.998	0.941	3.440
16	273.080	274.000	56.653	0.920	3.363
17	273.101	274.000	60.308	0.899	3.285
18	273.123	274.000	63.963	0.877	3.207
19	270.893	274.000	67.618	3.108	11.358
20	262.826	274.000	71.273	11.174	40.840



Y-Axis
73.10
58.48
43.86
29.24
14.62
0.00
261.00266.20271.40
263.60268.80274.00
ELEVATOR FORWARD GEOMETRY
X-Axis

A.16 Elevator Aft of Hinge Line Geometry

```
AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

ELEVATOR AFT OF HINGE GEOMETRY
SYMMETRICAL ABOUT CL

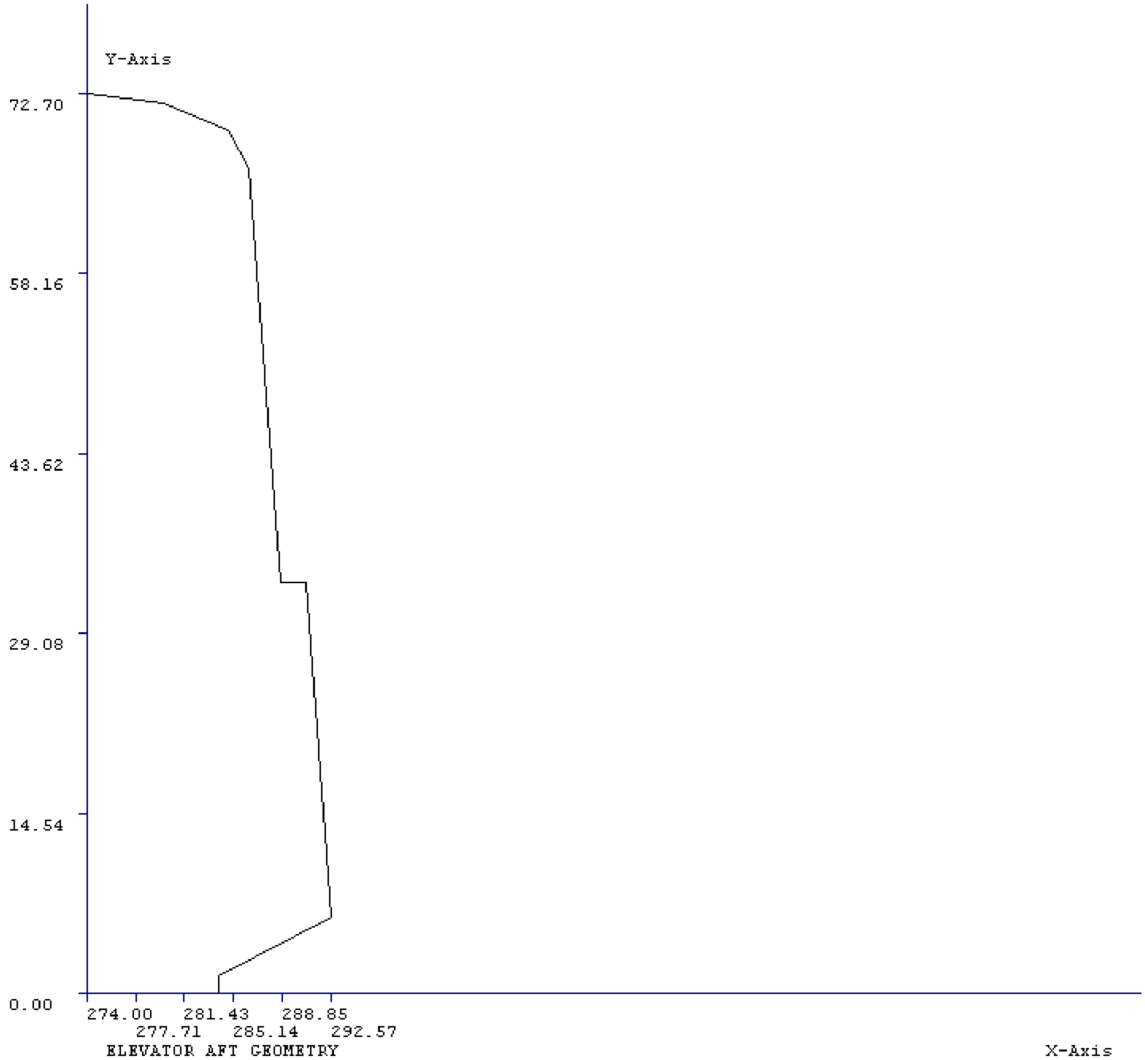
COORDINATES OF LEADING EDGE
POINT NO   FUS STA (XLE)   WING STA (YLE)
 1             274.000           0.000
 2             274.000          72.700

COORDINATES OF TRAILING EDGE
POINT NO   FUS STA (XTE)   WING STA (YTE)
 1             284.000           0.000
 2             284.000           1.445
 3             292.567           6.200
 4             290.662          33.200
 5             288.657          33.201
 6             286.300          66.600
 7             284.700          69.700
 8             279.800          71.900
 9             274.000          72.700

THE SURFACE IS DIVIDED INTO  20 INCREMENTS OF DY

AREA/SIDE         MAC    YLE(MAC)   XLE(MAC)   ASPECT RATIO
  1064.739     15.160     33.623    274.000      9.928

ELEMENT DATA
ELEM        XLE        XTE         Y          C         AREA
   1    274.000    284.671      1.818     10.671      38.790
   2    274.000    291.220      5.453     17.220      62.596
   3    274.000    292.363      9.088     18.363      66.751
   4    274.000    292.107     12.723     18.107      65.818
   5    274.000    291.850     16.358     17.850      64.886
   6    274.000    291.594     19.993     17.594      63.954
   7    274.000    291.337     23.628     17.337      63.021
   8    274.000    291.081     27.263     17.081      62.089
   9    274.000    290.825     30.898     16.825      61.157
  10    274.000    288.563     34.533     14.563      52.937
  11    274.000    288.307     38.168     14.307      52.004
  12    274.000    288.050     41.803     14.050      51.072
  13    274.000    287.794     45.438     13.794      50.139
  14    274.000    287.537     49.073     13.537      49.207
  15    274.000    287.280     52.708     13.280      48.274
  16    274.000    287.024     56.343     13.024      47.342
  17    274.000    286.767     59.978     12.767      46.409
  18    274.000    286.511     63.613     12.511      45.477
  19    274.000    285.966     67.248     11.966      43.496
  20    274.000    282.066     70.883      8.066      29.321
```


Y-Axis
72.70
58.16
43.62
29.08
14.54
0.00
274.00
277.71
281.43
285.14
288.85
292.57
ELEVATOR AFT GEOMETRY
X-Axis

A.17 Elevator Tab Geometry

AERODYNAMIC SURFACE GEOMETRIC PROPERTIES

ELEVATOR TAB GEOEMTRY
NOT SYM ABOUT CL

COORDINATES OF LEADING EDGE

POINT NO	FUS STA (XLE)	WING STA (YLE)
1	284.000	1.445
2	284.000	33.201

COORDINATES OF TRAILING EDGE

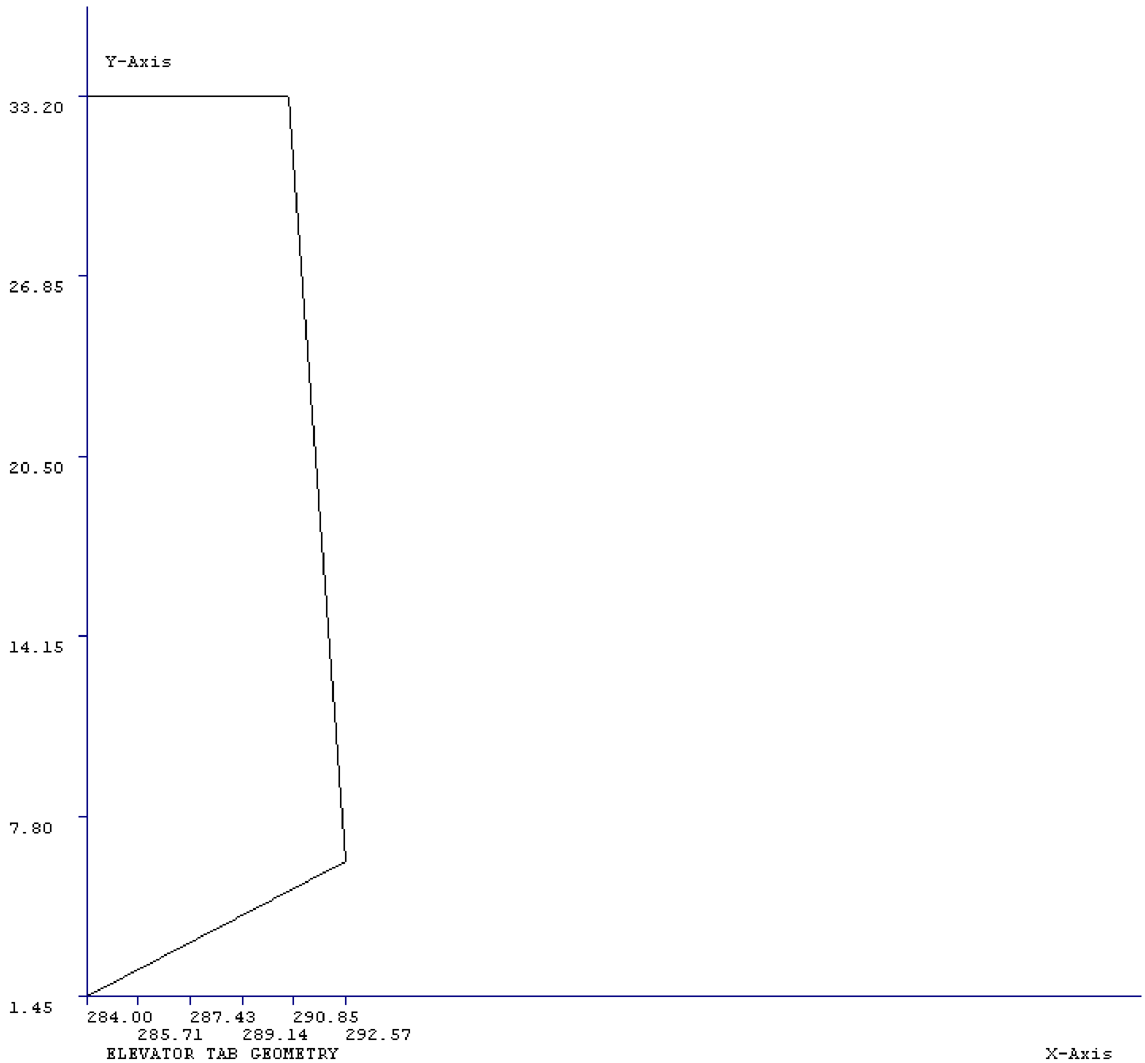
POINT NO	FUS STA (XTE)	WING STA (YTE)
1	284.000	1.445
2	292.567	6.200
3	290.662	33.201

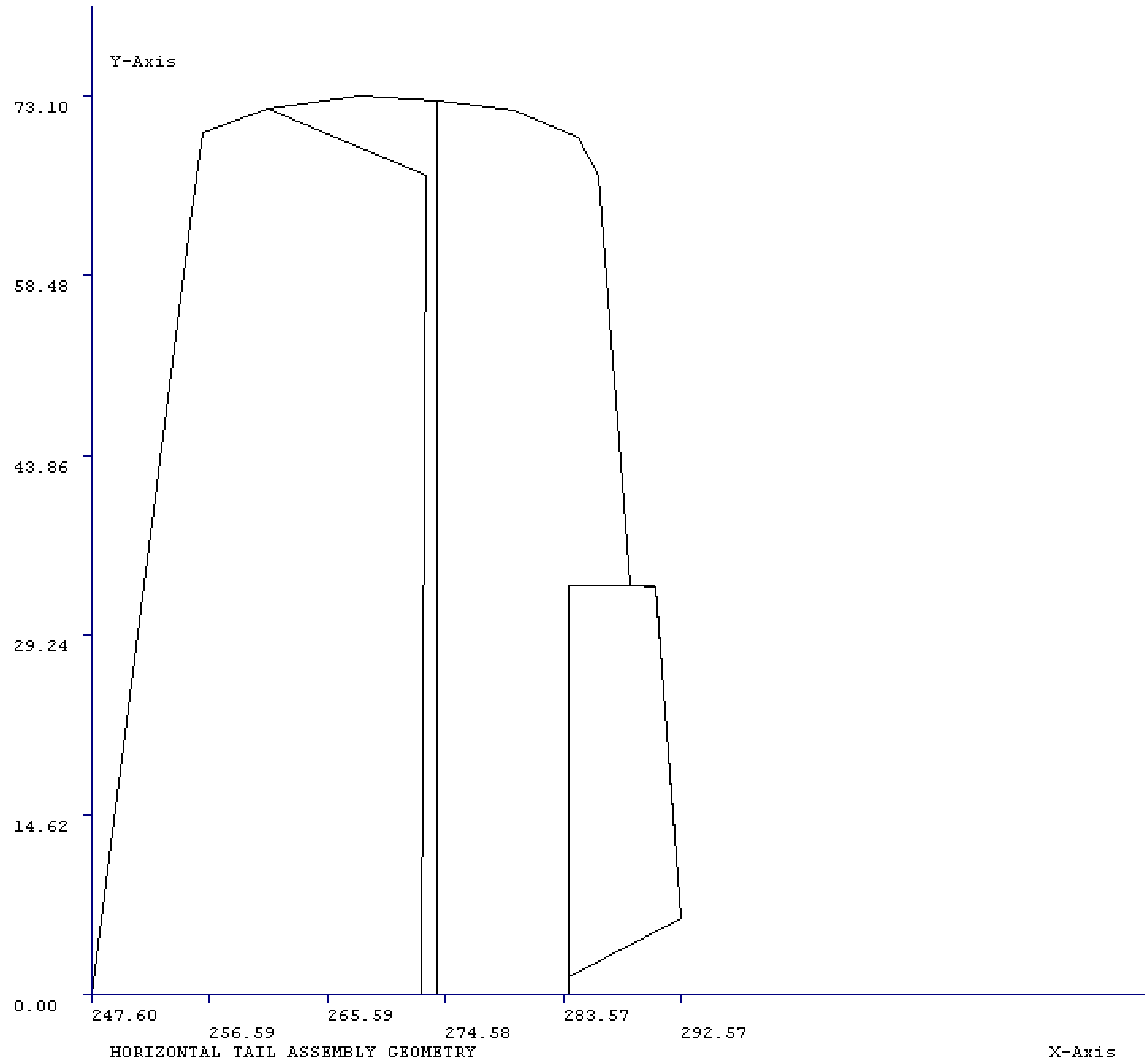
THE SURFACE IS DIVIDED INTO 20 INCREMENTS OF DY

AREA/SIDE	MAC	YLE(MAC)	XLE(MAC)	ASPECT RATIO
225.967	7.465	17.822	284.000	4.463

ELEMENT DATA

ELEM	XLE	XTE	Y	C	AREA
1	284.000	285.430	2.239	1.430	2.271
2	284.000	288.291	3.827	4.291	6.813
3	284.000	291.152	5.415	7.152	11.356
4	284.000	292.510	7.002	8.510	13.513
5	284.000	292.398	8.590	8.398	13.335
6	284.000	292.286	10.178	8.286	13.157
7	284.000	292.174	11.766	8.174	12.979
8	284.000	292.062	13.354	8.062	12.801
9	284.000	291.950	14.941	7.950	12.623
10	284.000	291.838	16.529	7.838	12.446
11	284.000	291.726	18.117	7.726	12.268
12	284.000	291.614	19.705	7.614	12.090
13	284.000	291.502	21.293	7.502	11.912
14	284.000	291.390	22.880	7.390	11.734
15	284.000	291.278	24.468	7.278	11.556
16	284.000	291.166	26.056	7.166	11.378
17	284.000	291.054	27.644	7.054	11.200
18	284.000	290.942	29.232	6.942	11.023
19	284.000	290.830	30.819	6.830	10.845
20	284.000	290.718	32.407	6.718	10.667


Y-Axis
33.20
26.85
20.50
14.15
7.80
1.45
284.00
285.71
287.43
289.14
290.85
292.57
ELEVATOR TAB GEOMETRY
X-Axis


Y-Axis
73.10
58.48
43.86
29.24
14.62
0.00
247.60
256.59
265.59
274.58
283.57
292.57
HORIZONTAL TAIL ASSEMBLY GEOMETRY
X-Axis

A.18 Structural Design Speed, Cat N, No Chosen Speeds

```
CALCULATIONS FOR STRUCURAL DESIGN SPEEDS AND LOAD FACTORS

INPUT

CATEGORY                                NORMAL
WING AREA (SQ FT)                          184.121
TAKE OFF WEIGHT (LBS)                     3400.000
MAX LEVEL SPEED (KNOTS)                    177.400
STALL SPEED CLEAN (KNOTS)                   62.200
STALL SPEED FLAPS EXTENDED (KNOTS)          58.600

CHOSEN SPEEDS AND LOAD FACTORS
VC =      1.000
VD =      1.000
VA =      1.000
VF =      1.000
+N =      1.000
-N =      1.000

OUTPUT

FAR 23 MINIMUM SPEEDS AND LOAD FACTORS
VCMIN =    141.808
VDMIN =    198.532
VAMIN =    121.250
VFMIN =    105.480
POS N =      3.800
NEG N =     -1.520

VERIFIED CHOSEN SPEEDS OR ADJUSTED SPEEDS & LOAD FACTORS
VC =    141.808
VD =    198.532
VA =    121.250
VF =    105.480
+N =      3.800
-N =     -1.520

MACH LIMITATION SPEEDS
SHOULDER =  12000.000
MC       =      0.269
MD       =      0.377
```

A.19 Structural Design Speed, Cat N, Vc Chosen

```
CALCULATIONS FOR STRUCURAL DESIGN SPEEDS AND LOAD FACTORS

INPUT

CATEGORY                                NORMAL
WING AREA (SQ FT)                          184.121
TAKE OFF WEIGHT (LBS)                     3400.000
MAX LEVEL SPEED (KNOTS)                    177.400
STALL SPEED CLEAN (KNOTS)                   62.200
STALL SPEED FLAPS EXTENDED (KNOTS)          58.600

CHOSEN SPEEDS AND LOAD FACTORS
VC =    170.000
VD =      1.000
VA =      1.000
VF =      1.000
+N =      1.000
-N =      1.000

OUTPUT

FAR 23 MINIMUM SPEEDS AND LOAD FACTORS
VCMIN =    141.808
VDMIN =    198.532
VAMIN =    121.250
VFMIN =    105.480
POS N =      3.800
NEG N =     -1.520

VERIFIED CHOSEN SPEEDS OR ADJUSTED SPEEDS & LOAD FACTORS
VC =    170.000
VD =    212.500
VA =    121.250
VF =    105.480
+N =      3.800
-N =     -1.520

MACH LIMITATION SPEEDS
SHOULDER =  12000.000
MC       =      0.323
MD       =      0.403
```

A.20 Mach Limit Lines

```
CALCULATIONS FOR STRUCURAL DESIGN SPEEDS AND LOAD FACTORS

INPUT

CATEGORY                                NORMAL
WING AREA (SQ FT)                          184.121
TAKE OFF WEIGHT (LBS)                     3400.000
MAX LEVEL SPEED (KNOTS)                    177.400
STALL SPEED CLEAN (KNOTS)                   62.200
STALL SPEED FLAPS EXTENDED (KNOTS)          58.600

CHOSEN SPEEDS AND LOAD FACTORS
VC =      1.000
VD =      1.000
VA =      1.000
VF =      1.000
+N =      1.000
-N =      1.000

OUTPUT

FAR 23 MINIMUM SPEEDS AND LOAD FACTORS
VCMIN =    141.808
VDMIN =    198.532
VAMIN =    121.250
VFMIN =    105.480
POS N =      3.800
NEG N =     -1.520

VERIFIED CHOSEN SPEEDS OR ADJUSTED SPEEDS & LOAD FACTORS
VC =    141.808
VD =    198.532
VA =    121.250
VF =    105.480
+N =      3.800
-N =     -1.520

MACH LIMITATION SPEEDS
SHOULDER =  12000.000
MC       =      0.269
MD       =      0.377

DATA TO PLOT MACH LIMITATION LINES ON FLIGHT LIMITS DIAGRAM

INPUT DATA

MC =      0.323
MD =      0.403
SHOULDER ALTITUDE      =  12000.000
MAX OPERATING ALTITUDE =  18000.000
INCREMENT OF ALTITUDE  =   1000.000

OUTPUT DATA

MNE =     0.3627
MFC =     0.4836

ALTITUDE         V(MC)        V(MNE)       V(MD)        V(FC)
 12000.000      170.163      191.077      212.308      254.770
 13000.000      166.829      187.334      208.149      249.779
 14000.000      163.536      183.636      204.040      244.848
 15000.000      160.283      179.984      199.982      239.979
 16000.000      157.071      176.377      195.975      235.169
 18000.000      150.768      169.299      188.110      225.731

NOTE: ALL SPEEDS ARE IN KNOTS EQUIVALENT AIR SPEED (KEAS).
```

A.21 Wing Aerodynamic Coefficients - Cruise

```
CRUISE CONFIGURATION

WING GEOMETRY CALCULATIONS
---- -------- ------------

 3 POINTS DEFINE THE LEADING EDGE
LEADING EDGE COORDINATES
 I       XLE       YLE
 1    45.000      0.000
 2    64.313     46.500
 3    72.000    201.000

 2 POINTS DEFINE THE TRAILING EDGE
TRAILING EDGE COORDINATES
 I       XTE       YTE
 1   146.000      0.000
 2   116.000    201.000

THE WING IS DIVIDED INTO  20 ELEMENTS OF DY
ELEMENT  YE     CE
 1      5.025     98.163
 2     15.075     92.489
 3     25.125     86.815
 4     35.175     81.141
 5     45.225     75.467
 6     55.275     73.000
 7     65.325     71.000
 8     75.375     69.000
 9     85.425     67.000
10     95.475     65.000
11    105.525     63.000
12    115.575     61.000
13    125.625     59.000
14    135.675     57.000
15    145.725     55.000
16    155.775     53.000
17    165.825     51.000
18    175.875     49.000
19    185.925     47.000
20    195.975     45.000

TOTAL AREA    = 26513.444
MAC           = 69.246
YBAR          = 87.854
XLE(MAC)      = 63.641
ASPECT RATIO  = 6.095
TOTAL SPAN    = 402.000
# OF ELEMENTS = 20.000
DY            = 10.050


ADDITIVE LIFT DISTRIBUTION
-------- ---- ------------

SELECTED WING STATIONS AND THEIR SLOPES OF LIFT CURVE

WING STA     SLOPE OF LIFT CURVE
     0.000          0.1075
   201.000          0.1075

ELEMENT     YE     CC(LA1)     C(LA1)  FOR CL=1.001
 1      5.025     91.056      0.928
 2     15.075     88.114      0.953
 3     25.125     85.065      0.980
 4     35.175     81.910      1.009
 5     45.225     78.644      1.042
 6     55.275     76.869      1.053
 7     65.325     75.208      1.059
```

8	75.375	73.424	1.064
9	85.425	71.507	1.067
10	95.475	69.448	1.068
11	105.525	67.236	1.067
12	115.575	64.852	1.063
13	125.625	62.277	1.056
14	135.675	59.479	1.043
15	145.725	56.419	1.026
16	155.775	53.034	1.001
17	165.825	49.229	0.965
18	175.875	44.827	0.915
19	185.925	39.454	0.839
20	195.975	31.830	0.707

BASIC LIFT DISTRIBUTION
----- ---- ------------

SELECTED WING STATIONS AND THEIR ANGLES FROM WL TO SECTION ZERO LIFT LINE

I	WING STA	ANGLE
1	0.000	5.000
2	46.500	4.577
3	109.279	4.028
4	201.000	1.900

AWO = 3.988

ELEM	REF ANGLE	Ao	CClb	Clb
1	4.954	0.966	5.098	0.052
2	4.863	0.875	4.348	0.047
3	4.771	0.783	3.655	0.042
4	4.680	0.692	3.017	0.037
5	4.589	0.600	2.436	0.032
6	4.500	0.512	2.010	0.028
7	4.412	0.424	1.619	0.023
8	4.325	0.336	1.247	0.018
9	4.237	0.248	0.895	0.013
10	4.149	0.161	0.561	0.009
11	4.061	0.073	0.246	0.004
12	3.882	-0.106	-0.348	-0.006
13	3.649	-0.339	-1.076	-0.018
14	3.416	-0.573	-1.754	-0.031
15	3.182	-0.806	-2.382	-0.043
16	2.949	-1.039	-2.959	-0.056
17	2.716	-1.272	-3.487	-0.068
18	2.483	-1.505	-3.964	-0.081
19	2.250	-1.738	-4.392	-0.093
20	2.017	-1.972	-4.769	-0.106

THERE IS NO DISCONTINUITY BETWEEN FLAP AND AILERON

ELEM	CC(lb)	C(lb) FAIRED
1	5.098	0.052
2	4.348	0.047
3	3.655	0.042
4	3.017	0.037
5	2.436	0.032
6	2.010	0.028
7	1.619	0.023
8	1.247	0.018
9	0.895	0.013
10	0.561	0.009
11	0.246	0.004
12	-0.348	-0.006
13	-1.076	-0.018
14	-1.754	-0.031
15	-2.382	-0.043
16	-2.959	-0.056
17	-3.487	-0.068

```
18     -3.964     -0.081
19     -4.392     -0.093
20     -4.769     -0.106

BCHECK = 0.000


STALL CALCULATIONS
----- ------------

SELECTED WING STAS , 2 CLMAXS & REYNOLDS NOS. AND CHORD

WING STA       CLMAX1          RN1          CLMAX2          RN2         CHORD
     0.000      1.450       3,000,000      1.660       9,000,000    101.000
    46.500      1.460       3,000,000      1.680       9,000,000     74.746
   109.279      1.480       3,000,000      1.700       9,000,000     62.253
   201.000      1.500       3,000,000      1.740       9,000,000     44.000

FOR 70 MPH
I= 1     R3N(I)=5,514,282     C3LMAX(I)=1.563
I= 2     R3N(I)=4,080,896     C3LMAX(I)=1.519
I= 3     R3N(I)=3,398,818     C3LMAX(I)=1.503
I= 4     R3N(I)=2,402,261     C3LMAX(I)=1.456

CL MAX (STALL) FOR EACH ELEMENT
 J       YE(J)    ELEMENT CL MAX
 1      5.025      1.558
 2     15.075      1.548
 3     25.125      1.539
 4     35.175      1.529
 5     45.225      1.520
 6     55.275      1.516
 7     65.325      1.514
 8     75.375      1.512
 9     85.425      1.509
10     95.475      1.507
11    105.525      1.504
12    115.575      1.500
13    125.625      1.495
14    135.675      1.490
15    145.725      1.484
16    155.775      1.479
17    165.825      1.474
18    175.875      1.469
19    185.925      1.464
20    195.975      1.458

WING STALL CL DISTRIBUTION
WING STALL CL = 1.410

 J      YE(J)       CL(J)
 1      5.025      1.360
 2     15.075      1.390
 3     25.125      1.424
 4     35.175      1.461
 5     45.225      1.502
 6     55.275      1.512
 7     65.325      1.516
 8     75.375      1.518
 9     85.425      1.518
10     95.475      1.515
11    105.525      1.509
12    115.575      1.493
13    125.625      1.470
14    135.675      1.441
15    145.725      1.403
16    155.775      1.355
17    165.825      1.293
18    175.875      1.209
19    185.925      1.090
20    195.975      0.891
```

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WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 0.000
 J        CL        CDI        CPD         CD         CM
 1      0.052      0.000      0.010      0.010     -0.030
 2      0.047      0.000      0.010      0.010     -0.030
 3      0.042      0.000      0.010      0.010     -0.030
 4      0.037      0.000      0.010      0.010     -0.030
 5      0.032      0.000      0.010      0.010     -0.030
 6      0.028      0.000      0.010      0.010     -0.030
 7      0.023      0.000      0.010      0.010     -0.030
 8      0.018      0.000      0.010      0.010     -0.030
 9      0.013      0.000      0.010      0.010     -0.030
10      0.009      0.000      0.010      0.010     -0.030
11      0.004      0.000      0.010      0.010     -0.030
12     -0.006      0.000      0.010      0.010     -0.030
13     -0.018      0.000      0.010      0.010     -0.030
14     -0.031      0.000      0.010      0.010     -0.030
15     -0.043      0.000      0.010      0.010     -0.030
16     -0.056      0.001      0.010      0.011     -0.030
17     -0.068      0.001      0.010      0.011     -0.030
18     -0.081      0.001      0.010      0.011     -0.030
19     -0.093      0.001      0.010      0.011     -0.030
20     -0.106      0.002      0.010      0.012     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 0.000
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = -3.988

CL(WING)=     0.0000
CD(WING)=     0.0103
CM(WING)=    -0.0300


WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.000
 J        CL        CDI        CPD         CD         CM
 1      0.980      0.073      0.010      0.083     -0.030
 2      1.000      0.070      0.010      0.080     -0.030
 3      1.022      0.066      0.010      0.076     -0.030
 4      1.047      0.062      0.010      0.072     -0.030
 5      1.074      0.057      0.010      0.067     -0.030
 6      1.081      0.055      0.010      0.065     -0.030
 7      1.082      0.053      0.010      0.063     -0.030
 8      1.082      0.051      0.010      0.061     -0.030
 9      1.081      0.050      0.010      0.060     -0.030
```

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10      1.077      0.049      0.010      0.059     -0.030
11      1.071      0.048      0.010      0.058     -0.030
12      1.057      0.046      0.010      0.056     -0.030
13      1.037      0.044      0.010      0.054     -0.030
14      1.013      0.043      0.010      0.053     -0.030
15      0.982      0.043      0.010      0.053     -0.030
16      0.945      0.043      0.010      0.053     -0.030
17      0.897      0.044      0.010      0.054     -0.030
18      0.834      0.046      0.010      0.056     -0.030
19      0.746      0.049      0.010      0.059     -0.030
20      0.601      0.051      0.010      0.061     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 12.444
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 8.456

CL(WING)=     1.0006
CD(WING)=     0.0641
CM(WING)=    -0.0300


WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.410
 J        CL        CDI        CPD        CD         CM
 1      1.360      0.139      0.010      0.149     -0.030
 2      1.390      0.133      0.010      0.143     -0.030
 3      1.424      0.126      0.010      0.136     -0.030
 4      1.461      0.119      0.010      0.129     -0.030
 5      1.502      0.109      0.010      0.119     -0.030
 6      1.512      0.105      0.010      0.115     -0.030
 7      1.516      0.102      0.010      0.112     -0.030
 8      1.518      0.100      0.010      0.110     -0.030
 9      1.518      0.097      0.010      0.107     -0.030
10      1.515      0.096      0.010      0.106     -0.030
11      1.509      0.094      0.010      0.104     -0.030
12      1.493      0.092      0.010      0.102     -0.030
13      1.470      0.091      0.010      0.101     -0.030
14      1.441      0.090      0.010      0.100     -0.030
15      1.403      0.090      0.010      0.100     -0.030
16      1.355      0.092      0.010      0.102     -0.030
17      1.293      0.096      0.010      0.106     -0.030
18      1.209      0.101      0.010      0.111     -0.030
19      1.090      0.108      0.010      0.118     -0.030
20      0.891      0.113      0.010      0.123     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 17.547
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 13.558

CL(WING)=     1.4109
CD(WING)=     0.1171
CM(WING)=    -0.0300


AIRPLANE LESS TAIL AERO COEFFICIENTS
-------- ---- ---- ---- ------------
FUS WIDTH = 3.833 FT     FUS LENGTH = 26.522 FT
ROOT QTR CHORD IS AT 31.800 PERCENT OF FUS
FUSELAGE FRONTAL AREA = 17.231 SQ FT
```

FUSELAGE ANGLE FROM WL TO FUS CL = -0.918
FACTOR TO ADJUST CM(FUS) PERKINS & HAIG EQ 5-31 TO MATCH WINDTUNNEL DATA = 1.000
TOTAL HORIZONTAL & VERTICAL TAIL AREA = 51.785 SQ FT
RANGE OF CL'S = -0.700 TO 1.500 IN INCREMENTS OF 0.100
AWO = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE

ANGLE(RW TO WL)	CL(WING)	CD(WING)	CD(F+T)	CD(A/P)	CM(WING)	CM(FUS)	CM(W+F)
-12.69918	-0.70043	0.03666	0.01658	0.05324	-0.03000	-0.03975	-0.06975
-11.45475	-0.60037	0.02968	0.01658	0.04625	-0.03000	-0.03461	-0.06461
-10.21032	-0.50030	0.02377	0.01658	0.04035	-0.03000	-0.02947	-0.05947
-8.96588	-0.40024	0.01893	0.01658	0.03551	-0.03000	-0.02434	-0.05434
-7.72145	-0.30018	0.01517	0.01658	0.03175	-0.03000	-0.01920	-0.04920
-6.47702	-0.20012	0.01249	0.01658	0.02907	-0.03000	-0.01406	-0.04406
-5.23259	-0.10006	0.01088	0.01658	0.02745	-0.03000	-0.00893	-0.03893
-3.98815	0.00000	0.01034	0.01658	0.02692	-0.03000	-0.00379	-0.03379
-2.74372	0.10006	0.01088	0.01658	0.02745	-0.03000	0.00135	-0.02865
-1.49929	0.20012	0.01249	0.01658	0.02907	-0.03000	0.00648	-0.02352
-0.25485	0.30018	0.01517	0.01658	0.03175	-0.03000	0.01162	-0.01838
0.98958	0.40024	0.01893	0.01658	0.03551	-0.03000	0.01676	-0.01324
2.23401	0.50030	0.02377	0.01658	0.04035	-0.03000	0.02189	-0.00811
3.47845	0.60037	0.02968	0.01658	0.04625	-0.03000	0.02703	-0.00297
4.72288	0.70043	0.03666	0.01658	0.05324	-0.03000	0.03217	0.00217
5.96731	0.80049	0.04472	0.01658	0.06129	-0.03000	0.03730	0.00730
7.21175	0.90055	0.05385	0.01658	0.07042	-0.03000	0.04244	0.01244
8.45618	1.00061	0.06405	0.01658	0.08063	-0.03000	0.04758	0.01758
9.70061	1.10067	0.07533	0.01658	0.09191	-0.03000	0.05271	0.02271
10.94505	1.20073	0.08769	0.01658	0.10426	-0.03000	0.05785	0.02785
12.18948	1.30079	0.10111	0.01658	0.11769	-0.03000	0.06299	0.03299
13.43391	1.40085	0.11562	0.01658	0.13219	-0.03000	0.06812	0.03812

EQUATION FOR AERO COEFFICIENTS FOR AIRPLANE LESS TAIL

CD=0.027+0.000*CL +0.054*CL^2 +0.000*CL^3 +0.000*CL^4

CM=-0.03379+0.051*CL

CL=0.320+0.080*ANGLE(RW TO WL)
WHERE ANGLE(RW TO WL)=ANGLE MEASURED FROM RELATIVE WIND TO WATERLINE

CM=-0.017+0.004*ANGLE(RW TO WL)

A.22 Wing Aerodynamic Coefficients - Landing

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LANDING CONFIGURATION

WING GEOMETRY CALCULATIONS
---- -------- ------------

 3 POINTS DEFINE THE LEADING EDGE
LEADING EDGE COORDINATES
 I       XLE       YLE
 1    45.000      0.000
 2    64.313     46.500
 3    72.000    201.000

 2 POINTS DEFINE THE TRAILING EDGE
TRAILING EDGE COORDINATES
 I       XTE       YTE
 1   146.000      0.000
 2   116.000    201.000

THE WING IS DIVIDED INTO  20 ELEMENTS OF DY
ELEMENT   YE     CE
 1      5.025     98.163
 2     15.075     92.489
 3     25.125     86.815
 4     35.175     81.141
 5     45.225     75.467
 6     55.275     73.000
 7     65.325     71.000
 8     75.375     69.000
 9     85.425     67.000
10     95.475     65.000
11    105.525     63.000
12    115.575     61.000
13    125.625     59.000
14    135.675     57.000
15    145.725     55.000
16    155.775     53.000
17    165.825     51.000
18    175.875     49.000
19    185.925     47.000
20    195.975     45.000

TOTAL AREA    = 26513.444
MAC           = 69.246
YBAR          = 87.854
XLE(MAC)      = 63.641
ASPECT RATIO  = 6.095
TOTAL SPAN    = 402.000
# OF ELEMENTS = 20.000
DY            = 10.050



ADDITIVE LIFT DISTRIBUTION
-------- ---- ------------

SELECTED WING STATIONS AND THEIR SLOPES OF LIFT CURVE

WING STA     SLOPE OF LIFT CURVE
     0.000          0.1075
   201.000          0.1075

ELEMENT      YE      CC(LA1)      C(LA1)  FOR CL=1.001
 1      5.025     91.056      0.928
 2     15.075     88.114      0.953
 3     25.125     85.065      0.980
 4     35.175     81.910      1.009
 5     45.225     78.644      1.042
```

6	55.275	76.869	1.053
7	65.325	75.208	1.059
8	75.375	73.424	1.064
9	85.425	71.507	1.067
10	95.475	69.448	1.068
11	105.525	67.236	1.067
12	115.575	64.852	1.063
13	125.625	62.277	1.056
14	135.675	59.479	1.043
15	145.725	56.419	1.026
16	155.775	53.034	1.001
17	165.825	49.229	0.965
18	175.875	44.827	0.915
19	185.925	39.454	0.839
20	195.975	31.830	0.707

BASIC LIFT DISTRIBUTION
----- ---- ------------

SELECTED WING STATIONS AND THEIR ANGLES FROM WL TO SECTION ZERO LIFT LINE

I	WING STA	ANGLE
1	0.000	20.000
2	46.500	19.577
3	109.279	19.028
4	109.280	4.028
5	201.000	1.900

AWO = 13.564

ELEM	REF ANGLE	Ao	CClb	Clb
1	19.954	6.390	33.717	0.343
2	19.863	6.299	31.314	0.339
3	19.771	6.208	28.966	0.334
4	19.680	6.116	26.674	0.329
5	19.589	6.025	24.438	0.324
6	19.500	5.936	23.293	0.319
7	19.412	5.849	22.319	0.314
8	19.325	5.761	21.365	0.310
9	19.237	5.673	20.429	0.305
10	19.149	5.585	19.512	0.300
11	19.061	5.497	18.614	0.295
12	3.882	-9.682	-31.745	-0.520
13	3.649	-9.915	-31.443	-0.533
14	3.416	-10.148	-31.092	-0.545
15	3.182	-10.381	-30.690	-0.558
16	2.949	-10.615	-30.238	-0.571
17	2.716	-10.848	-29.737	-0.583
18	2.483	-11.081	-29.185	-0.596
19	2.250	-11.314	-28.582	-0.608
20	2.017	-11.547	-27.930	-0.621

THERE IS NO DISCONTINUITY BETWEEN FLAP AND AILERON

ELEM	CC(lb)	C(lb) FAIRED
1	33.717	0.343
2	31.314	0.339
3	28.966	0.334
4	26.674	0.329
5	24.438	0.324
6	23.293	0.319
7	22.319	0.314
8	21.365	0.310
9	20.429	0.305
10	19.512	0.300
11	18.614	0.295
12	-31.745	-0.520
13	-31.443	-0.533
14	-31.092	-0.545

```
15    -30.690     -0.558
16    -30.238     -0.571
17    -29.737     -0.583
18    -29.185     -0.596
19    -28.582     -0.608
20    -27.930     -0.621

BCHECK = 0.000


STALL CALCULATIONS
----- ------------

SELECTED WING STAS , 2 CLMAXS & REYNOLDS NOS. AND CHORD

WING STA       CLMAX1          RN1         CLMAX2         RN2         CHORD
     0.000      2.360       3,000,000      2.620       9,000,000    101.000
    46.500      2.370       3,000,000      2.640       9,000,000     74.746
   109.279      2.440       3,000,000      2.660       9,000,000     62.253
   109.280      1.480       3,000,000      1.700       9,000,000     62.253
   201.000      1.500       3,000,000      1.740       9,000,000     44.000

FOR 70 MPH
I= 1     R3N(I)=5,514,282     C3LMAX(I)=2.501
I= 2     R3N(I)=4,080,896     C3LMAX(I)=2.443
I= 3     R3N(I)=3,398,818     C3LMAX(I)=2.464
I= 4     R3N(I)=3,398,818     C3LMAX(I)=1.503
I= 5     R3N(I)=2,402,261     C3LMAX(I)=1.456

CL MAX (STALL) FOR EACH ELEMENT
 J        YE(J)    ELEMENT CL MAX
 1       5.025      2.494
 2      15.075      2.482
 3      25.125      2.469
 4      35.175      2.457
 5      45.225      2.444
 6      55.275      2.446
 7      65.325      2.449
 8      75.375      2.453
 9      85.425      2.456
10      95.475      2.459
11     105.525      2.463
12     115.575      1.500
13     125.625      1.495
14     135.675      1.490
15     145.725      1.484
16     155.775      1.479
17     165.825      1.474
18     175.875      1.469
19     185.925      1.464
20     195.975      1.458

WING STALL CL DISTRIBUTION
WING STALL CL = 1.910

 J       YE(J)        CL(J)
 1       5.025      2.115
 2      15.075      2.158
 3      25.125      2.205
 4      35.175      2.257
 5      45.225      2.314
 6      55.275      2.330
 7      65.325      2.338
 8      75.375      2.342
 9      85.425      2.343
10      95.475      2.341
11     105.525      2.334
12     115.575      1.510
13     125.625      1.483
14     135.675      1.448
15     145.725      1.401
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16    155.775      1.341
17    165.825      1.261
18    175.875      1.152
19    185.925      0.995
20    195.975      0.730


WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.040
 2   109.279      0.040
 3   109.280      0.010
 4   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.450
 2   109.279     -0.450
 3   109.280     -0.030
 4   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 0.000
 J        CL         CDI         CPD         CD          CM
 1      0.343      0.019      0.040      0.059     -0.450
 2      0.339      0.019      0.040      0.059     -0.450
 3      0.334      0.018      0.040      0.058     -0.450
 4      0.329      0.018      0.040      0.058     -0.450
 5      0.324      0.017      0.040      0.057     -0.450
 6      0.319      0.017      0.040      0.057     -0.450
 7      0.314      0.016      0.040      0.056     -0.450
 8      0.310      0.016      0.040      0.056     -0.450
 9      0.305      0.015      0.040      0.055     -0.450
10      0.300      0.015      0.040      0.055     -0.450
11      0.295      0.014      0.040      0.054     -0.450
12     -0.520      0.044      0.010      0.054     -0.030
13     -0.533      0.046      0.010      0.056     -0.030
14     -0.545      0.048      0.010      0.058     -0.030
15     -0.558      0.051      0.010      0.061     -0.030
16     -0.571      0.053      0.010      0.063     -0.030
17     -0.583      0.055      0.010      0.065     -0.030
18     -0.596      0.058      0.010      0.068     -0.030
19     -0.608      0.060      0.010      0.070     -0.030
20     -0.621      0.063      0.010      0.073     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 13.564
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 0.000
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = -13.564

CL(WING)=     0.0000
CD(WING)=     0.0588
CM(WING)=    -0.3327


WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.040
 2   109.279      0.040
 3   109.280      0.010
 4   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.450
 2   109.279     -0.450
 3   109.280     -0.030
```

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 4   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.000
 J        CL        CDI        CPD        CD         CM
 1      1.271      0.156      0.040      0.196     -0.450
 2      1.291      0.152      0.040      0.192     -0.450
 3      1.314      0.147      0.040      0.187     -0.450
 4      1.338      0.143      0.040      0.183     -0.450
 5      1.366      0.137      0.040      0.177     -0.450
 6      1.372      0.135      0.040      0.175     -0.450
 7      1.374      0.132      0.040      0.172     -0.450
 8      1.374      0.130      0.040      0.170     -0.450
 9      1.372      0.128      0.040      0.168     -0.450
10      1.369      0.127      0.040      0.167     -0.450
11      1.363      0.125      0.040      0.165     -0.450
12      0.543     -0.022      0.010     -0.012     -0.030
13      0.523     -0.021      0.010     -0.011     -0.030
14      0.498     -0.020      0.010     -0.010     -0.030
15      0.468     -0.019      0.010     -0.009     -0.030
16      0.430     -0.016      0.010     -0.006     -0.030
17      0.382     -0.013      0.010     -0.003     -0.030
18      0.319     -0.009      0.010      0.001     -0.030
19      0.231     -0.004      0.010      0.006     -0.030
20      0.087      0.000      0.010      0.010     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 13.564
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 12.444
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = -1.120

CL(WING)=     1.0006
CD(WING)=     0.1126
CM(WING)=    -0.3327


WING AERO COEFFICIENT DISTRIBUTIONS
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TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.040
 2   109.279      0.040
 3   109.280      0.010
 4   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.450
 2   109.279     -0.450
 3   109.280     -0.030
 4   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.910
 J        CL        CDI        CPD        CD         CM
 1      2.115      0.387      0.040      0.427     -0.450
 2      2.158      0.376      0.040      0.416     -0.450
 3      2.205      0.364      0.040      0.404     -0.450
 4      2.257      0.350      0.040      0.390     -0.450
 5      2.314      0.334      0.040      0.374     -0.450
 6      2.330      0.326      0.040      0.366     -0.450
 7      2.338      0.321      0.040      0.361     -0.450
 8      2.342      0.316      0.040      0.356     -0.450
 9      2.343      0.313      0.040      0.353     -0.450
10      2.341      0.310      0.040      0.350     -0.450
11      2.334      0.308      0.040      0.348     -0.450
12      1.510      0.001      0.010      0.011     -0.030
13      1.483      0.001      0.010      0.011     -0.030
14      1.448      0.004      0.010      0.014     -0.030
15      1.401      0.009      0.010      0.019     -0.030
16      1.341      0.016      0.010      0.026     -0.030
17      1.261      0.026      0.010      0.036     -0.030
18      1.152      0.040      0.010      0.050     -0.030
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19      0.995      0.056      0.010      0.066     -0.030
20      0.730      0.069      0.010      0.079     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 13.564
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 23.769
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 10.205

CL(WING)=     1.9112
CD(WING)=     0.2548
CM(WING)=    -0.3327


AIRPLANE LESS TAIL AERO COEFFICIENTS
-------- ---- ---- ---- ------------
FUS WIDTH = 3.833 FT     FUS LENGTH = 26.522 FT
ROOT QTR CHORD IS AT 31.800 PERCENT OF FUS
FUSELAGE FRONTAL AREA = 17.231 SQ FT
FUSELAGE ANGLE FROM WL TO FUS CL = -0.918
FACTOR TO ADJUST CM(FUS) PERKINS & HAIG EQ 5-31 TO MATCH WINDTUNNEL DATA = 1.000
TOTAL HORIZONTAL & VERTICAL TAIL AREA = 51.785 SQ FT
RANGE OF CL'S = -0.700 TO 1.500 IN INCREMENTS OF 0.100
AWO = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE
```

ANGLE(RW TO WL)	CL(WING)	CD(WING)	CD(F+T)	CD(A/P)	CM(WING)	CM(FUS)	CM(W+F)
-22.27492	-0.70043	0.08516	0.01658	0.10174	-0.33265	-0.03975	-0.37240
-21.03049	-0.60037	0.07818	0.01658	0.09475	-0.33265	-0.03461	-0.36726
-19.78606	-0.50030	0.07227	0.01658	0.08885	-0.33265	-0.02947	-0.36212
-18.54162	-0.40024	0.06744	0.01658	0.08401	-0.33265	-0.02434	-0.35699
-17.29719	-0.30018	0.06368	0.01658	0.08025	-0.33265	-0.01920	-0.35185
-16.05276	-0.20012	0.06099	0.01658	0.07757	-0.33265	-0.01406	-0.34671
-14.80832	-0.10006	0.05938	0.01658	0.07595	-0.33265	-0.00893	-0.34158
-13.56389	0.00000	0.05884	0.01658	0.07542	-0.33265	-0.00379	-0.33644
-12.31946	0.10006	0.05938	0.01658	0.07595	-0.33265	0.00135	-0.33130
-11.07502	0.20012	0.06099	0.01658	0.07757	-0.33265	0.00648	-0.32617
-9.83059	0.30018	0.06368	0.01658	0.08025	-0.33265	0.01162	-0.32103
-8.58616	0.40024	0.06744	0.01658	0.08401	-0.33265	0.01676	-0.31589
-7.34173	0.50030	0.07227	0.01658	0.08885	-0.33265	0.02189	-0.31076
-6.09729	0.60037	0.07818	0.01658	0.09475	-0.33265	0.02703	-0.30562
-4.85286	0.70043	0.08516	0.01658	0.10174	-0.33265	0.03217	-0.30048
-3.60843	0.80049	0.09322	0.01658	0.10979	-0.33265	0.03730	-0.29535
-2.36399	0.90055	0.10235	0.01658	0.11892	-0.33265	0.04244	-0.29021
-1.11956	1.00061	0.11255	0.01658	0.12913	-0.33265	0.04758	-0.28507
0.12487	1.10067	0.12383	0.01658	0.14041	-0.33265	0.05271	-0.27994
1.36931	1.20073	0.13619	0.01658	0.15276	-0.33265	0.05785	-0.27480
2.61374	1.30079	0.14962	0.01658	0.16619	-0.33265	0.06299	-0.26966
3.85817	1.40085	0.16412	0.01658	0.18069	-0.33265	0.06812	-0.26453

```
EQUATION FOR AERO COEFFICIENTS FOR AIRPLANE LESS TAIL

CD=0.075+0.000*CL +0.054*CL^2 +0.000*CL^3 +0.000*CL^4

CM=-0.33644+0.051*CL

CL=1.090+0.080*ANGLE(RW TO WL)
WHERE ANGLE(RW TO WL)=ANGLE MEASURED FROM RELATIVE WIND TO WATERLINE

CM=-0.280+0.004*ANGLE(RW TO WL)
```

A.23 V-n Data

GEOMETRY

MAC	XTC	XTF	XW	ZW	S	CATEGORY
69.246	253.591	261.041	80.953	87.725	184.121	NORMAL

STRUCTURAL SPEEDS

VA	VC	VD	VF	VPF	MC	MD	MAX WT
121.300	170.000	212.500	105.500	105.500	0.323	0.403	3400.000

LIMIT POSITIVE LOAD FACTOR = 3.800

ALTITUDES

0	12000	18000

AERO COEFFICIENTS AND CG'S
AERO COEFFICIENTS WERE OBTAINED AT MACH = 0.100

FOR CRUISE

STALL CL = 1.400
NEG STALL CL = -0.590

LIFT COEFFICIENTS ARE	0.320	0.080	0.000	0.000	0.000
DRAG COEFFICIENTS ARE	0.027	0.000	0.054	0.000	0.000
PITCHING MOMENT COEFFICIENTS ARE	-0.017	0.004	0.000	0.000	0.000

CG	WT	XCG	ZCG
CG1	3400.000	85.100	93.000
CG2	3400.000	77.490	93.000
CG3	2800.000	72.640	92.000
CG4	2063.000	73.090	90.730

FOR LANDING

STALL CL = 1.600
NEG STALL CL = -0.410

LIFT COEFFICIENTS ARE	1.090	0.080	0.000	0.000	0.000
DRAG COEFFICIENTS ARE	0.075	0.002	0.054	0.000	0.000
PITCHING MOMENT COEFFICIENTS ARE	-0.280	0.004	0.000	0.000	0.000

CG	WT	XCG	ZCG
CG5	3400.000	85.100	93.000
CG6	3400.000	77.490	93.000
CG7	2800.000	72.640	92.000
CG8	2063.000	73.090	90.730

CRUISE CONFIGURATION AT 0 FEET BALANCED FLIGHT LOAD DATA

FOR CG1 FS= 85.100 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
1	STALL 1G	61.556	1.00	13.38	1.004	1.39	5973	3269	130	-458
2	STALL +N	115.412	3.80	14.64	1.016	1.50	24290	12410	511	-1961
3	MAN A	121.300	3.80	12.82	1.017	1.36	22186	12416	488	-1632
4	MAN C	170.000	3.80	4.57	1.035	0.70	2257	12593	325	-44
5	MAN D	212.500	3.80	1.56	1.056	0.45	-20390	12737	170	724
6	MAN -D	212.500	0.00	-3.63	1.056	0.02	-62927	377	-389	787
7	MAN -C	170.000	-1.52	-7.04	1.035	-0.26	-57472	-4727	-457	-27
8	STALL -N	111.248	-1.52	-11.80	1.014	-0.63	-34584	-4864	-305	-633

9	STALL -1G	91.473	-1.00	-11.62	1.010	-0.61	-23030	-3189	-203	-403
10	GUST +C	170.000	3.95	4.90	1.035	0.72	3986	13094	350	-122
11	GUST +D	212.500	2.87	0.30	1.056	0.35	-30694	9741	30	891
12	GUST -D	212.500	-0.88	-4.83	1.056	-0.09	-72799	-2481	-511	565
13	GUST -C	170.000	-1.96	-8.00	1.035	-0.34	-62401	-6142	-513	-258
14	BAL A	121.300	1.00	0.58	1.017	0.37	-9317	3373	19	281
15	BAL C	170.000	1.00	-1.51	1.035	0.20	-29003	3513	-105	617
16	BAL D	212.500	1.00	-2.25	1.056	0.13	-51673	3643	-246	931
17	ST ROL A	121.300	2.53	7.25	1.017	0.91	7853	8345	265	-400
18	ST ROL C	170.000	2.53	1.81	1.035	0.47	-11898	8486	125	433
19	ST ROL D	212.500	2.53	-0.16	1.056	0.31	-34525	8627	-22	928
20	AC ROLL	115.412	3.25	11.92	1.016	1.28	17957	10642	408	-1261

FOR CG2 FS= 77.490 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
21	STALL 1G	62.754	1.00	13.46	1.005	1.40	6264	3413	-17	-482
22	STALL +N	117.881	3.80	14.64	1.016	1.50	25364	12953	-49	-2047
23	MAN A	121.300	3.80	13.60	1.017	1.42	24215	12983	-62	-1855
24	MAN C	170.000	3.80	4.94	1.035	0.73	4157	13143	-231	-130
25	MAN D	212.500	3.80	1.80	1.056	0.47	-18391	13318	-387	681
26	MAN -D	212.500	0.00	-3.62	1.056	0.02	-62905	384	-388	787
27	MAN -C	170.000	-1.52	-7.20	1.035	-0.27	-58250	-4951	-232	-61
28	STALL -N	113.625	-1.52	-11.80	1.015	-0.63	-36094	-5079	-85	-660
29	STALL -1G	93.371	-1.00	-11.68	1.010	-0.62	-24092	-3350	-58	-427
30	GUST +C	170.000	3.95	5.29	1.035	0.76	5989	13673	-228	-218
31	GUST +D	212.500	2.87	0.48	1.056	0.36	-29237	10165	-392	873
32	GUST -D	212.500	-0.87	-4.87	1.056	-0.09	-73165	-2587	-381	555
33	GUST -C	170.000	-1.96	-8.20	1.035	-0.36	-63391	-6426	-224	-309
34	BAL A	121.300	1.00	0.78	1.017	0.38	-8803	3523	-127	273
35	BAL C	170.000	1.00	-1.41	1.035	0.20	-28489	3662	-252	618
36	BAL D	212.500	1.00	-2.19	1.056	0.14	-51105	3808	-393	935
37	ST ROL A	121.300	2.53	7.74	1.017	0.95	9114	8707	-105	-484
38	ST ROL C	170.000	2.53	2.06	1.035	0.49	-10643	8851	-247	403
39	ST ROL D	212.500	2.54	0.00	1.056	0.32	-33193	9014	-393	917
40	AC ROLL	117.881	3.25	11.96	1.016	1.29	18851	11136	-72	-1326

FOR CG3 FS= 72.640 WL= 92.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
41	STALL 1G	58.147	1.00	13.30	1.004	1.38	5279	2902	-95	-403
42	STALL +N	109.036	3.80	14.51	1.014	1.49	21356	10986	-346	-1716
43	MAN A	121.300	3.80	10.89	1.017	1.20	17239	11023	-384	-1139
44	MAN C	170.000	3.80	3.62	1.035	0.62	-2631	11177	-532	154
45	MAN D	212.500	3.80	0.97	1.056	0.40	-25234	11329	-679	815
46	MAN -D	212.500	0.00	-3.62	1.056	0.02	-62905	384	-384	787
47	MAN -C	170.000	-1.52	-6.65	1.035	-0.23	-55459	-4148	-117	58
48	STALL -N	104.911	-1.52	-11.77	1.013	-0.63	-30653	-4296	41	-556
49	STALL -1G	86.310	-1.00	-11.58	1.009	-0.61	-20435	-2820	25	-354
50	GUST +C	170.000	4.42	4.81	1.035	0.72	3489	12950	-573	-99
51	GUST +D	212.500	3.17	0.20	1.056	0.34	-31510	9504	-632	900
52	GUST -D	212.500	-1.17	-5.04	1.056	-0.10	-74538	-2984	-287	517
53	GUST -C	170.000	-2.42	-8.41	1.035	-0.37	-64515	-6747	-38	-369
54	BAL A	121.300	1.00	0.10	1.017	0.33	-10549	3015	-204	296
55	BAL C	170.000	1.00	-1.77	1.035	0.17	-30323	3129	-326	613
56	BAL D	212.500	1.00	-2.42	1.056	0.12	-52991	3260	-464	920
57	ST ROL A	121.300	2.53	5.97	1.017	0.80	4566	7399	-310	-203
58	ST ROL C	170.000	2.53	1.17	1.035	0.42	-15194	7528	-442	502
59	ST ROL D	212.500	2.54	-0.56	1.056	0.27	-37763	7686	-584	949
60	AC ROLL	109.036	3.24	11.77	1.014	1.27	15671	9398	-319	-1094

FOR CG4 FS= 73.090 WL= 90.730

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
61	STALL 1G	50.121	1.00	13.15	1.003	1.37	3853	2137	-67	-293
62	STALL +N	94.460	3.80	14.20	1.010	1.46	15460	8087	-246	-1228
63	MAN A	121.300	3.80	6.99	1.017	0.89	7178	8151	-309	-357
64	MAN C	170.000	3.80	1.67	1.035	0.46	-12621	8276	-438	450
65	MAN D	212.500	3.80	-0.25	1.056	0.30	-35219	8425	-578	934
66	MAN -D	212.500	0.00	-3.62	1.056	0.02	-62905	384	-378	787
67	MAN -C	170.000	-1.52	-5.87	1.035	-0.16	-51414	-2982	-158	209

68	STALL -N	91.059	-1.52	-11.58	1.010	-0.61	-22762	-3143	17	-395
69	STALL -1G	74.626	-1.00	-11.48	1.006	-0.60	-15156	-2074	11	-256
70	GUST +C	170.000	5.24	3.71	1.035	0.63	-2150	11317	-507	136
71	GUST +D	212.500	3.68	-0.36	1.056	0.29	-36102	8168	-571	940
72	GUST -D	212.500	-1.69	-5.13	1.056	-0.11	-75269	-3195	-286	496
73	GUST -C	170.000	-3.23	-8.32	1.035	-0.37	-64037	-6611	-61	-343
74	BAL A	121.300	1.00	-0.92	1.017	0.25	-13181	2249	-176	314
75	BAL C	170.000	1.00	-2.28	1.035	0.13	-32962	2362	-295	598
76	BAL D	212.500	1.00	-2.73	1.056	0.09	-55605	2501	-432	894
77	ST ROL A	121.300	2.54	3.40	1.017	0.60	-2047	5485	-252	98
78	ST ROL C	170.000	2.54	-0.11	1.035	0.31	-21798	5608	-375	592
79	ST ROL D	212.500	2.53	-1.38	1.056	0.20	-44464	5738	-512	963
80	AC ROLL	94.460	3.24	11.50	1.010	1.25	11280	6917	-226	-778

LANDING CONFIGURATION AT 0 FEET BALANCED FLIGHT LOAD DATA

FOR CG5 FS= 85.100 WL= 93.000

CASE	CONDITION	V1(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
81	STAL 2/3G	49.041	0.67	5.88	1.003	1.56	-26666	2360	-98	73
82	STALL 1GL	59.591	1.00	6.16	1.004	1.58	-39197	3537	-142	94
83	STALL 2G	82.550	2.00	6.94	1.008	1.65	-74269	7067	-259	103
84	MAN 2G VF	105.500	2.00	-0.25	1.013	1.07	-135176	7428	-623	998
85	MAN 0G VF	105.500	0.00	-11.67	1.013	0.15	-157314	909	-895	730
86	GUST VF	105.500	1.90	-0.80	1.013	1.03	-136248	7109	-638	1029
87	GUST -VF	105.500	0.09	-11.14	1.013	0.19	-156291	1204	-883	785
88	BAL VF	105.500	1.00	-5.91	1.013	0.61	-146154	4170	-766	1106
89	BAL 1.4VSF	83.428	1.00	-2.54	1.008	0.89	-87296	3822	-427	688
90	LEV LAND	48.441	0.66	6.30	1.003	1.59	-25847	2353	-93	57

ANG= 6.410

FOR CG6 FS= 77.490 WL= 93.000

CASE	CONDITION	V1(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
91	STAL 2/3G	50.092	0.67	5.88	1.003	1.56	-27821	2463	-200	76
92	STALL 1GL	60.825	1.00	6.16	1.004	1.58	-40836	3685	-295	98
93	STALL 2G	84.243	2.00	6.94	1.008	1.65	-77344	7361	-563	107
94	MAN 2G VF	105.500	2.00	0.25	1.013	1.11	-134202	7717	-904	965
95	MAN 0G VF	105.500	0.00	-11.67	1.013	0.15	-157314	909	-895	730
96	GUST VF	105.500	1.90	-0.36	1.013	1.06	-135381	7367	-905	1004
97	GUST -VF	105.500	0.10	-11.09	1.013	0.20	-156202	1230	-897	790
98	BAL VF	105.500	1.00	-5.67	1.013	0.63	-145693	4306	-907	1111
99	BAL 1.4VSF	85.155	1.00	-2.48	1.008	0.89	-90870	4005	-591	715
100	LEV LAND	49.792	0.66	6.05	1.003	1.57	-27416	2455	-198	69

ANG= 6.235

FOR CG7 FS= 72.640 WL= 92.000

CASE	CONDITION	V1(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
101	STAL 2/3G	46.334	0.67	5.71	1.002	1.55	-23867	2088	-220	70
102	STALL 1GL	56.105	1.00	6.05	1.004	1.57	-34807	3117	-324	88
103	STALL 2G	77.837	2.00	6.75	1.007	1.63	-66235	6223	-629	109
104	MAN 2G VF	105.500	2.00	-1.59	1.013	0.96	-137771	6655	-1049	1066
105	MAN 0G VF	105.500	0.00	-11.69	1.013	0.15	-157365	894	-891	728
106	GUST VF	105.500	2.04	-1.40	1.013	0.98	-137398	6766	-1052	1058
107	GUST -VF	105.500	-0.05	-11.93	1.013	0.13	-157829	761	-887	701
108	BAL VF	105.500	1.00	-6.58	1.013	0.56	-147448	3788	-974	1087
109	BAL 1.4VSF	78.548	1.00	-2.62	1.007	0.88	-77464	3364	-573	611
110	LEV LAND	45.434	0.67	6.49	1.002	1.61	-22670	2090	-214	45

ANG= 6.511

FOR CG8 FS= 73.090 WL= 90.730

CASE	CONDITION	V1(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
111	STAL 2/3G	39.923	0.66	5.46	1.002	1.53	-17789	1530	-160	57
112	STALL 1GL	48.310	1.00	5.88	1.003	1.56	-25877	2290	-235	71
113	STALL 2G	67.360	2.00	6.44	1.005	1.61	-49857	4586	-459	102
114	MAN 2G VF	105.500	2.00	-4.24	1.013	0.75	-142910	5129	-993	1124
115	MAN 0G VF	105.500	0.00	-11.72	1.013	0.14	-157429	876	-886	724
116	GUST VF	105.500	2.29	-3.18	1.013	0.83	-140850	5740	-1007	1113
117	GUST -VF	105.500	-0.29	-12.83	1.013	0.06	-159568	261	-869	596
118	BAL VF	105.500	1.00	-7.95	1.013	0.45	-150109	3006	-941	1028
119	BAL 1.4VSF	67.633	1.00	-2.70	1.005	0.87	-57491	2476	-417	454
120	LEV LAND	38.423	0.67	6.94	1.002	1.64	-16101	1528	-150	22

ANG= 7.136

CRUISE CONFIGURATION AT 12000 FEET BALANCED FLIGHT LOAD DATA

FOR CG1 FS= 85.100 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
121	STALL 1G	60.921	1.00	13.75	1.007	1.42	6110	3274	132	-476
122	STALL +N	113.953	3.80	14.98	1.024	1.54	24736	12394	515	-2014
123	MAN A	121.300	3.80	12.69	1.028	1.36	22183	12419	488	-1604
124	MAN C	170.000	3.80	4.48	1.056	0.70	2276	12599	324	-24
125	MAN D	212.404	3.80	1.52	1.093	0.45	-20274	12753	171	733
126	MAN -D	212.404	0.00	-3.50	1.093	0.02	-62794	401	-387	787
127	MAN -C	170.000	-1.52	-6.89	1.056	-0.26	-57433	-4716	-457	-13
128	STALL -N	109.892	-1.52	-11.88	1.022	-0.65	-34114	-4851	-302	-639
129	STALL -1G	90.082	-1.00	-11.80	1.015	-0.63	-22684	-3191	-200	-415
130	GUST +C	170.000	4.23	5.39	1.056	0.77	7087	13990	394	-246
131	GUST +D	212.404	3.08	0.57	1.093	0.37	-28337	10409	61	865
132	GUST -D	212.404	-1.08	-4.94	1.093	-0.11	-75018	-3138	-538	510
133	GUST -C	170.000	-2.23	-8.44	1.056	-0.39	-65561	-7048	-550	-404
134	BAL A	121.300	1.00	0.57	1.028	0.37	-9334	3368	19	281
135	BAL C	170.000	1.00	-1.49	1.056	0.19	-29057	3498	-106	615
136	BAL D	212.404	1.00	-2.18	1.093	0.13	-51606	3647	-245	925
137	ST ROL A	121.300	2.53	7.19	1.028	0.91	7879	8353	265	-391
138	ST ROL C	170.000	2.54	1.79	1.056	0.47	-11843	8502	125	437
139	ST ROL D	212.404	2.53	-0.15	1.093	0.31	-34431	8638	-21	926
140	AC ROLL	113.953	3.25	12.21	1.024	1.32	18404	10630	412	-1302

FOR CG2 FS= 77.490 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
141	STALL 1G	62.090	1.00	13.75	1.007	1.42	6349	3402	-16	-495
142	STALL +N	116.396	3.80	15.02	1.025	1.55	25925	12964	-44	-2112
143	MAN A	121.300	3.80	13.47	1.028	1.42	24221	12989	-63	-1825
144	MAN C	170.000	3.80	4.84	1.056	0.73	4182	13150	-232	-108
145	MAN D	212.404	3.80	1.74	1.093	0.47	-18384	13302	-387	694
146	MAN -D	212.404	0.00	-3.50	1.093	0.02	-62794	401	-388	787
147	MAN -C	170.000	-1.52	-7.04	1.056	-0.27	-58231	-4945	-232	-48
148	STALL -N	112.251	-1.52	-11.88	1.023	-0.65	-35621	-5069	-83	-667
149	STALL -1G	92.057	-1.00	-11.80	1.016	-0.63	-23701	-3336	-56	-434
150	GUST +C	170.000	4.23	5.81	1.056	0.81	9254	14617	-224	-357
151	GUST +D	212.404	3.08	0.75	1.093	0.39	-26756	10869	-391	843
152	GUST -D	212.404	-1.08	-5.00	1.093	-0.11	-75513	-3281	-379	496
153	GUST -C	170.000	-2.23	-8.65	1.056	-0.41	-66676	-7367	-220	-465
154	BAL A	121.300	1.00	0.79	1.028	0.38	-8754	3537	-127	273
155	BAL C	170.000	1.00	-1.39	1.056	0.20	-28534	3650	-252	616
156	BAL D	212.404	1.00	-2.12	1.093	0.14	-51088	3797	-393	929
157	ST ROL A	121.300	2.54	7.69	1.028	0.95	9189	8729	-105	-478
158	ST ROL C	170.000	2.54	2.03	1.056	0.49	-10572	8871	-247	408
159	ST ROL D	212.404	2.53	0.00	1.093	0.32	-33170	9004	-393	916
160	AC ROLL	116.396	3.25	12.24	1.025	1.32	19294	11117	-68	-1365

FOR CG3 FS= 72.640 WL= 92.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
161	STALL 1G	57.441	1.00	13.60	1.006	1.41	5343	2886	-93	-414
162	STALL +N	107.482	3.80	14.94	1.021	1.53	21830	10976	-342	-1778
163	MAN A	121.300	3.80	10.77	1.028	1.20	17192	11012	-385	-1114
164	MAN C	170.000	3.80	3.55	1.056	0.62	-2611	11183	-532	168
165	MAN D	212.404	3.80	0.94	1.093	0.40	-25186	11325	-679	820
166	MAN -D	212.404	0.00	-3.50	1.093	0.02	-62861	381	-383	785
167	MAN -C	170.000	-1.52	-6.52	1.056	-0.23	-55462	-4148	-118	67
168	STALL -N	103.685	-1.52	-11.88	1.020	-0.64	-30314	-4302	44	-566
169	STALL -1G	85.289	-1.00	-11.72	1.013	-0.63	-20208	-2825	27	-363
170	GUST +C	170.000	4.78	5.39	1.056	0.77	7087	13990	-598	-246
171	GUST +D	212.404	3.43	0.51	1.093	0.36	-28852	10259	-651	871
172	GUST -D	212.404	-1.43	-5.18	1.093	-0.13	-77074	-3732	-265	451
173	GUST -C	170.000	-2.79	-8.94	1.056	-0.43	-68199	-7802	-5	-551
174	BAL A	121.300	1.00	0.07	1.028	0.33	-10635	2990	-203	297

175	BAL C	170.000	1.00	-1.73	1.056	0.17	-30331	3127	-326	611
176	BAL D	212.404	1.00	-2.33	1.093	0.12	-52906	3269	-464	915
177	ST ROL A	121.300	2.53	5.92	1.028	0.80	4583	7404	-310	-196
178	ST ROL C	170.000	2.53	1.15	1.056	0.42	-15186	7530	-442	505
179	ST ROL D	212.404	2.53	-0.54	1.093	0.27	-37753	7672	-583	946
180	AC ROLL	107.482	3.25	12.16	1.021	1.31	16185	9403	-316	-1144

FOR CG4 FS= 73.090 WL= 90.730

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
181	STALL 1G	49.573	1.00	13.38	1.004	1.39	3874	2120	-66	-297
182	STALL +N	93.029	3.80	14.69	1.016	1.51	15859	8085	-243	-1283
183	MAN A	121.300	3.80	6.92	1.028	0.89	7194	8156	-310	-348
184	MAN C	170.000	3.80	1.64	1.056	0.46	-12643	8270	-438	455
185	MAN D	212.404	3.80	-0.24	1.093	0.30	-35206	8412	-577	932
186	MAN -D	212.404	0.00	-3.50	1.093	0.02	-62861	381	-378	785
187	MAN -C	170.000	-1.52	-5.74	1.056	-0.16	-51371	-2969	-159	217
188	STALL -N	89.588	-1.52	-11.80	1.015	-0.63	-22433	-3156	20	-410
189	STALL -1G	73.508	-1.00	-11.68	1.010	-0.62	-14930	-2076	12	-265
190	GUST +C	170.000	5.79	4.40	1.056	0.69	1856	12477	-533	-6
191	GUST +D	212.404	4.07	-0.01	1.093	0.32	-33260	8978	-591	917
192	GUST -D	212.404	-2.07	-5.29	1.093	-0.14	-77993	-3998	-265	423
193	GUST -C	170.000	-3.79	-8.93	1.056	-0.43	-68123	-7780	-29	-546
194	BAL A	121.300	1.00	-0.94	1.028	0.24	-13246	2230	-176	314
195	BAL C	170.000	1.00	-2.23	1.056	0.13	-32944	2368	-296	597
196	BAL D	212.404	1.00	-2.65	1.093	0.09	-55576	2495	-431	889
197	ST ROL A	121.300	2.53	3.36	1.028	0.59	-2070	5479	-252	102
198	ST ROL C	170.000	2.54	-0.11	1.056	0.31	-21801	5607	-375	592
199	ST ROL D	212.404	2.53	-1.33	1.093	0.20	-44422	5734	-512	957
200	AC ROLL	93.029	3.24	11.90	1.016	1.28	11634	6905	-223	-816

CRUISE CONFIGURATION AT 18000 FEET BALANCED FLIGHT LOAD DATA

FOR CG1 FS= 85.100 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
201	STALL 1G	60.520	1.00	13.94	1.009	1.44	6165	3269	132	-484
202	STALL +N	113.453	3.80	15.07	1.031	1.56	24934	12404	517	-2026
203	MAN A	121.300	3.80	12.58	1.036	1.36	22166	12417	487	-1581
204	MAN C	150.838	3.80	6.67	1.057	0.88	10865	12539	388	-481
205	MAN D	188.197	3.80	2.89	1.093	0.57	-6789	12663	260	349
206	MAN -D	188.197	0.00	-3.50	1.093	0.02	-49297	315	-304	618
207	MAN -C	150.838	-1.52	-7.77	1.057	-0.33	-48855	-4758	-402	-176
208	STALL -N	109.480	-1.52	-11.88	1.029	-0.65	-34011	-4858	-301	-640
209	STALL -1G	89.623	-1.00	-11.88	1.019	-0.64	-22632	-3210	-200	-422
210	GUST +C	150.838	3.96	7.11	1.057	0.92	12670	13059	415	-591
211	GUST +D	188.197	2.90	1.38	1.093	0.44	-16846	9741	121	594
212	GUST -D	188.197	-0.90	-5.03	1.093	-0.12	-59521	-2645	-430	383
213	GUST -C	150.838	-1.96	-8.99	1.057	-0.44	-53900	-6202	-459	-445
214	BAL A	121.300	1.00	0.57	1.036	0.37	-9323	3371	19	281
215	BAL C	150.838	1.00	-0.92	1.057	0.24	-20524	3437	-52	484
216	BAL D	188.197	1.00	-1.81	1.093	0.16	-38084	3569	-161	742
217	ST ROL A	121.300	2.53	7.13	1.036	0.91	7865	8350	264	-382
218	ST ROL C	150.838	2.53	3.23	1.057	0.59	-3340	8431	182	177
219	ST ROL D	188.197	2.53	0.75	1.093	0.39	-21005	8533	65	662
220	AC ROLL	113.453	3.25	12.30	1.031	1.33	18610	10642	413	-1313

FOR CG2 FS= 77.490 WL= 93.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
221	STALL 1G	61.748	1.00	13.94	1.009	1.44	6421	3404	-15	-504
222	STALL +N	115.884	3.80	15.10	1.032	1.56	26143	12977	-43	-2125
223	MAN A	121.300	3.80	13.38	1.036	1.42	24249	13000	-64	-1805
224	MAN C	150.838	3.80	7.13	1.057	0.92	12750	13082	-167	-596
225	MAN D	188.197	3.80	3.17	1.093	0.60	-4915	13207	-296	291
226	MAN -D	188.197	0.00	-3.50	1.093	0.02	-49297	315	-305	618
227	MAN -C	150.838	-1.52	-7.97	1.057	-0.35	-49672	-4992	-177	-217
228	STALL -N	111.771	-1.52	-11.88	1.030	-0.65	-35482	-5073	-82	-668
229	STALL -1G	91.475	-1.00	-11.88	1.020	-0.64	-23592	-3348	-55	-441
230	GUST +C	150.838	3.97	7.60	1.057	0.96	14707	13644	-163	-722

231	GUST +D	188.197	2.90	1.60	1.093	0.46	-15348	10177	-304	564
232	GUST -D	188.197	-0.90	-5.10	1.093	-0.12	-59948	-2768	-297	371
233	GUST -C	150.838	-1.96	-9.24	1.057	-0.46	-54932	-6496	-169	-506
234	BAL A	121.300	1.00	0.75	1.036	0.38	-8835	3513	-127	274
235	BAL C	150.838	1.00	-0.78	1.057	0.25	-19930	3610	-199	482
236	BAL D	188.197	1.00	-1.74	1.093	0.17	-37591	3712	-309	744
237	ST ROL A	121.300	2.54	7.63	1.036	0.95	9185	8729	-105	-468
238	ST ROL C	150.838	2.53	3.55	1.057	0.62	-2050	8805	-189	132
239	ST ROL D	188.197	2.54	0.96	1.093	0.40	-19647	8927	-306	642
240	AC ROLL	115.884	3.25	12.30	1.032	1.33	19456	11114	-67	-1371

FOR CG3 FS= 72.640 WL= 92.000

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
241	STALL 1G	57.135	1.00	13.75	1.008	1.42	5381	2882	-93	-419
242	STALL +N	106.877	3.80	15.07	1.027	1.55	22021	10976	-340	-1795
243	MAN A	121.300	3.80	10.71	1.036	1.20	17251	11030	-385	-1103
244	MAN C	150.838	3.80	5.47	1.057	0.78	5917	11112	-473	-210
245	MAN D	188.197	3.80	2.15	1.093	0.51	-11701	11236	-592	482
246	MAN -D	188.197	0.00	-3.51	1.093	0.01	-49391	287	-301	616
247	MAN -C	150.838	-1.52	-7.30	1.057	-0.29	-46912	-4200	-64	-85
248	STALL -N	103.065	-1.52	-11.93	1.025	-0.65	-30155	-4309	45	-571
249	STALL -1G	84.769	-1.00	-11.80	1.017	-0.64	-20116	-2834	28	-369
250	GUST +C	150.838	4.49	7.13	1.057	0.92	12750	13082	-516	-596
251	GUST +D	188.197	3.23	1.31	1.093	0.43	-17325	9602	-551	603
252	GUST -D	188.197	-1.24	-5.35	1.093	-0.15	-61655	-3262	-198	319
253	GUST -C	150.838	-2.49	-9.66	1.057	-0.49	-56660	-6989	22	-612
254	BAL A	121.300	1.00	0.10	1.036	0.33	-10544	3016	-204	296
255	BAL C	150.838	1.00	-1.22	1.057	0.22	-21755	3079	-273	485
256	BAL D	188.197	1.00	-2.02	1.093	0.14	-39442	3174	-381	734
257	ST ROL A	121.300	2.53	5.88	1.036	0.80	4592	7407	-310	-190
258	ST ROL C	150.838	2.53	2.43	1.057	0.52	-6651	7470	-386	278
259	ST ROL D	188.197	2.54	0.27	1.093	0.34	-24215	7599	-500	702
260	AC ROLL	106.877	3.25	12.27	1.027	1.32	16382	9405	-314	-1159

FOR CG4 FS= 73.090 WL= 90.730

CASE	CONDITION	V(EAS)	NZ	ALPHA	G CORR	CL	M(W+F)	LZW	LT	DX
261	STALL 1G	49.347	1.00	13.60	1.006	1.41	3941	2129	-66	-305
262	STALL +N	92.377	3.80	14.89	1.020	1.53	16028	8081	-242	-1304
263	MAN A	121.300	3.80	6.86	1.036	0.89	7175	8151	-310	-339
264	MAN C	150.838	3.80	3.07	1.057	0.58	-4040	8228	-384	200
265	MAN D	188.197	3.80	0.65	1.093	0.38	-21716	8326	-494	672
266	MAN -D	188.197	0.00	-3.51	1.093	0.01	-49391	287	-296	616
267	MAN -C	150.838	-1.52	-6.32	1.057	-0.21	-42864	-3036	-107	84
268	STALL -N	89.060	-1.52	-11.84	1.019	-0.64	-22281	-3150	21	-412
269	STALL -1G	73.122	-1.00	-11.72	1.012	-0.62	-14844	-2074	13	-266
270	GUST +C	150.838	5.45	5.97	1.057	0.82	7981	11708	-461	-318
271	GUST +D	188.197	3.85	0.71	1.093	0.38	-21315	8442	-497	666
272	GUST -D	188.197	-1.86	-5.55	1.093	-0.16	-62947	-3635	-195	278
273	GUST -C	150.838	-3.45	-9.78	1.057	-0.50	-57151	-7129	5	-643
274	BAL A	121.300	1.00	-0.92	1.036	0.24	-13224	2237	-176	313
275	BAL C	150.838	1.00	-1.86	1.057	0.16	-24394	2312	-244	479
276	BAL D	188.197	1.00	-2.41	1.093	0.11	-42089	2406	-350	713
277	ST ROL A	121.300	2.54	3.34	1.036	0.60	-2046	5486	-252	104
278	ST ROL C	150.838	2.53	0.83	1.057	0.39	-13293	5539	-322	420
279	ST ROL D	188.197	2.53	-0.73	1.093	0.26	-30888	5660	-430	748
280	AC ROLL	92.377	3.24	12.06	1.020	1.30	11794	6900	-221	-831

A.24 Critical Wing Loads

CRITICAL LOAD SUMMARY

INPUT FOR WING

FULL DOWN AILERON DEFLECTION, DEG = 15.000
AIRFOIL PITCHING MOMENT COEFFICIENT WITH NO AILERON DEFLECTION = -0.030

ALTITUDES ARE 0 12000 18000

CRITICAL WING LOADS

CASE	ANGLE	CL	V KEAS	CONFIG	CG	ALT	COND	FAR
222	PHAA	1.561	115.884	CRUISE	CG2	18000	STALL +N	23.333(b)
25	PLAA	0.472	212.500	CRUISE	CG2	0	MAN D	23.333(b)
150	PMAA	0.808	170.000	CRUISE	CG2	12000	GUST +C	23.333 (c)OR(b)
173	NAA	-0.432	170.000	CRUISE	CG3	12000	GUST -C	23.333(b)
40	ACRL	1.288	117.881	CRUISE	CG2	0	AC ROLL	23.333(a)(2)
138	TORS	0.470	170.000	CRUISE	CG1	12000	ST ROL C	23.333(b)

25. Critical Horizontal Tail Loads

GENERAL INPUT FOR CALCULATION OF HORIZONTAL TAIL LOADS

SLOPE OF LIFT CURVE OF WING, CL/RAD 4.605
INCIDENCE OF HORIZ TAIL, WL TO CHORD, DEG 2.000
WING AREA, SQR FT 184.121
HORIZONTAL TAIL AREA, SQR FT 36.944
ASPECT RATIO OF WING 6.095

FULL TRAILING EDGE UP ELEVATOR DEFLECTION, DEG 30.000
FULL TRAILING EDGE DOWN ELEVATOR DEFLECTION, DEG 20.000

TOTAL ELEVATOR AREA (LH + RH), SQR FT 16.403
TOTAL ELEVATOR AREA FORWARD OF HINGE LINE (LH + RH), SQR FT 1.639
TOTAL ELEVATOR AREA AFT OF HINGE LINE (LH + RH), SQR FT 14.792

ASPECT RATIO OF HORIZ TAIL 4.017
SLOPE OF LIFT CURVE OF HORIZONTAL TAIL, CL/RAD 4.195

RATIO OF ELEVATOR AREA TO TAIL AREA 0.444
ANGLE WL TO ZERO LIFT LINE OF WING FOR CRUISE CONFIG 3.988
ANGLE WL TO ZERO LIFT LINE OF WING FOR ENROUTE CONFIG 0.000
ANGLE WL TO ZERO LIFT LINE OF WING FOR LANDING CONFIG 13.564
AIRPLANE MANEUVER LOAD FACTOR 3.800

EFFECT OF ELEV DEFLECTION IS 61.4 PERCENT OF SLOPE OF LIFT CURVE OF HORIZ TAIL

LENGTH OF AIRPLANE, FT 26.522
MAC OF WING, FT 5.771
FUS STA OF 25 PERCENT MAC OF HORIZ TAIL 261.027
FUS STA OF 50 PERCENT MAC OF HORIZ TAIL 270.357

A.25 CRITICAL HORIZONTAL TAIL LOADS

```
UP BALANCING TAIL LOAD FLAPS RETRACTED, FAR 23.421
     ANGLE WL TO WING ZERO LIFT LINE, DEG    3.988
     REFERENCE CASE NUMBER                   202
     BALANCED CONDITION                      STALL +N
     CG                                      CG1
     ALTITUDE                                18000
     V KEAS                                  113.453
     ANGLE OF ATTACK LOAD (CP 25 %), LBS     914.87
     TAIL ANGLE OF ATTACK, DEG               7.75
     CAMBER LOAD DUE ELEV (CP 50 %), LBS     -398.86
     ELEV DEFLECT, DEG (TE DN IS +)          -5.51
     ELEVATOR LOAD, LBS                      -42.444
     TOTAL BALANCED TAIL LOAD ,LBS           516.01
     CP TOTAL BAL LOAD, PERCENT TAIL MAC     5.67

DOWN BALANCING TAIL LOAD FLAPS RETRACTED,FAR 23.421
     ANGLE WL TO WING ZERO LIFT LINE, DEG    3.988
     REFERENCE CASE NUMBER                   45
     BALANCED CONDITION                      MAN D
     CG                                      CG3
     ALTITUDE                                0
     V KEAS                                  212.500
     ANGLE OF ATTACK LOAD (CP 25 %), LBS     234.32
     TAIL ANGLE OF ATTACK, DEG               0.56
     CAMBER LOAD DUE ELEV (CP 50 %), LBS     -844.83
     ELEV DEFLECT, DEG (TE DN IS +)          -3.33
     ELEVATOR LOAD, LBS                      -313.776
     TOTAL BALANCED TAIL LOAD ,LBS           -610.5
     CP TOTAL BAL LOAD, PERCENT TAIL MAC     59.59

UP BALANCING TAIL LOAD FLAPS EXTENDED, FAR 23.421
     ANGLE WL TO WING ZERO LIFT LINE, DEG     13.564
     REFERENCE CASE NUMBER                    81
     BALANCED CONDITION                       STAL 2/3G
     CG                                       CG5
     ALTITUDE                                 0
     V KEAS                                   52.86
     ANGLE OF ATTACK LOAD (CP 25 %), LBS      -69.29
     TAIL ANGLE OF ATTACK, DEG                -2.71
     CAMBER LOAD DUE ELEV (CP 50 %), LBS      -58.55
     ELEV DEFLECT, DEG (TE DN IS +)           -3.73
     ELEVATOR LOAD, LBS                       -32.983
     TOTAL BALANCED TAIL LOAD ,LBS            -127.83
     CP TOTAL BAL LOAD, PERCENT TAIL MAC      36.44

DOWN BALANCING TAIL LOAD FLAPS EXTENDED, FAR 23.421
     ANGLE WL TO WING ZERO LIFT LINE, DEG        13.564
     REFERENCE CASE NUMBER                          106
     BALANCED CONDITION                         GUST VF
     CG                                             CG7
     ALTITUDE                                         0
     V KEAS                                       105.5
     ANGLE OF ATTACK LOAD (CP 25 %), LBS         -535.3
     TAIL ANGLE OF ATTACK, DEG                    -5.25
     CAMBER LOAD DUE ELEV (CP 50 %), LBS        -491.98
     ELEV DEFLECT, DEG (TE DN IS +)               -7.86
     ELEVATOR LOAD, LBS                        -271.009
     TOTAL BALANCED TAIL LOAD ,LBS             -1027.28
     CP TOTAL BAL LOAD, PERCENT TAIL MAC          36.97

UNCHECKED MANEUVER DOWN TAIL LOAD (ELEV TE UP), FAR 23.423(A)(1)
     ANGLE WL TO WING ZERO LIFT LINE, DEG         3.988
     REFERENCE CASE NUMBER                          274
     BALANCED CONDITION                           BAL A
     CG                                             CG4
     ALTITUDE                                         0
     V KEAS                                     121.300
```

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     BALANCED TAIL LOAD, LBS                    -163.409
     LOAD DUE ANGLE OF ATTACK(CP 25%),LBS        -51.490
     BAL LOAD DUE TO ELEV, LBS                  -111.919
     LD INCREM'T EL UP TE                      30 (CP 50%),LBS -1346.496
     TOTAL TAIL LOAD, LBS                      -1397.987
     UNBALANCED MOM @ CG, IN-LBS 243541.407
     ELEV DEFLECT TE UP, DEG                      30.000
     ELEVATOR LOAD, LBS                         -555.954

UNCHECKED MANEUVER UP TAIL LOAD (ELEV TE DN), FAR 23.423(A)(2)
     ANGLE WL TO WING ZERO LIFT LINE, DEG          3.988
     REFERENCE CASE NUMBER                           154
     BALANCED CONDITION                            BAL A
     CG                                              CG2
     ALTITUDE                                          0
     V KEAS                                        121.3
     BALANCED TAIL LOAD, LBS                    -113.160
     BAL LD DUE ANG OF ATTACK(CP 25%),LB          65.884
     BAL LOAD DUE TO ELEV (CP 50%), LBS         -179.044
     LD INCREM'T EL DN TE                      20 (CP 50%),LBS  1161.450
     TOTAL TAIL LOAD, LBS                       1227.334
     UNBALANCED MOM @ CG, IN-LBS             -258537.078
     ELEV DEFLECT TE DN, DEG                      20.000
     ELEVATOR LOAD, LBS                          482.372

DOWN LOAD CHECKED MANEUVER TAIL LOAD, FAR 23.423(B)
     ANGLE WL TO WING ZERO LIFT LINE, DEG          3.988
     REFERENCE CASE NUMBER                            56
     BALANCED CONDITION                            BAL D
     CG                                              CG3
     PITCH MOMENT OF INERTIA, SLUG FT^2         2242.777
     ALTITUDE                                          0
     V KEAS                                      212.500
     BALANCED TAIL LOAD, LBS                    -446.773
     ANGLE OF ATTACK LOAD(CP AT 25%),LBS        -461.785
     BAL LOAD DUE ELEV (CP 50%), LBS              15.013
     TAIL ANGLE OF ATTACK, DEG                    -1.115
     MANEUVER LOAD INCREMENT, LBS               -218.344
     TOTAL CHECKED MAN TAIL LOAD, LBS           -665.116

UP LOAD CHECKED MANEUVER TAIL LOAD, FAR23.423(B)
     ANGLE WL TO WING ZERO LIFT LINE, DEG          3.988
     REFERENCE CASE NUMBER                           204
     BALANCED CONDITION                            MAN C
     CG                                              CG1
     PITCH MOMENT OF INERTIA, SLUG FT^2         2723.372
     ALTITUDE                                          0
     V KEAS                                      150.838
     BALANCED TAIL LOAD, LBS                     388.725
     ANGLE OF ATTACK LOAD(CP AT 25%),LBS         708.619
     BAL LOAD DUE ELEV (CP 50%), LBS            -319.895
     TAIL ANGLE OF ATTACK, DEG                     3.397
     MANEUVER LOAD INCREMENT, LBS                398.638
     TOTAL CHECKED MAN TAIL LOAD, LBS            787.362

UP GUST TAIL LOAD FLAPS RETRACTED, FAR 23.425(A)(1)
     REFERENCE CASE NUMBERS  15 AND  10
     FOR CONDITIONS BAL C AND GUST +C
     V KEAS                                      170.000
     CG                                              CG1
     ALTITUDE                                          0
     BALANCED TAIL LOAD, LBS                    -104.578
     BAL ANG OF ATTAC LD(CP 25%),LBS            -180.432
     BAL LOAD DUE ELEV(CP 50%),LBS                75.854
     GUST INCREMENT TAIL LD(CP 25%),LBS          946.407
     TOTAL UP GUST LOAD, LBS                     841.829
     TOT GUST ANG OF ATTAC LD(CP 25%),LBS        765.975
     TOT GUST LOAD DUE ELEV(CP 50%),LBS           75.854

DOWN GUST TAIL LOAD FLAPS RETRACTED, FAR 23.425(A)(1)
     REFERENCE CASE NUMBERS  55 AND  53
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     FOR CONDITIONS     BAL C  AND    GUST -C
     V KEAS                                    170.000
     CG                                            CG3
     ALTITUDE                                        0
     BALANCED TAIL LOAD, LBS                  -308.340
     GUST INCREMENT TAIL LD(CP 25%),LBS       -902.690
     TOTAL DOWN GUST LOAD, LBS               -1211.031
     TOT GUST ANG OF ATTAC LD(CP 25%),LBS    -1117.598
     TOT GUST LOAD DUE ELEV(CP 50%),LBS        -93.432

UP GUST TAIL LOAD FLAPS EXTENDED, FAR 23.425(A)(2)
     REFERENCE CASE NUMBERS  88 AND  86
     FOR CONDITIONS BAL VF AND GUST VF
     V KEAS                                    105.500
     CG                                            CG5
     ALTITUDE                                        0
     BALANCED TAIL LOAD, LBS                  -765.781
     GUST INCREMENT TAIL LD(CP 25%),LBS        293.665
     TOTAL UP GUST LOAD, LBS                  -472.116
     TOT GUST ANG OF ATTAC LD(CP 25%),LBS     -479.994
     TOT GUST LOAD DUE ELEV(CP 50%),LBS          7.878

DOWN GUST TAIL LOAD FLAPS EXTENDED, FAR 23.425(A)(2)
     REFERENCE CASE NUMBERS  108 AND  107
     FOR CONDITIONS BAL VF AND GUST -VF
     V KEAS                                    105.500
     CG                                            CG7
     ALTITUDE                                        0
     BALANCED TAIL LOAD, LBS                  -966.241
     GUST INCREMENT TAIL LD(CP 25%),LBS       -280.099
     TOTAL DN GUST LOAD, LBS                 -1246.340
     TOT GUST ANG OF ATTAC LD(CP 25%),LBS    -1090.368
     TOT GUST LOAD DUE ELEV(CP 50%),LBS       -155.972

UNSYMMETRICAL LOAD, FAR 23.427(a)
     REFERENCE CASE NUMBER                         107
     FOR CONDITION                            GUST -VF
     V KEAS                                    105.500
     CG                                            CG7
     ALTITUDE                                        0
     RH SIDE TAIL LOAD                           0.000
     LH SIDE TAIL LOAD                           0.000
     TOTAL TAIL LOAD                             0.000
     DISTRIBUTION SAME AS                     107 CASE
```

A.26 Critical Vertical Tail Loads

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INPUT FOR VERTICAL TAIL
----- --- -------- ----
FULL RUDDER DEFLECTION                     30.000
GROSS WEIGHT, LBS                        3400.000
WING AREA, SQR FT                         184.121
WING MAC, FT                                5.771
VERTICAL TAIL AREA, SQR FT                 14.843
RUDDER AREA, SQR FT                         5.236
AREA OF RUDDER FWD OF HINGE LINE, SQR FT         0.570
AREA OF RUDDER AFT OF HINGE LINE, SQR FT         4.630
ASPECT RATIO OF VERT TAIL                        1.520
SLOPE OF LIFT CURVE OF VERTICAL TAIL, CL/RAD     2.713
MAC OF VERTICAL TAIL, FT                         3.367
FS 25 PERCENT VMAC                             267.000
LENGTH OF AIRPLANE, FT                          26.522
WING SPAN, FT                                   33.500
WEIGHT OF WING, LBS                            306.000

CRITICAL VERTICAL TAIL LOADS
-------- -------- ---- -----

MANEUVER LOAD FOR SUDDEN FULL RUDDER DEFLECTION, FAR 23.441(a)(1)
     REFERENCE CASE NUMBER                        14
     TOTAL TAIL LOAD                             586
     DISTRIBUTION                         CAMBER, CP AT 50 PERCENT CHORD
     LOAD ON RUDDER                              165

MANEUVER LOAD FOR YAW TO SIDESLIP OF 19.5 DEG WITH RUDDER MAINTAINED ATFULL DEFLECTION, FAR 23.441(a)(2)
     REFERENCE CASE NUMBER                        14
     TOTAL TAIL LOAD                             -97
     LOAD DUE TO YAW 19.5 DEG                   -684
     DISTRIBUTION                         ANGLE OF ATTACK, CP AT 25 PERCENT CHORD
     LOAD DUE TO RUDDER DEFLECTION               586
     DISTRIBUTION                         CAMBER, CP AT 50 PERCENT CHORD
     LOAD ON RUDDER                              109

MANEUVER LOAD FOR A YAW OF 15 DEG WITH RUDDER IN NEUTRAL, FAR 23.441(A)(3)
     REFERENCE CASE NUMBER                        14
     TOTAL TAIL LOAD                            -526
     DISTRIBUTION                         ANGLE OF ATTACK, CP AT 25 PERCENT CHORD

SIDE GUST LOAD AT VC, FAR 23.443(b)
     REFERENCE CASE NUMBER                        35
     MOMENT OF INERTIA,IZZ,SLUG-FT SQR      4169.164
     TOTAL TAIL LOAD, LBS                        604
     DISTRIBUTION                         ANGLE OF ATTACK, CP AT 25 PERCENT CHORD
```

A.27 Critical Fuselage Loads

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CRITICAL FUSELAGE LOADS
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MAXIMUM TOTAL FUSELAGE LOAD ACTING DOWN ON WING FOR BALANCED AIRPLANE
     TOTAL FUSELAGE DOWN LOAD ACTING ON WING  13321.261
     NZ                                           4.233
     TAIL LOAD                                 -224.189
     CONDITION                                  GUST +C
     CASE NUMBER                                150.000

     THIS CONDITION MAY BE CRITICAL FOR DOWN SHEAR JUST AFT OF THE REAR WING
     ATTACHMENT OR JUST FORWARD OF THE FORWARD WING ATTACHMENT OR BOTH
     THIS IS PROBABLY THE MOST CRITICAL AFT FUSELAGE DOWN BENDING AND SHEAR
     CONDITION FOR AFT MOUNTED ENGINE CONFIGURATION

MAXIMUM AFT FUSELAGE DOWN BENDING FOR BALANCED CONDITIONS
     TOTAL FUSELAGE DOWN LOAD ACTING ON WING  12526.779
     CONDITION                                  GUST +C
     CASE NUMBER                                170.000
     TAIL LOAD                                 -597.752
     NZ                                           4.783

MAXIMUM AFT FUSELAGE UP BENDING FOR BALANCED CONDITIONS
     TOTAL FUSELAGE LOAD ACTING ON WING       -6364.046
     CONDITION                                  GUST -C
     CASE NUMBER                                133.000
     TAIL LOAD                                 -549.930
     NZ                                          -2.235

UNCHECKED PULL UP MANEUVER (DOWN TAIL LOAD)
     BALANCED DOWN TAIL LOAD
     UNBALANCED INCREMENT OF TAIL LOAD
     TOTAL TAIL LOAD
     XCG
     FS 50 PERCENT HORIZ TAIL
     UNBALANCED MOMENT ABOUT CG
     SEE HORIZONTAL TAIL LOADS FOR FURTHER DATA

CHECKED PULL UP MANEUVER (DOWN TAIL LOAD)
     BALANCED DOWN TAIL LOAD
     UNBALANCED INCREMENT OF TAIL LOAD
     TOTAL TAIL LOAD
     XCG
     FS 50 PERCENT HORIZ TAIL
     UNBALANCED MOMENT ABOUT CG
     SEE HORIZONTAL TAIL LOADS FOR FURTHER DATA

LANDING CONDITIONS
     FORWARD FUSELAGE IS CRITICAL FOR UP BENDING FOR 3 WHEEL LEVEL LANDING
     FORWARD FUSELAGE MAY BE CRITICAL FOR DOWN BENDING FOR 2 WHEEL LEVEL LANDING

GREATEST VERTICAL INERTIA FACTOR FOR CONCENTRATED WEIGHT INSTALLATIONS
     NZ                         5.790
     CASE NUMBER              190.000
     CONDITION                GUST +C
     BALANCING TAIL LOAD     -533.145
     PITCHING ACCELERATION WILL ADD ALGEBRAICLY TO VERTICAL INERTIA AT ALL FUS STATIONS
     PITCHING ACCELERATION WILL ADD ALGEBRAICALLY TO VERTICAL INERTIA AT ALL FUS STATIONS
```

A.28 Critical Aileron Loads

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AILERON LOADS

INPUT

VA = 121.300  VC = 170.000  VD = 212.500 KNOTS

MAX DOWN AILERON DEFLECTION    = 15.000 DEGREES
MAX UP AILERON DEFLECTION      = -10.000 DEGREES
AILERON AREA FWD OF HINGE LINE = 1.300  SQ FT
AILERON AREA AFT OF HINGE LINE = 5.188  SQ FT
AILERON AREA                   = 6.488  SQ FT

OUTPUT

AILERON DEFLECTIONS AT VA = 15.000 DEG AND -10.000 DEG
AILERON DEFLECTIONS AT VC = 10.703 DEG AND -7.135 DEG
AILERON DEFLECTIONS AT VC = 4.281 DEG AND -2.854 DEG

CRITICAL LOAD FOR DOWN AILERON IS 272.113 LBS AT 170.000 KNOTS
CRITICAL LOAD FOR UP AILERON IS -181.409 LBS AT 170.000 KNOTS

CHORDWISE DISTRIBUTION IS CONSTANT FROM LEADING EDGE TO HINGE LINE THEN
THEN TAPERS TO NOTHING AT TRAILING EDGE.

PRESSURE FWD OF HINGE LINE FOR DOWN AILERON IS 0.485 LBS/SQ IN
PRESSURE FWD OF HINGE LINE FOR UP AILERON IS -0.324 LBS/SQ IN

CHECK2 =    272.113
```

A.29 Critical Flap Loads

```
INPUT

STALLING SPEED AT MAX TAKE-OFF WEIGHT WITH FLAPS RETRACTED = 62.200 KNOTS
STALLING SPEED AT MAX TAKE-OFF WEIGHT FLAPS EXTENDED       = 58.600 KNOTS

DESIGN FLAP SPEED   = 105.480 KNOTS
MAX TAKE-OFF WEIGHT = 3400.000 POUNDS

GUST LOAD FACTOR WITH FLAPS EXTENDED = 1.900
FLAP AREA ON ONE SIDE OF AIRPLANE    = 10.700 SQ FT

TOTAL AREA OF WING   = 184.121 SQ FT
MAX FLAP  DEFLECTION = 40.000 DEG

RATIO OF FLAP CHORD TO WING CHORD = 0.258

MAX HP OF ONE ENGINE = 250.000
BUTT LINE OF ENGINE  = 68.000

FRONTAL AREA OF NACELLE  = 8.200 SQ FT
PROPELLER DIAMETER       = 85.000 INCHES

OUTPUT

       1G STALL   2G STALL   2G AT VF  1.9G GUST AT VF
CLF       1.715      1.715      1.574      1.563
CLW       1.586      1.586      0.979      0.930
LF      213.625    427.250    635.249    630.660

CRITICAL FLAP LOAD PER FAR 23.345(a)= 635.249 LBS

CHORDWISE DISTRIBUTION TAPERS FROM THE LEADING EDGE TO THE TRAILING EDGE.
THE PRESSURE AT THE TRAILING EDGE IS HALF THE PRESSURE AT THE LEADING EDGE.
THE PRESSURE AT THE LEADING EDGE IS = 0.550 PSI

OUTBOARD BL OF SLIPSTREAM   = 113.172
INBOARD BL OF SLIPSTREAM    = 22.828
SLIPSTREAM VELOCITY AT FLAP = 125.099 KNOTS
INCREASE FLAP LOAD IN SLIPSTREAM AT VF BY FACTOR = 1.407
PER FAR 23.345(C)(2) AND FAR 23.345(E)

INCREASE FLAP LOAD AT VF BY FACTOR OF 1.301
CRITICAL FLAP LOAD COMBINED WITH HORIZONTAL GUST = 826.324 LBS
PER FAR 23.345(B)(1) AND FAR 23.345(e) FOR HORIZONTAL 25 FPS GUST
```

A.30 Tab Loads

```
INPUT

HORIZONTAL TAIL TAB LOADS

VC (KEAS)                              170.000
TAB MAC                                  7.465
AREA OF TAB (SQR INCHES)               226.000
BUTT LINE OF TAB MAC                    17.822
CHORD OF AIRFOIL AT 17.822              42.166
MAX DEFLECTION OF TAB (DEG)             15.000

OUTPUT

E=CHORD OF TAB/CHORD OF AIRFOIL          0.177
TAB LOAD (LBS)                          84.650
TAB LE PSI                               0.499
TAB TE PSI                               0.250
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A.31 Airloads for Case 22 PHAA

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CRUISE CONFIGURATION

WING GEOMETRY CALCULATIONS
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 3 POINTS DEFINE THE LEADING EDGE
LEADING EDGE COORDINATES
 I       XLE       YLE
 1    45.000      0.000
 2    64.313     46.500
 3    72.000    201.000

 2 POINTS DEFINE THE TRAILING EDGE
TRAILING EDGE COORDINATES
 I       XTE       YTE
 1   146.000      0.000
 2   116.000    201.000

THE WING IS DIVIDED INTO  20 ELEMENTS OF DY
ELEMENT   YE     CE
 1      5.025     98.163
 2     15.075     92.489
 3     25.125     86.815
 4     35.175     81.141
 5     45.225     75.467
 6     55.275     73.000
 7     65.325     71.000
 8     75.375     69.000
 9     85.425     67.000
10     95.475     65.000
11    105.525     63.000
12    115.575     61.000
13    125.625     59.000
14    135.675     57.000
15    145.725     55.000
16    155.775     53.000
17    165.825     51.000
18    175.875     49.000
19    185.925     47.000
20    195.975     45.000

TOTAL AREA    = 26513.444
MAC           = 69.246
YBAR          = 87.854
XLE(MAC)      = 63.641
ASPECT RATIO  = 6.095
TOTAL SPAN    = 402.000
# OF ELEMENTS = 20.000
DY            = 10.050



ADDITIVE LIFT DISTRIBUTION
-------- ---- ------------

SELECTED WING STATIONS AND THEIR SLOPES OF LIFT CURVE

WING STA     SLOPE OF LIFT CURVE
     0.000          0.1075
   201.000          0.1075

ELEMENT     YE     CC(LA1)     C(LA1)  FOR CL=1.001
 1      5.025     91.056      0.928
 2     15.075     88.114      0.953
 3     25.125     85.065      0.980
 4     35.175     81.910      1.009
 5     45.225     78.644      1.042
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 6     55.275     76.869      1.053
 7     65.325     75.208      1.059
 8     75.375     73.424      1.064
 9     85.425     71.507      1.067
10     95.475     69.448      1.068
11    105.525     67.236      1.067
12    115.575     64.852      1.063
13    125.625     62.277      1.056
14    135.675     59.479      1.043
15    145.725     56.419      1.026
16    155.775     53.034      1.001
17    165.825     49.229      0.965
18    175.875     44.827      0.915
19    185.925     39.454      0.839
20    195.975     31.830      0.707


BASIC LIFT DISTRIBUTION
----- ---- ------------

SELECTED WING STATIONS AND THEIR ANGLES FROM WL TO SECTION ZERO LIFT LINE

 I  WING STA       ANGLE
 1     0.000      5.000
 2    46.500      4.577
 3   109.279      4.028
 4   201.000      1.900

AWO = 3.988

ELEM REF ANGLE      Ao         CClb        Clb
 1      4.954      0.966      5.098      0.052
 2      4.863      0.875      4.348      0.047
 3      4.771      0.783      3.655      0.042
 4      4.680      0.692      3.017      0.037
 5      4.589      0.600      2.436      0.032
 6      4.500      0.512      2.010      0.028
 7      4.412      0.424      1.619      0.023
 8      4.325      0.336      1.247      0.018
 9      4.237      0.248      0.895      0.013
10      4.149      0.161      0.561      0.009
11      4.061      0.073      0.246      0.004
12      3.882     -0.106     -0.348     -0.006
13      3.649     -0.339     -1.076     -0.018
14      3.416     -0.573     -1.754     -0.031
15      3.182     -0.806     -2.382     -0.043
16      2.949     -1.039     -2.959     -0.056
17      2.716     -1.272     -3.487     -0.068
18      2.483     -1.505     -3.964     -0.081
19      2.250     -1.738     -4.392     -0.093
20      2.017     -1.972     -4.769     -0.106

THERE IS NO DISCONTINUITY BETWEEN FLAP AND AILERON

ELEM    CC(lb)   C(lb) FAIRED
 1      5.098      0.052
 2      4.348      0.047
 3      3.655      0.042
 4      3.017      0.037
 5      2.436      0.032
 6      2.010      0.028
 7      1.619      0.023
 8      1.247      0.018
 9      0.895      0.013
10      0.561      0.009
11      0.246      0.004
12     -0.348     -0.006
13     -1.076     -0.018
14     -1.754     -0.031
15     -2.382     -0.043
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16     -2.959     -0.056
17     -3.487     -0.068
18     -3.964     -0.081
19     -4.392     -0.093
20     -4.769     -0.106
BCHECK = 0.000
WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.520
 J        CL        CDI        CPD         CD         CM
 1      1.462      0.160      0.010      0.170     -0.030
 2      1.495      0.153      0.010      0.163     -0.030
 3      1.531      0.146      0.010      0.156     -0.030
 4      1.572      0.137      0.010      0.147     -0.030
 5      1.616      0.126      0.010      0.136     -0.030
 6      1.628      0.122      0.010      0.132     -0.030
 7      1.633      0.118      0.010      0.128     -0.030
 8      1.636      0.115      0.010      0.125     -0.030
 9      1.636      0.113      0.010      0.123     -0.030
10      1.633      0.111      0.010      0.121     -0.030
11      1.626      0.110      0.010      0.120     -0.030
12      1.610      0.108      0.010      0.118     -0.030
13      1.586      0.106      0.010      0.116     -0.030
14      1.555      0.105      0.010      0.115     -0.030
15      1.516      0.106      0.010      0.116     -0.030
16      1.465      0.109      0.010      0.119     -0.030
17      1.399      0.113      0.010      0.123     -0.030
18      1.310      0.119      0.010      0.129     -0.030
19      1.183      0.127      0.010      0.137     -0.030
20      0.969      0.134      0.010      0.144     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 18.915
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 14.927

CL(WING)=     1.5209
CD(WING)=     0.1344
CM(WING)=    -0.0300


AIRLOADS FOR  22  PHAA
CL =  1.52  V KTS (EAS) =  117.4
        J          Y          X          L          D          M
        1      5.025     71.628    466.177    -67.858   -942.615
        2     15.075     74.383    448.385    -68.501   -836.792
        3     25.125     77.139    430.259    -69.070   -737.269
        4     35.175     79.895    411.787    -69.572   -644.045
        5     45.225     82.650    392.954    -70.015   -557.119
        6     55.275     83.000    382.538    -69.532   -521.302
        7     65.325     83.000    372.930    -68.680   -493.129
        8     75.375     83.000    362.822    -67.553   -465.738
        9     85.425     83.000    352.179    -66.132   -439.130
       10     95.475     83.000    340.958    -64.392   -413.304
       11    105.525     83.000    329.102    -62.304   -388.261
       12    115.575     83.000    315.513    -59.892   -364.001
       13    125.625     83.000    300.595    -57.078   -340.523
       14    135.675     83.000    284.839    -53.775   -317.827
       15    145.725     83.000    268.051    -49.916   -295.915
       16    155.775     83.000    249.943    -45.414   -274.785
       17    165.825     83.000    230.045    -40.144   -254.437
```

18	175.875	83.000	207.519	-33.908	-234.873
19	185.925	83.000	180.539	-26.322	-216.090
20	195.975	83.000	142.858	-16.199	-198.091

WL WING REF PLANE AT PLANE OF SYMMETRY OF AIRPLANE IS 78.5
SLOPE OF WING REF PLANE (DIHEDRAL) IS 6

X	Y	Z	FX	FZ	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	-16.199	142.858	-16.199	142.858	0.000	-198.091	0.000
83.000	185.925	98.040	-26.322	180.539	-42.520	323.397	1435.721	-431.294	162.798
83.000	175.875	96.984	-33.908	207.519	-76.429	530.916	4685.859	-711.084	590.127
83.000	165.825	95.928	-40.144	230.045	-116.573	760.961	10021.569	-1046.257	1358.234
83.000	155.775	94.871	-45.414	249.943	-161.987	1010.904	17669.231	-1444.185	2529.793
83.000	145.725	93.815	-49.916	268.051	-211.904	1278.956	27828.819	-1911.214	4157.765
83.000	135.675	92.759	-53.775	284.839	-265.679	1563.795	40682.323	-2452.885	6287.397
83.000	125.625	91.703	-57.078	300.595	-322.757	1864.390	56398.459	-3074.055	8957.469
83.000	115.575	90.647	-59.892	315.513	-382.649	2179.903	75135.577	-3778.996	12201.180
83.000	105.525	89.590	-62.304	329.102	-444.953	2509.005	97043.605	-4571.464	16046.803
83.000	95.475	88.534	-64.392	340.958	-509.345	2849.963	122259.108	-5454.787	20518.579
83.000	85.425	87.478	-66.132	352.179	-575.477	3202.143	150901.240	-6431.955	25637.494
83.000	75.375	86.422	-67.553	362.822	-643.030	3564.965	183082.774	-7505.589	31421.036
83.000	65.325	85.365	-68.680	372.930	-711.710	3937.895	218910.670	-8677.972	37883.489
83.000	55.275	84.309	-69.532	382.538	-781.242	4320.433	258486.516	-9951.076	45036.172
82.650	45.225	83.253	-70.015	392.954	-851.257	4713.387	301906.871	-12843.808	52887.655
79.895	35.175	82.197	-69.572	411.787	-920.829	5125.174	349276.412	-27375.066	61442.785
77.139	25.125	81.141	-69.070	430.259	-989.899	5555.432	400784.408	-43207.741	70697.115
74.383	15.075	80.084	-68.501	448.385	-1058.400	6003.818	456616.504	-60398.503	80645.600
71.628	5.025	79.028	-67.858	466.177	-1126.258	6469.995	516954.874	-79003.001	91282.518

A.32 Airloads for Case 142 Pt. A (Ref. Case 160 ACRL)

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CRUISE CONFIGURATION

WING GEOMETRY CALCULATIONS
---- -------- ------------

 3 POINTS DEFINE THE LEADING EDGE
LEADING EDGE COORDINATES
 I       XLE       YLE
 1    45.000      0.000
 2    64.313     46.500
 3    72.000    201.000

 2 POINTS DEFINE THE TRAILING EDGE
TRAILING EDGE COORDINATES
 I       XTE       YTE
 1   146.000      0.000
 2   116.000    201.000

THE WING IS DIVIDED INTO  20 ELEMENTS OF DY
ELEMENT   YE      CE
 1      5.025     98.163
 2     15.075     92.489
 3     25.125     86.815
 4     35.175     81.141
 5     45.225     75.467
 6     55.275     73.000
 7     65.325     71.000
 8     75.375     69.000
 9     85.425     67.000
10     95.475     65.000
11    105.525     63.000
12    115.575     61.000
13    125.625     59.000
14    135.675     57.000
15    145.725     55.000
16    155.775     53.000
17    165.825     51.000
18    175.875     49.000
19    185.925     47.000
20    195.975     45.000

TOTAL AREA    = 26513.444
MAC           = 69.246
YBAR          = 87.854
XLE(MAC)      = 63.641
ASPECT RATIO  = 6.095
TOTAL SPAN    = 402.000
# OF ELEMENTS = 20.000
DY            = 10.050



ADDITIVE LIFT DISTRIBUTION
-------- ---- ------------

SELECTED WING STATIONS AND THEIR SLOPES OF LIFT CURVE

WING STA     SLOPE OF LIFT CURVE
     0.000          0.1075
   201.000          0.1075

ELEMENT     YE     CC(LA1)     C(LA1)  FOR CL=1.001
 1      5.025     91.056      0.928
 2     15.075     88.114      0.953
 3     25.125     85.065      0.980
 4     35.175     81.910      1.009
 5     45.225     78.644      1.042
```

6	55.275	76.869	1.053
7	65.325	75.208	1.059
8	75.375	73.424	1.064
9	85.425	71.507	1.067
10	95.475	69.448	1.068
11	105.525	67.236	1.067
12	115.575	64.852	1.063
13	125.625	62.277	1.056
14	135.675	59.479	1.043
15	145.725	56.419	1.026
16	155.775	53.034	1.001
17	165.825	49.229	0.965
18	175.875	44.827	0.915
19	185.925	39.454	0.839
20	195.975	31.830	0.707

BASIC LIFT DISTRIBUTION
----- ---- ------------

SELECTED WING STATIONS AND THEIR ANGLES FROM WL TO SECTION ZERO LIFT LINE

I	WING STA	ANGLE
1	0.000	5.000
2	46.500	4.577
3	109.279	4.028
4	201.000	1.900

AWO = 3.988

ELEM	REF ANGLE	Ao	CClb	Clb
1	4.954	0.966	5.098	0.052
2	4.863	0.875	4.348	0.047
3	4.771	0.783	3.655	0.042
4	4.680	0.692	3.017	0.037
5	4.589	0.600	2.436	0.032
6	4.500	0.512	2.010	0.028
7	4.412	0.424	1.619	0.023
8	4.325	0.336	1.247	0.018
9	4.237	0.248	0.895	0.013
10	4.149	0.161	0.561	0.009
11	4.061	0.073	0.246	0.004
12	3.882	-0.106	-0.348	-0.006
13	3.649	-0.339	-1.076	-0.018
14	3.416	-0.573	-1.754	-0.031
15	3.182	-0.806	-2.382	-0.043
16	2.949	-1.039	-2.959	-0.056
17	2.716	-1.272	-3.487	-0.068
18	2.483	-1.505	-3.964	-0.081
19	2.250	-1.738	-4.392	-0.093
20	2.017	-1.972	-4.769	-0.106

THERE IS NO DISCONTINUITY BETWEEN FLAP AND AILERON

ELEM	CC(lb)	C(lb) FAIRED
1	5.098	0.052
2	4.348	0.047
3	3.655	0.042
4	3.017	0.037
5	2.436	0.032
6	2.010	0.028
7	1.619	0.023
8	1.247	0.018
9	0.895	0.013
10	0.561	0.009
11	0.246	0.004
12	-0.348	-0.006
13	-1.076	-0.018
14	-1.754	-0.031
15	-2.382	-0.043

```
16     -2.959     -0.056
17     -3.487     -0.068
18     -3.964     -0.081
19     -4.392     -0.093
20     -4.769     -0.106
BCHECK = 0.000
WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 1.550
 J        CL        CDI        CPD         CD         CM
 1      1.490      0.166      0.010      0.176     -0.030
 2      1.524      0.159      0.010      0.169     -0.030
 3      1.561      0.151      0.010      0.161     -0.030
 4      1.602      0.142      0.010      0.152     -0.030
 5      1.648      0.131      0.010      0.141     -0.030
 6      1.660      0.126      0.010      0.136     -0.030
 7      1.665      0.123      0.010      0.133     -0.030
 8      1.667      0.120      0.010      0.130     -0.030
 9      1.668      0.117      0.010      0.127     -0.030
10      1.665      0.115      0.010      0.125     -0.030
11      1.658      0.114      0.010      0.124     -0.030
12      1.642      0.112      0.010      0.122     -0.030
13      1.618      0.110      0.010      0.120     -0.030
14      1.587      0.110      0.010      0.120     -0.030
15      1.547      0.111      0.010      0.121     -0.030
16      1.495      0.113      0.010      0.123     -0.030
17      1.428      0.118      0.010      0.128     -0.030
18      1.337      0.125      0.010      0.135     -0.030
19      1.208      0.133      0.010      0.143     -0.030
20      0.990      0.140      0.010      0.150     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 19.289
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 15.301

CL(WING)=     1.5509
CD(WING)=     0.1394
CM(WING)=    -0.0300


AIRLOADS FOR  142  PT. A (REF. CASE 160 ACRL)
CL =  1.55  V KTS (EAS) =  116
         J          Y          X          L          D          M
         1      5.025     71.628    463.568    -69.691   -920.267
         2     15.075     74.383    445.877    -70.303   -816.954
         3     25.125     77.139    427.849    -70.841   -719.790
         4     35.175     79.895    409.473    -71.310   -628.776
         5     45.225     82.650    390.734    -71.718   -543.911
         6     55.275     83.000    380.386    -71.208   -508.943
         7     65.325     83.000    370.844    -70.323   -481.438
         8     75.375     83.000    360.806    -69.161   -454.696
         9     85.425     83.000    350.237    -67.699   -428.719
        10     95.475     83.000    339.094    -65.913   -403.505
        11    105.525     83.000    327.320    -63.773   -379.056
        12    115.575     83.000    313.842    -61.301   -355.371
        13    125.625     83.000    299.051    -58.420   -332.450
        14    135.675     83.000    283.427    -55.041   -310.292
        15    145.725     83.000    266.779    -51.098   -288.899
        16    155.775     83.000    248.817    -46.501   -268.270
        17    165.825     83.000    229.077    -41.124   -248.405
```

18	175.875	83.000	206.725	-34.762	-229.304
19	185.925	83.000	179.945	-27.025	-210.967
20	195.975	83.000	142.523	-16.699	-193.394

WL WING REF PLANE AT PLANE OF SYMMETRY OF AIRPLANE IS 78.5
SLOPE OF WING REF PLANE (DIHEDRAL) IS 6

X	Y	Z	FX	FZ	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	-16.699	142.523	-16.699	142.523	0.000	-193.394	0.000
83.000	185.925	98.040	-27.025	179.945	-43.724	322.468	1432.356	-422.003	167.828
83.000	175.875	96.984	-34.762	206.725	-78.486	529.194	4673.160	-697.496	607.255
83.000	165.825	95.928	-41.124	229.077	-119.610	758.271	9991.555	-1028.811	1396.042
83.000	155.775	94.871	-46.501	248.817	-166.111	1007.088	17612.175	-1423.431	2598.121
83.000	145.725	93.815	-51.098	266.779	-217.209	1273.866	27733.406	-1887.801	4267.538
83.000	135.675	92.759	-55.041	283.427	-272.251	1557.294	40535.763	-2427.540	6450.493
83.000	125.625	91.703	-58.420	299.051	-330.671	1856.345	56186.566	-3047.579	9186.613
83.000	115.575	90.647	-61.301	313.842	-391.972	2170.187	74842.833	-3752.249	12509.855
83.000	105.525	89.590	-63.773	327.320	-455.745	2497.507	96653.208	-4545.358	16449.172
83.000	95.475	88.534	-65.913	339.094	-521.658	2836.600	121753.150	-5430.281	21029.409
83.000	85.425	87.478	-67.699	350.237	-589.357	3186.837	150260.982	-6410.044	26272.070
83.000	75.375	86.422	-69.161	360.806	-658.518	3547.643	182288.695	-7487.296	32195.107
83.000	65.325	85.365	-70.323	370.844	-728.841	3918.487	217942.508	-8664.346	38813.211
83.000	55.275	84.309	-71.208	380.386	-800.049	4298.873	257323.307	-9943.186	46138.065
82.650	45.225	83.253	-71.718	390.734	-871.767	4689.607	300526.984	-12835.035	54178.555
79.895	35.175	82.197	-71.310	409.473	-943.077	5099.080	347657.539	-27307.161	62939.814
77.139	25.125	81.141	-70.841	427.849	-1013.918	5526.929	398903.294	-43073.953	72417.736
74.383	15.075	80.084	-70.303	445.877	-1084.221	5972.806	454448.926	-60191.701	82607.607
71.628	5.025	79.028	-69.691	463.568	-1153.912	6436.374	514475.622	-78715.666	93504.025

A.33 Airloads for Case 138 TORS

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CRUISE CONFIGURATION

WING GEOMETRY CALCULATIONS
---- -------- ------------

 3 POINTS DEFINE THE LEADING EDGE
LEADING EDGE COORDINATES
 I       XLE       YLE
 1    45.000      0.000
 2    64.313     46.500
 3    72.000    201.000

 2 POINTS DEFINE THE TRAILING EDGE
TRAILING EDGE COORDINATES
 I       XTE       YTE
 1   146.000      0.000
 2   116.000    201.000

THE WING IS DIVIDED INTO  20 ELEMENTS OF DY
ELEMENT   YE      CE
 1      5.025     98.163
 2     15.075     92.489
 3     25.125     86.815
 4     35.175     81.141
 5     45.225     75.467
 6     55.275     73.000
 7     65.325     71.000
 8     75.375     69.000
 9     85.425     67.000
10     95.475     65.000
11    105.525     63.000
12    115.575     61.000
13    125.625     59.000
14    135.675     57.000
15    145.725     55.000
16    155.775     53.000
17    165.825     51.000
18    175.875     49.000
19    185.925     47.000
20    195.975     45.000

TOTAL AREA    = 26513.444
MAC           = 69.246
YBAR          = 87.854
XLE(MAC)      = 63.641
ASPECT RATIO  = 6.095
TOTAL SPAN    = 402.000
# OF ELEMENTS = 20.000
DY            = 10.050



ADDITIVE LIFT DISTRIBUTION
-------- ---- ------------

SELECTED WING STATIONS AND THEIR SLOPES OF LIFT CURVE

WING STA     SLOPE OF LIFT CURVE
     0.000          0.1075
   201.000          0.1075

ELEMENT      YE      CC(LA1)      C(LA1)  FOR CL=1.001
 1      5.025     91.056      0.928
 2     15.075     88.114      0.953
 3     25.125     85.065      0.980
 4     35.175     81.910      1.009
 5     45.225     78.644      1.042
```

6	55.275	76.869	1.053
7	65.325	75.208	1.059
8	75.375	73.424	1.064
9	85.425	71.507	1.067
10	95.475	69.448	1.068
11	105.525	67.236	1.067
12	115.575	64.852	1.063
13	125.625	62.277	1.056
14	135.675	59.479	1.043
15	145.725	56.419	1.026
16	155.775	53.034	1.001
17	165.825	49.229	0.965
18	175.875	44.827	0.915
19	185.925	39.454	0.839
20	195.975	31.830	0.707

BASIC LIFT DISTRIBUTION
----- ---- ------------

SELECTED WING STATIONS AND THEIR ANGLES FROM WL TO SECTION ZERO LIFT LINE

I	WING STA	ANGLE
1	0.000	5.000
2	46.500	4.577
3	109.279	4.028
4	201.000	1.900

AWO = 3.988

ELEM	REF ANGLE	Ao	CClb	Clb
1	4.954	0.966	5.098	0.052
2	4.863	0.875	4.348	0.047
3	4.771	0.783	3.655	0.042
4	4.680	0.692	3.017	0.037
5	4.589	0.600	2.436	0.032
6	4.500	0.512	2.010	0.028
7	4.412	0.424	1.619	0.023
8	4.325	0.336	1.247	0.018
9	4.237	0.248	0.895	0.013
10	4.149	0.161	0.561	0.009
11	4.061	0.073	0.246	0.004
12	3.882	-0.106	-0.348	-0.006
13	3.649	-0.339	-1.076	-0.018
14	3.416	-0.573	-1.754	-0.031
15	3.182	-0.806	-2.382	-0.043
16	2.949	-1.039	-2.959	-0.056
17	2.716	-1.272	-3.487	-0.068
18	2.483	-1.505	-3.964	-0.081
19	2.250	-1.738	-4.392	-0.093
20	2.017	-1.972	-4.769	-0.106

THERE IS NO DISCONTINUITY BETWEEN FLAP AND AILERON

ELEM	CC(lb)	C(lb) FAIRED
1	5.098	0.052
2	4.348	0.047
3	3.655	0.042
4	3.017	0.037
5	2.436	0.032
6	2.010	0.028
7	1.619	0.023
8	1.247	0.018
9	0.895	0.013
10	0.561	0.009
11	0.246	0.004
12	-0.348	-0.006
13	-1.076	-0.018
14	-1.754	-0.031
15	-2.382	-0.043

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16     -2.959     -0.056
17     -3.487     -0.068
18     -3.964     -0.081
19     -4.392     -0.093
20     -4.769     -0.106
BCHECK = 0.000
WING AERO COEFFICIENT DISTRIBUTIONS
---- ---- ----------- -------------

TAU = 0.050

SELECTED WING STATIONS AND THEIR PROFILE DRAG COEFFICIENTS
 1     0.000      0.010
 2   201.000      0.010

SELECTED WING STATIONS AND THEIR MOMENT COEFFICIENTS
 1     0.000     -0.030
 2   201.000     -0.030

SPANWISE AERO COFFICIENT DISTRIBUTIONS FOR CL = 0.470
 J        CL        CDI        CPD        CD         CM
 1      0.488      0.019      0.010      0.029     -0.030
 2      0.495      0.018      0.010      0.028     -0.030
 3      0.503      0.017      0.010      0.027     -0.030
 4      0.512      0.016      0.010      0.026     -0.030
 5      0.522      0.015      0.010      0.025     -0.030
 6      0.522      0.014      0.010      0.024     -0.030
 7      0.521      0.013      0.010      0.023     -0.030
 8      0.518      0.012      0.010      0.022     -0.030
 9      0.515      0.012      0.010      0.022     -0.030
10      0.511      0.011      0.010      0.021     -0.030
11      0.506      0.011      0.010      0.021     -0.030
12      0.494      0.010      0.010      0.020     -0.030
13      0.478      0.009      0.010      0.019     -0.030
14      0.460      0.008      0.010      0.018     -0.030
15      0.439      0.007      0.010      0.017     -0.030
16      0.414      0.007      0.010      0.017     -0.030
17      0.385      0.007      0.010      0.017     -0.030
18      0.349      0.007      0.010      0.017     -0.030
19      0.301      0.007      0.010      0.017     -0.030
20      0.226      0.007      0.010      0.017     -0.030

AWO   = ANGLE FROM WL TO WING ZERO LIFT LINE = 3.988
ALPHA = ANGLE FROM RELATIVE WIND TO WING ZERO LIFT LINE = 5.849
ANRW2WL = ANGLE FROM RELATIVE WIND TO WATERLINE = 1.861

CL(WING)=     0.4703
CD(WING)=     0.0222
CM(WING)=    -0.0300


AIRLOADS FOR  138  TORS
CL =  .47  V KTS (EAS) =  170
         J          Y          X          L          D          M
         1      5.025     71.628    327.928      9.079  -1976.496
         2     15.075     74.383    313.300      7.737  -1754.605
         3     25.125     77.139    298.714      6.428  -1545.922
         4     35.175     79.895    284.164      5.148  -1350.447
         5     45.225     82.650    269.644      3.896  -1168.179
         6     55.275     83.000    261.003      3.350  -1093.078
         7     65.325     83.000    252.980      2.949  -1034.003
         8     75.375     83.000    244.688      2.597   -976.570
         9     85.425     83.000    236.103      2.297   -920.777
        10     95.475     83.000    227.194      2.052   -866.625
        11    105.525     83.000    217.919      1.866   -814.114
        12    115.575     83.000    206.181      1.603   -763.244
        13    125.625     83.000    192.913      1.352   -714.016
        14    135.675     83.000    179.276      1.205   -666.428
        15    145.725     83.000    165.140      1.167   -620.481
        16    155.775     83.000    150.311      1.246   -576.175
        17    165.825     83.000    134.472      1.449   -533.510
```

18	175.875	83.000	117.065	1.788	-492.486
19	185.925	83.000	96.884	2.280	-453.103
20	195.975	83.000	69.813	2.965	-415.361

WL WING REF PLANE AT PLANE OF SYMMETRY OF AIRPLANE IS 78.5
SLOPE OF WING REF PLANE (DIHEDRAL) IS 6

X	Y	Z	FX	FZ	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	2.965	69.813	2.965	69.813	0.000	-415.361	0.000
83.000	185.925	98.040	2.280	96.884	5.245	166.697	701.624	-865.334	-29.796
83.000	175.875	96.984	1.788	117.065	7.034	283.763	2376.933	-1352.284	-82.511
83.000	165.825	95.928	1.449	134.472	8.483	418.235	5228.749	-1878.371	-153.198
83.000	155.775	94.871	1.246	150.311	9.729	568.546	9432.008	-2445.595	-238.449
83.000	145.725	93.815	1.167	165.140	10.896	733.686	15145.895	-3055.812	-336.224
83.000	135.675	92.759	1.205	179.276	12.101	912.962	22519.442	-3710.746	-445.730
83.000	125.625	91.703	1.352	192.913	13.453	1105.875	31694.710	-4411.999	-567.343
83.000	115.575	90.647	1.603	206.181	15.056	1312.056	42808.751	-5161.058	-702.542
83.000	105.525	89.590	1.866	217.919	16.922	1529.975	55994.912	-5959.297	-853.855
83.000	95.475	88.534	2.052	227.194	18.974	1757.168	71371.159	-6808.080	-1023.922
83.000	85.425	87.478	2.297	236.103	21.271	1993.272	89030.701	-7708.852	-1214.613
83.000	75.375	86.422	2.597	244.688	23.868	2237.960	109063.082	-8662.995	-1428.390
83.000	65.325	85.365	2.949	252.980	26.818	2490.940	131554.581	-9671.834	-1668.268
83.000	55.275	84.309	3.350	261.003	30.168	2751.943	156588.530	-10736.638	-1937.785
82.650	45.225	83.253	3.896	269.644	34.064	3021.587	184245.562	-12835.050	-2240.973
79.895	35.175	82.197	5.148	284.164	39.212	3305.752	214612.512	-22475.733	-2583.312
77.139	25.125	81.141	6.428	298.714	45.640	3604.465	247835.315	-33089.490	-2977.390
74.383	15.075	80.084	7.737	313.300	53.377	3917.765	284060.192	-44728.271	-3436.068
71.628	5.025	79.028	9.079	327.928	62.455	4245.693	323433.734	-57444.092	-3972.502

A.34 Wing Inertia Loads

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WING INERTIA

INPUT DATA

WING PANEL WEIGHT ON ONE SIDE =  165 LBS
RATIO OF DENSITY PER SQ FT OF TIP AREA TO ROOT AREA = 0.950
DENSITY AT ROOT =  2.213 LB/SQ FT
DENSITY AT TIP  =  2.102 LB/SQ FT

 3 POINTS DEFINE LEADING EDGE OF WING
      ITEM          X          Y
         1     45.000      0.000
         2     64.313     46.500
         3     72.000    201.000

 2 POINTS DEFINE TRAILING EDGE OF WING
      ITEM          X          Y
         1    146.000      0.000
         2    116.000    201.000

BL OF INBOARD RIB IS  23
WING CL TO TIP IS DIVIDED INTO  20 ELEMENTS
WL REF PLANE AT PLANE OF SYMMETRY IS  78.5
SLOPE OF WRP IS  6 DEGREES
RATIO OF DENSITY OF TIP TO ROOT IS 0.950
WING PANEL (TIP TO TIP) INERTIA IXX =  4330144.253 LB-IN^2

WEIGHT OF WING PANEL IS    165.000

CONCENTRATED WEIGHT IS      0.000  LBS
AT COORDINATES X, Y, Z         0.000 ,       0.000 ,       0.000
CONCENTRATED WEIGHT IS      0.000  LBS
AT COORDINATES X, Y, Z         0.000 ,       0.000 ,       0.000
CONCENTRATED WEIGHT IS      0.000  LBS
AT COORDINATES X, Y, Z         0.000 ,       0.000 ,       0.000
CONCENTRATED WEIGHT IS      0.000  LBS
AT COORDINATES X, Y, Z         0.000 ,       0.000 ,       0.000

 6 CASES ARE CALCULATED
SR NO    CASE NO         NZ          NX       UNBAL MOM
    1         22     -3.800      0.607           0.000
    2        160     -3.250      0.401     -149043.000
    3        138     -2.540     -0.132           0.000
    4       1001     -1.000      0.000           0.000
    5       1002      0.000      1.000           0.000
    6       1003      0.000      0.000     -100000.000


OUTPUT DATA


CASE = 22  NZ = -3.800  NX = 0.607 THETADOT = 0.000 UNBAL MOM = 0.000
         X          Y          Z    FX    FZ  DMYY    SX    SZ         MXX         MYY         MZZ
    83.000    195.975     99.096     4   -25   283     4   -25           0         283           0
    83.000    185.925     98.040     4   -26   309     8   -51        -253         596         -40
    83.000    175.875     96.984     4   -28   337    13   -79        -770         942        -123
    83.000    165.825     95.928     5   -29   366    17  -108       -1563        1322        -250
    83.000    155.775     94.871     5   -30   397    22  -138       -2646        1737        -422
    83.000    145.725     93.815     5   -31   429    27  -169       -4030        2189        -643
    83.000    135.675     92.759     5   -32   462    32  -201       -5726        2679        -914
    83.000    125.625     91.703     5   -34   496    37  -235       -7749        3209       -1237
    83.000    115.575     90.647     6   -35   532    43  -270      -10109        3780       -1614
    83.000    105.525     89.590     6   -36   569    49  -306      -12820        4395       -2046
    83.000     95.475     88.534     6   -37   607    55  -343      -15894        5054       -2537
    83.000     85.425     87.478     6   -39   647    61  -382      -19344        5759       -3087
    83.000     75.375     86.422     6   -40   688    67  -422      -23182        6511       -3700
    83.000     65.325     85.365     7   -41   731    74  -463      -27421        7314       -4377
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83.000	55.275	84.309	7	-42	775	81	-505	-32074	8167	-5119
82.650	45.225	83.253	7	-44	831	88	-549	-37153	9259	-5930
79.895	35.175	82.197	8	-47	963	95	-597	-42675	11828	-6811
77.139	25.125	81.141	6	-36	786	101	-633	-48674	14360	-7769
74.383	15.075	80.084	0	0	0	101	-633	-55037	16212	-8784
71.628	5.025	79.028	0	0	0	101	-633	-61400	18063	-9800

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

CASE = 142 NZ = -3.250 NX = 0.401 THETADOT = -13.287 UNBAL MOM = -149043.000

X	Y	Z	FX	FZ	DMYY	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	3	-66	744	3	-66	0	744	0
83.000	185.925	98.040	3	-67	785	5	-133	-664	1532	-27
83.000	175.875	96.984	3	-67	826	8	-200	-2001	2363	-81
83.000	165.825	95.928	3	-68	864	11	-268	-4014	3236	-165
83.000	155.775	94.871	3	-68	899	15	-336	-6708	4147	-279
83.000	145.725	93.815	3	-68	932	18	-404	-10085	5095	-425
83.000	135.675	92.759	3	-68	962	21	-471	-14142	6076	-604
83.000	125.625	91.703	4	-67	989	25	-538	-18879	7087	-818
83.000	115.575	90.647	4	-66	1012	28	-605	-24289	8125	-1067
83.000	105.525	89.590	4	-65	1030	32	-670	-30366	9185	-1353
83.000	95.475	88.534	4	-64	1045	36	-734	-37100	10264	-1677
83.000	85.425	87.478	4	-63	1054	40	-797	-44481	11357	-2041
83.000	75.375	86.422	4	-61	1059	44	-859	-52494	12458	-2446
83.000	65.325	85.365	4	-60	1058	49	-918	-61124	13563	-2893
83.000	55.275	84.309	4	-58	1051	53	-976	-70352	14665	-3384
82.650	45.225	83.253	5	-56	1051	58	-1032	-80160	16113	-3920
79.895	35.175	82.197	5	-56	1130	63	-1087	-90526	20147	-4502
77.139	25.125	81.141	4	-39	851	67	-1126	-101453	24061	-5135
74.383	15.075	80.084	0	0	0	67	-1126	-112774	27235	-5806
71.628	5.025	79.028	0	0	0	67	-1126	-124096	30410	-6478

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

CASE = 138 NZ = -2.540 NX = -0.132 THETADOT = 0.000 UNBAL MOM = 0.000

X	Y	Z	FX	FZ	DMYY	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	-1	-17	189	-1	-17	0	189	0
83.000	185.925	98.040	-1	-18	207	-2	-34	-169	395	9
83.000	175.875	96.984	-1	-18	225	-3	-53	-514	618	27
83.000	165.825	95.928	-1	-19	245	-4	-72	-1045	860	54
83.000	155.775	94.871	-1	-20	265	-5	-92	-1769	1122	92
83.000	145.725	93.815	-1	-21	286	-6	-113	-2693	1403	140
83.000	135.675	92.759	-1	-22	309	-7	-135	-3828	1705	199
83.000	125.625	91.703	-1	-22	332	-8	-157	-5180	2030	269
83.000	115.575	90.647	-1	-23	355	-9	-180	-6757	2377	351
83.000	105.525	89.590	-1	-24	380	-11	-204	-8569	2747	445
83.000	95.475	88.534	-1	-25	406	-12	-229	-10624	3142	551
83.000	85.425	87.478	-1	-26	433	-13	-255	-12930	3562	671
83.000	75.375	86.422	-1	-27	460	-15	-282	-15495	4008	804
83.000	65.325	85.365	-1	-28	489	-16	-309	-18329	4481	951
83.000	55.275	84.309	-1	-28	518	-18	-338	-21439	4982	1112
82.650	45.225	83.253	-2	-29	555	-19	-367	-24834	5637	1289
79.895	35.175	82.197	-2	-32	644	-21	-399	-28525	7272	1480
77.139	25.125	81.141	-1	-24	526	-22	-423	-32535	8875	1688
74.383	15.075	80.084	0	0	0	-22	-423	-36788	10018	1909
71.628	5.025	79.028	0	0	0	-22	-423	-41041	11161	2130

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

CASE = 1001 NZ = -1.000 NX = 0.000 THETADOT = 0.000 UNBAL MOM = 0.000

X	Y	Z	FX	FZ	DMYY	SX	SZ	MXX	MYY	MZZ

83.000	195.975	99.096	0	-7	74	0	-7	0	74	0
83.000	185.925	98.040	0	-7	81	0	-14	-66	156	0
83.000	175.875	96.984	0	-7	89	0	-21	-203	245	0
83.000	165.825	95.928	0	-8	96	0	-28	-411	341	0
83.000	155.775	94.871	0	-8	104	0	-36	-696	445	0
83.000	145.725	93.815	0	-8	113	0	-44	-1060	558	0
83.000	135.675	92.759	0	-9	121	0	-53	-1507	680	0
83.000	125.625	91.703	0	-9	131	0	-62	-2039	810	0
83.000	115.575	90.647	0	-9	140	0	-71	-2660	950	0
83.000	105.525	89.590	0	-10	150	0	-80	-3374	1100	0
83.000	95.475	88.534	0	-10	160	0	-90	-4183	1260	0
83.000	85.425	87.478	0	-10	170	0	-100	-5091	1430	0
83.000	75.375	86.422	0	-11	181	0	-111	-6100	1611	0
83.000	65.325	85.365	0	-11	192	0	-122	-7216	1804	0
83.000	55.275	84.309	0	-11	204	0	-133	-8440	2008	0
82.650	45.225	83.253	0	-12	219	0	-145	-9777	2273	0
79.895	35.175	82.197	0	-12	253	0	-157	-11230	2924	0
77.139	25.125	81.141	0	-10	207	0	-167	-12809	3564	0
74.383	15.075	80.084	0	0	0	0	-167	-14484	4023	0
71.628	5.025	79.028	0	0	0	0	-167	-16158	4482	0

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

CASE = 1002 NZ = 0.000 NX = 1.000 THETADOT = 0.000 UNBAL MOM = 0.000

X	Y	Z	FX	FZ	DMYY	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	7	0	0	7	0	0	0	0
83.000	185.925	98.040	7	0	0	14	0	0	7	-66
83.000	175.875	96.984	7	0	0	21	0	0	21	-203
83.000	165.825	95.928	8	0	0	28	0	0	43	-411
83.000	155.775	94.871	8	0	0	36	0	0	73	-696
83.000	145.725	93.815	8	0	0	44	0	0	111	-1060
83.000	135.675	92.759	9	0	0	53	0	0	158	-1507
83.000	125.625	91.703	9	0	0	62	0	0	214	-2039
83.000	115.575	90.647	9	0	0	71	0	0	280	-2660
83.000	105.525	89.590	10	0	0	80	0	0	355	-3374
83.000	95.475	88.534	10	0	0	90	0	0	440	-4183
83.000	85.425	87.478	10	0	0	100	0	0	535	-5091
83.000	75.375	86.422	11	0	0	111	0	0	641	-6100
83.000	65.325	85.365	11	0	0	122	0	0	758	-7216
83.000	55.275	84.309	11	0	0	133	0	0	887	-8440
82.650	45.225	83.253	12	0	0	145	0	0	1028	-9777
79.895	35.175	82.197	12	0	0	157	0	0	1180	-11230
77.139	25.125	81.141	10	0	0	167	0	0	1346	-12809
74.383	15.075	80.084	0	0	0	167	0	0	1522	-14484
71.628	5.025	79.028	0	0	0	167	0	0	1698	-16158

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

CASE = 1003 NZ = 0.000 NX = 0.000 THETADOT = -8.915 UNBAL MOM = -100000.000

X	Y	Z	FX	FZ	DMYY	SX	SZ	MXX	MYY	MZZ
83.000	195.975	99.096	0	-30	337	0	-30	0	337	0
83.000	185.925	98.040	0	-30	350	0	-60	-301	686	0
83.000	175.875	96.984	0	-29	360	0	-89	-901	1047	0
83.000	165.825	95.928	0	-29	369	0	-118	-1796	1416	0
83.000	155.775	94.871	0	-28	376	0	-146	-2982	1792	0
83.000	145.725	93.815	0	-28	380	0	-174	-4454	2171	0
83.000	135.675	92.759	0	-27	381	0	-201	-6203	2552	0
83.000	125.625	91.703	0	-26	379	0	-226	-8220	2931	0
83.000	115.575	90.647	0	-24	374	0	-251	-10496	3304	0
83.000	105.525	89.590	0	-23	365	0	-274	-13017	3669	0
83.000	95.475	88.534	0	-22	352	0	-296	-15772	4021	0
83.000	85.425	87.478	0	-20	336	0	-316	-18744	4357	0
83.000	75.375	86.422	0	-18	315	0	-334	-21918	4673	0
83.000	65.325	85.365	0	-16	290	0	-350	-25276	4963	0

83.000	55.275	84.309	0	-14	260	0	-365	-28798	5223	0
82.650	45.225	83.253	0	-12	228	0	-377	-32463	5579	0
79.895	35.175	82.197	0	-10	206	0	-387	-36250	6823	0
77.139	25.125	81.141	0	-6	120	0	-392	-40139	8010	0
74.383	15.075	80.084	0	0	0	0	-392	-44083	9091	0
71.628	5.025	79.028	0	0	0	0	-392	-48028	10173	0

CONCENTRATED WEIGHTS

0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000
0.000	0.000	0.000	0.000	0.000

A.35 Net Loads, Case 22 PHAA

NET LOADS FOR CASE NUMBER: 22

X	Y	Z	FX	FZ	SX	SZ	MX	MY	MZ
83.000	195.975	99.096	-12.188	117.727	-12.188	117.727	0.000	84.630	0.000
83.000	185.925	98.040	-22.120	154.214	-34.308	271.941	1183.157	164.989	122.487
83.000	175.875	96.984	-29.515	179.993	-63.822	451.933	3916.161	231.080	467.279
83.000	165.825	95.928	-35.558	201.310	-99.381	653.243	8458.090	275.595	1108.694
83.000	155.775	94.871	-40.634	219.993	-140.015	873.236	15023.183	292.664	2107.469
83.000	145.725	93.815	-44.941	236.880	-184.956	1110.117	23799.208	277.448	3514.618
83.000	135.675	92.759	-48.604	252.440	-233.560	1362.557	34955.881	225.931	5373.426
83.000	125.625	91.703	-51.710	266.962	-285.271	1629.518	48649.574	134.787	7720.706
83.000	115.575	90.647	-54.326	280.639	-339.596	1910.157	65026.232	1.288	10587.676
83.000	105.525	89.590	-56.538	292.979	-396.134	2203.136	84223.308	-176.765	14000.616
83.000	95.475	88.534	-58.426	303.581	-454.561	2506.717	106364.825	-401.138	17981.768
83.000	85.425	87.478	-59.965	313.541	-514.526	2820.258	131557.332	-673.242	22550.104
83.000	75.375	86.422	-61.184	322.916	-575.710	3143.174	159900.926	-994.109	27721.091
83.000	65.325	85.365	-62.107	331.750	-637.817	3474.924	191489.827	-1364.424	33506.978
83.000	55.275	84.309	-62.755	340.077	-700.572	3815.001	226412.816	-1784.549	39917.041
82.650	45.225	83.253	-62.989	348.933	-763.561	4163.934	264753.577	-3584.848	46957.794
79.895	35.175	82.197	-61.996	364.321	-825.558	4528.256	306601.115	-15546.584	54631.583
77.139	25.125	81.141	-63.289	394.035	-888.846	4922.291	352110.084	-28847.591	62928.437
74.383	15.075	80.084	-68.501	448.385	-957.347	5370.676	401579.106	-44186.949	71861.341
71.628	5.025	79.028	-67.858	466.177	-1025.205	5836.853	455554.402	-60940.042	81482.679

A.36 Net Loads, Case 160 ACCEL ROLL 100% Side

NET LOADS FOR CASE NUMBER: 142

X	Y	Z	FX	FZ	SX	SZ	MX	MY	MZ
83.000	195.975	99.096	-14.048	76.420	-14.048	76.420	0.000	550.267	0.000
83.000	185.925	98.040	-24.247	113.095	-38.295	189.515	768.019	1109.944	141.182
83.000	175.875	96.984	-31.858	139.331	-70.154	328.846	2672.647	1665.773	526.052
83.000	165.825	95.928	-38.092	161.340	-108.246	490.187	5977.550	2206.907	1231.095
83.000	155.775	94.871	-43.342	180.943	-151.587	671.130	10903.924	2723.623	2318.963
83.000	145.725	93.815	-47.810	198.975	-199.397	870.105	17648.781	3206.907	3842.414
83.000	135.675	92.759	-51.623	215.902	-251.020	1086.006	26393.334	3648.234	5846.353
83.000	125.625	91.703	-54.872	232.014	-305.892	1318.021	37307.699	4039.430	8369.106
83.000	115.575	90.647	-57.622	247.505	-363.514	1565.526	50553.806	4372.575	11443.319
83.000	105.525	89.590	-59.962	261.898	-423.476	1827.424	66287.341	4639.930	15096.631
83.000	95.475	88.534	-61.970	274.803	-485.445	2102.227	84652.956	4833.838	19352.562
83.000	85.425	87.478	-63.623	287.294	-549.068	2389.521	105780.337	4946.638	24231.288
83.000	75.375	86.422	-64.951	299.431	-614.019	2688.952	129795.022	4970.686	29749.422
83.000	65.325	85.365	-65.979	311.258	-679.998	3000.209	156818.985	4898.323	35920.312
83.000	55.275	84.309	-66.728	322.811	-746.726	3323.020	186971.089	4721.838	42754.290
82.650	45.225	83.253	-67.074	335.052	-813.800	3658.072	220367.444	3278.008	50258.882
79.895	35.175	82.197	-66.302	353.755	-880.102	4011.827	257131.071	-7160.169	58437.570
77.139	25.125	81.141	-67.019	388.624	-947.121	4400.451	297449.931	-19013.120	67282.595
74.383	15.075	80.084	-70.303	445.877	-1017.424	4846.328	341674.463	-32956.221	76801.162
71.628	5.025	79.028	-69.691	463.568	-1087.115	5309.896	390380.059	-48305.539	87026.276

A.37 Net Loads, Case 138 STEADY ROLL Max Torsion

NET LOADS FOR CASE NUMBER: 138

X	Y	Z	FX	FZ	SX	SZ	MX	MY	MZ
83.000	195.975	99.096	2.093	53.015	2.093	53.015	0.000	-226.385	0.000
83.000	185.925	98.040	1.367	79.288	3.461	132.303	532.805	-470.519	-21.036
83.000	175.875	96.984	0.833	98.666	4.294	230.969	1862.450	-733.957	-55.815
83.000	165.825	95.928	0.453	115.265	4.747	346.234	4183.687	-1018.046	-98.970
83.000	155.775	94.871	0.207	130.292	4.954	476.526	7663.334	-1323.962	-146.673
83.000	145.725	93.815	0.086	144.305	5.040	620.831	12452.419	-1652.732	-196.460
83.000	135.675	92.759	0.081	157.620	5.121	778.450	18691.767	-2005.248	-247.112
83.000	125.625	91.703	0.185	170.431	5.306	948.882	26515.192	-2382.269	-298.579
83.000	115.575	90.647	0.394	182.870	5.700	1131.752	36051.452	-2784.433	-351.907
83.000	105.525	89.590	0.613	193.774	6.313	1325.525	47425.556	-3212.262	-409.193
83.000	95.475	88.534	0.756	202.210	7.069	1527.735	60747.086	-3666.260	-472.641
83.000	85.425	87.478	0.957	210.277	8.026	1738.012	76100.826	-4147.004	-543.685
83.000	75.375	86.422	1.213	218.014	9.239	1956.026	93567.847	-4655.003	-624.346
83.000	65.325	85.365	1.521	225.454	10.760	2181.481	113225.913	-5190.703	-717.198
83.000	55.275	84.309	1.878	232.621	12.637	2414.102	135149.794	-5754.486	-825.335
82.650	45.225	83.253	2.369	240.219	15.006	2654.321	159411.519	-7198.162	-952.340
79.895	35.175	82.197	3.502	252.438	18.508	2906.759	186087.445	-15203.364	-1103.153
77.139	25.125	81.141	5.171	274.501	23.680	3181.260	215300.372	-24214.030	-1289.160
74.383	15.075	80.084	7.737	313.300	31.417	3494.560	247272.037	-34709.831	-1527.139
71.628	5.025	79.028	9.079	327.928	40.495	3822.488	282392.366	-46282.671	-1842.875

A.38 Loads on Reciprocating Engine

```
ENGINE MOUNT LOADS FOR:
INTERNATIONAL IO-520-BB
MCDUFFY 3-84

INPUT DATA

LIMIT LOAD FACTOR                                      3.800
ENGINE WEIGHT, LBS                                   505.000
ENGINE CG
X CO-ORDINATE                                         22.000
Y CO-ORDINATE                                          0.000
Z CO-ORDINATE                                         92.000
PROPELLER WEIGHT, LBS                                 74.000
PROPELLER DIAMETER, INCHES                            84.000
NUMBER OF PROPELLER BLADES                                 3
PROPELLER TAKE-OFF RPM                                  2700
PROPELLER MAX CONT RPM                                  2500
PROPELLER CG
X CO-ORDINATE                                        -10.000
Y CO-ORDINATE                                          0.000
Z CO-ORDINATE                                        100.000
ENGINE TYPE                                       RECIPROCAL
TAKE-OFF HORSEPOWER                                  285.000
MAX CONT HORSEPOWER                                  265.000
TAKE-OFF TORQUE, FT-LBS                              554.388
NUMBER OF CYLINDERS                                        6

OUTPUT DATA

LIMIT TAKE-OFF TORQUE WITH 75% LIMIT MANEUVER VERTICAL LOAD FACTOR
     FAR                                          23.361(A)(1)
     VERTICAL LOAD FACTOR                                2.850
     VERTICAL DOWN LOAD, LBS                          1650.150
     APPLIED AT
     X CO-ORDINATE                                      17.910
     Y CO-ORDINATE                                       0.000
     Z CO-ORDINATE                                      93.022
     ENGINE MOUNT TORQUE, FT-LBS                      -554.388

FACTOR TIMES MAX CONT TORQUE WITH 100% LIMIT MANEUVER VERTICAL LOAD FACTOR
     FAR                                          23.361(A)(2)
     VERTICAL LOAD FACTOR                              3.800
     VERTICAL DOWN LOAD, LBS                        2200.200
     APPLIED AT
     X CO-ORDINATE                                    17.910
     Y CO-ORDINATE                                     0.000
     Z CO-ORDINATE                                    93.022
     TORQUE FACTOR                                     1.330
     MAX CONT TORQUE                                 556.723
     ENGINE MOUNT TORQUE, FT-LBS                    -740.441

SIDE LOAD IDEPENDENT OF OTHER FLIGHT LOADS
     FAR                                            23.363(A)&(B)
     VERTICAL LOAD FACTOR                              0.000
     SIDE LOAD FACTOR                                  1.330
     SIDE LOAD, LBS                                  770.070
     APPLIED AT
     X CO-ORDINATE                                    17.910
     Y CO-ORDINATE                                     0.000
     Z CO-ORDINATE                                    93.022
```

A.39 Landing Loads with respect to Ground Line

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CALCULATION OF LANDING GEAR GEOMETRY

INPUT

MAX LANDING WEIGHT           =   3230.000
GROSS WEIGHT                 =   3400.000
GEAR LOAD FACTOR             =      2.500
WING LIFT FACTOR             =      0.667
ROLLING RADIUS MAIN GEAR     =      8.000
ROLLING RADIUS NOSE GEAR     =      5.700
TREAD BETWEEN MAIN WHEELS    =    114.500
TAIL DOWN ANGLE GROUND TO WL =     15.000

XMG 25% DEF =     96.300   ZMG 25% DEF =     55.900
XNG 25% DEF =      1.900   ZNG 25% DEF =     46.900

XMG STATIC DEF =     96.700   ZMG STATIC DEF =     59.600
XNG STATIC DEF =      2.400   ZNG STATIC DEF =     49.500

XMG FULLY EXTENDED DEF =     96.200   ZMG FULLY EXTENDED DEF =     54.200  REF
XNG FULLY EXTENDED DEF =      1.600   ZNG FULLY EXTENDED DEF =     45.400  REF

WEIGHT/CG DATA:

     CG NO        WCG        XCG        ZCG
     5.000   3230.000     85.100     93.000
     5.000   3230.000     76.120     93.000
     7.000   2800.000     72.640     92.000

CALCULATED LANDING GEAR GEOMETRY

DRAG LOAD FACTOR, K =      0.324
GAMMA = ARCTAN(K)   =     17.978

GROUND ANGLE FOR 3 WHEEL LEVEL LANDING          =      4.057
GROUND ANGLE FOR LEVEL LANDING NOSE WHEEL CLEAR =      4.057
GROUND ANGLE FOR GROUND ROLL                    =      4.724
GROUND ANGLE FOR TAIL DOWN ATTITUDE             =     15.000

BETA = GAMMA - GROUND ANGLE = ANGLE BETWEEN RESULTANT LOAD AND FUS STA
BETA FOR 3 WHEEL LEV LAND         =     13.921
BETA FOR LEV LAND NOSE WHEEL CLEAR =     13.921
BETA FOR GROUND ROLL              =     -4.724
BETA FOR TAIL DOWN ATTITUDE       =     15.000

                                             AP         BP         DP         CP
3 PT AND 2 PT LEV LAND FOR CG NO    5     69.666     19.796     89.462
3 PT AND 2 PT LEV LAND FOR CG NO    5     60.950     28.512     89.462
3 PT AND 2 PT LEV LAND FOR CG NO    7     57.812     31.649     89.462
GROUND ROLL FOR CG NO               5     86.001      8.809     94.811     42.241
```

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GROUND ROLL FOR CG NO             5     77.052     17.759     94.811     42.981
GROUND ROLL FOR CG NO             7     73.501     21.309     94.811     42.271
TAIL DOWN ATTITUDE FOR CG NO      5      0.000      1.216      0.000
TAIL DOWN ATTITUDE FOR CG NO      5      0.000      9.890      0.000
TAIL DOWN ATTITUDE FOR CG NO      7      0.000     13.511      0.000


LIMIT GROUND REACTIONS AND LOAD FACTORS                         CG    7 2800.000  LBS  XCG=  72.640 ZCG=  92.000
---------------------------------------                         CG    5 3230.000  LBS  XCG=  76.120 ZCG=  93.000
VALUES ARE WITH RESPECT TO GROUND LINE -- DENOTED BY P (PRIME)  CG    5 3230.000  LBS  XCG=  85.100 ZCG=  93.000
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----------------------------------------------
CASE CONDITION         CG POINT   FUSELAGE  <- - - - - - - - - - NOSE GEAR - - - - - - - -> < - - - - - - - - - - - - - LEFT MAIN GEAR
- - - - - - - - - - - - - -><- - AIRPLANE- ->
     ----------------     OF      AXIS     ------------------------------------------------------------------------------------------
----------------------------------------------
 NO  DESCRIPTION       NO LOAD    ANGLE     EQUATIONS              VNP    DNP     SN RESULT EQUATIONS               VMP    DMP     SM
RESM EQUATIONS              NVP    NDP     NS
---------------------------------------------------------------------------------------------------------------------------------------
----------------------------------------------
 1 THREE WHEEL LEVEL    5         4.057     VP= 2.5WBP/DP         1787    580      0   1879 VMP= 1.25WAP/DP       3144   1020       0
3305 NVP=SUM(VP)        3.167  0.811   0.00
 2 LANDING              5                   DNP=.3244VPN          2574    835      0   2706 DMP= .324VMP          2751    893       0
2892         /W+ .667   3.167  0.811   0.00
 3 INCLINED REACTIONS   7         DEG WITH  S=0                   2476    804      0   2604 S=0                   2262    734       0
2378                    3.167  0.811   0.00
-------------------------                   -------------------------------------------------------------------------------------
----- NDP=SUM(DP)/W      -------------------
 4 TWO WHEEL LEVEL      5 CENTER  GROUND      NOSE                   0      0      0      0 VMP= 1.25W            4038   1310       0
4245                    3.167  0.811   0.00
 5 LANDING              5                     WHEEL                  0      0      0      0 DMP= .324VMP          4038   1310       0
4245 NS=SUM(S)/W        3.167  0.811   0.00
 6 INCLINED REACTIONS   7 OF      LINE        CLEAR                  0      0      0      0 S=0                   3500   1136       0
3680                    3.167  0.811   0.00
-------------------------                   -------------------------------------------------------------------------------------
-----                    -------------------
 7 TAIL DOWN LANDING    5 EACH    15.000      NOSE                   0      0      0      0 VMP= 1.25W            4038      0       0
4038                    3.167  0.000   0.00
 8 VERTICAL             5         DEG WITH    WHEEL                  0      0      0      0 DMP=0                 4038      0       0
4038                    3.167  0.000   0.00
 9 REACTIONS            7 WHEEL   GROUND      CLEAR                  0      0      0      0 S=0                   3500      0       0
3500                    3.167  0.000   0.00
                                  LINE
-------------------------                   -------------------------------------------------------------------------------------
-----                    -------------------
10 ONE WHEEL LEVEL      5         4.057       NOSE                   0      0      0      0 VMP= 1.25W            4038   1310       0
4245                    1.917  0.406   0.00
11 LANDING              5         DEG WITH    WHEEL                  0      0      0      0 DMP= .324VMP          4038   1310       0
4245                    1.917  0.406   0.00
12 INCLINED REACTIONS   7         GROUND      CLEAR                  0      0      0      0 S=0                   3500   1136       0
3680                    1.917  0.406   0.00
   LEFT WHEEL LOADED              LINE
```

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-----------------------------------------
13          NOSE       5                    VNP=1.33-2VMP         1498      0      0   1498 VMP=.665WAP/          1512   1210       0
1936 NVP=SUM(VP)/W      1.330  0.712   0.00
14 GROUND    GEAR      5                    DNP=0                 1825      0      0   1825     (.8CP+DP)         1348   1079       0
1727                    1.330  0.635   0.00
15 ROLL-     DOWN      7                    S=0                   1596      0      0   1596 DMP=.8VMP  S=0       1064    851       0
1363                    1.330  0.608   0.00
-- MAIN -----------------                    ---------------------------------------------------------------------------------------
----- NDP=SUM(DP)/W      ------------------
16 WHEELS    NOSE      5 GROUND              NOSE                    0      0      0      0 VMP=.665W             2261   1809       0
2895                    1.330  1.064   0.00
17 BRAKED    GEAR      5        4.724        WHEEL                   0      0      0      0 DMP=.8VMP             2261   1809       0
2895 NS=SUM(S)/W        1.330  1.064   0.00
18           CLEAR     7 CONTACT             CLEAR                   0      0      0      0 S=0                   1862   1490       0
2385                    1.330  1.064   0.00
-------------------------        DEG         ---------------------------------------------------------------------------------------
-----                    ------------------
19           LT DRIFT  5 POINT               NOSE WHEEL              0      0      0      0                       2261      0   -1700
2261                    1.330  0.000  -0.83
20 GROUND    RT DRIFT  5        WITH         CLEAR                   0      0      0      0                       2261      0    1122
2261                    1.330  0.000   0.83
-- ROLL -----------------                    ---------------------------------------------- VMP=.665W        --------------------
-----                    ------------------
21 SIDE LOAD LT DRIFT  5        GROUND       NOSE WHEEL              0      0      0      0 S(LD)=-.5W INBD       2261      0   -1700
2261                    1.330  0.000  -0.83
22 LEFT      RT DRIFT  5                     CLEAR                   0      0      0      0 S(RD)=.33W OUTB       2261      0    1122
2261                    1.330  0.000   0.83
-- GEAR -----------------        LINE        ----------------------------------------------                  --------------------
-----                    ------------------
23 VALUES    LT DRIFT  7                     NOSE WHEEL              0      0      0      0                       1862      0   -1400
1862                    1.330  0.000  -0.83
24           RT DRIFT  7                     CLEAR                   0      0      0      0                       1862      0     924
1862                    1.330  0.000   0.83
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---------------------------------------------
25           AFT       5 CL AXLE                                  711    569      0    910
26           FWD       5 CL AXLE                                  711   -284      0    766
27 SUPPLE-   SIDE      5 GROUND                                   711      0    498    711
-- MENTARY ----------------------                                -------------------------
28           AFT       5 CL AXLE           DNP=.8VNP  S=0        1433   1146      0   1835
29           FWD       5 CL AXLE           DNP=-.4VNP S=0        1433   -573      0   1543
30 NOSE      SIDE      5 GROUND            DNP=0  S=+.7VNP       1433      0   1003   1433
-- WHEEL ------------------------                                -------------------------
31 CONDITION AFT       7 CL AXLE                                 1416   1133      0   1813
32           FWD       7 CL AXLE                                 1416   -566      0   1525
33           SIDE      7 GROUND                                  1416      0    991   1416
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LIMIT GROUND REACTIONS AND LOAD FACTORS       CG  7 2800.000  LBS  XCG=  72.640 ZCG=  92.000
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---------------------------------------      CG  5 3230.000  LBS  XCG=  76.120 ZCG=  93.000
VALUES ARE WITH RESPECT TO AIRPLANE DATUM     CG  5 3230.000  LBS  XCG=  85.100 ZCG=  93.000
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-----------------------------------------
CASE CONDITION         CG POINT < - - - - - - - - - - - - - NOSE GEAR - - - - - - - - - - - - - >< - - - - - - - - - - - MAIN GEAR - - - - -
- - - - -  >< - - - - AIRPLANE - - - - >
     ----------------      OF     ----------------------------------------------------------------------------------------------
-----------------------------------------
 NO  DESCRIPTION        NO LOAD   PHIN      RESULT   EQUATIONS                VN      DN       SN  PHIM    RESM EQUATIONS                VM
DM      SM    NR     NV     ND     NS
------------------------------------------------------------------------------------------------------------------------------------------
-----------------------------------------
 1 THREE WHEEL LEVEL    5         13.921      1879 VN=RCOS(PHIN)         1823    452       0  13.92   3305                            3208
795      0  3.287  3.216  0.679  0.000
 2 LANDING               5         13.921      2706 DN=RSIN(PHIN)         2626    651       0  13.92   2892                            2807
696      0  3.287  3.216  0.679  0.000
 3 INCLINED REACTIONS   7         13.921      2604 SN=SNP                2527    626       0  13.92   2378                            2308
572      0  3.287  3.216  0.679  0.000
---------------------         -------------------------------------------------------------------------                       -------
-----------------------------------------
 4 TWO WHEEL LEVEL      5 CENTER  0.000          0 NOSE                     0      0       0  13.92   4245                            4120
1021      0  3.287  3.216  0.679  0.000
 5 LANDING               5         0.000          0 WHEEL                    0      0       0  13.92   4245 VM=RCOS(PHIM)             4120
1021      0  3.287  3.216  0.679  0.000
 6 INCLINED REACTIONS   7 OF      0.000          0 CLEAR                    0      0       0  13.92   3680                            3572
885      0  3.287  3.216  0.679  0.000
---------------------         -------------------------------------------------------------------------                       -------
-----------------------------------------
 7 TAIL DOWN LANDING    5 EACH    0.000          0 NOSE                     0      0       0 -15.00   4038 DM=RSIN(PHIM)             3900
-1045      0  3.167  3.059 -0.820  0.000
 8 VERTICAL              5         0.000          0 WHEEL                    0      0       0 -15.00   4038                            3900
-1045      0  3.167  3.059 -0.820  0.000
 9 REACTIONS             7 WHEEL   0.000          0 CLEAR                    0      0       0 -15.00   3500 SM=SMP                    3381
-906      0  3.167  3.059 -0.820  0.000

---------------------         -------------------------------------------------------------------------                       -------
-----------------------------------------
10 ONE WHEEL LEVEL      5         0.000          0 NOSE                     0      0       0  13.92   4245                            4120
1021      0  1.975  1.941  0.363  0.000
11 LANDING               5         0.000          0 WHEEL                    0      0       0  13.92   4245                            4120
1021      0  1.975  1.941  0.363  0.000
12 INCLINED REACTIONS   7         0.000          0 CLEAR                    0      0       0  13.92   3680                            3572
885      0  1.975  1.941  0.363  0.000
   LEFT WHEEL LOADED
---------------------         -------------------------------------------------------------------------                       -------
-----------------------------------------
13          NOSE       5         -4.724      1498 VN=RCOS(PHIN)         1493   -123       0  33.94   1936                            1606
1081      0  1.508  1.384  0.600  0.000
14 GROUND    GEAR       5         -4.724      1825 DN=RSIN(PHIN)         1819   -150       0  33.94   1727                            1433
964      0  1.474  1.378  0.523  0.000
15 ROLL-     DOWN       7         -4.724      1596 SN=SNP                1591   -131       0  33.94   1363                            1130
761      0  1.462  1.376  0.496  0.000
```

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-- MAIN    -----------------------------------------------------------------------------                       -------
-----------------------------------------
16 WHEELS    NOSE       5 GROUND  0.000          0 NOSE                     0      0      0  33.94   2895                          2402
1616      0  1.703  1.413  0.951  0.000
17 BRAKED    GEAR       5         0.000          0 WHEEL                    0      0      0  33.94   2895                          2402
1616      0  1.703  1.413  0.951  0.000
18           CLEAR      7 CONTACT 0.000          0 CLEAR                    0      0      0  33.94   2385                          1978
1331      0  1.703  1.413  0.951  0.000
---------------------           -------------------------------------------------------------------------                       -------
-----------------------------------------
19           LT DRIFT   5 POINT   0.000          0 NOSE WHEEL               0      0      0  -4.72   2261                          2253
-186  -1700  1.330  1.325 -0.110 -0.830
20 GROUND    RT DRIFT   5         0.000          0 CLEAR                    0      0      0  -4.72   2261                          2253
-186   1122  1.330  1.325 -0.110  0.830
---------------------           -------------------------------------------------------------------------                       -------
-----------------------------------------
21 SIDE LOAD LT DRIFT   5         0.000          0 NOSE WHEEL               0      0      0  -4.72   2261                          2253
-186  -1700  1.330  1.325 -0.110 -0.830
22 LEFT      RT DRIFT   5         0.000          0 CLEAR                    0      0      0  -4.72   2261                          2253
-186   1122  1.330  1.325 -0.110  0.830
---------------------           -------------------------------------------------------------------------                       -------
-----------------------------------------
23 VALUES    LT DRIFT   7         0.000          0 NOSE WHEEL               0      0      0  -4.72   1862                          1856
-153  -1400  1.330  1.325 -0.110 -0.830
24           RT DRIFT   7         0.000          0 CLEAR                    0      0      0  -4.72   1862                          1856
-153    924  1.330  1.325 -0.110  0.830
--------------------------------------------------------------------------------------------------------------------------------------
-----------------------------------------
25           AFT        5 CL AXLE 33.938       910                       755    508      0
26           FWD        5 CL AXLE -26.527      766                       685   -342      0
27 SUPPLE-   SIDE       5 GROUND  -4.724       711                       708    -59    498
-- MENTARY  --------------------------------------                      --------------------
28           AFT        5 CL AXLE 33.938      1835 VN=RCOS(PHIN)        1522   1024      0
29           FWD        5 CL AXLE -26.527     1543 DN=RSIN(PHIN)        1381   -689      0
30 NOSE      SIDE       5 GROUND  -4.724      1433 SN=SNP               1428   -118   1003
--          --------------------------------------                      --------------------
31 CONDITION AFT        7 CL AXLE 33.938      1813                      1504   1012      0
32           FWD        7 CL AXLE -26.527     1525                      1365   -681      0
33           SIDE       7 GROUND  -4.724      1416                      1411   -117    991
---------------------------------------------------------------------------------------------



LIMIT UNBALANCED MOMENTS                                                                    CG  7 2800.000  LBS  XCG=  72.640 ZCG=  92.000
------------------------                                                                    CG  5 3230.000  LBS  XCG=  76.120 ZCG=  93.000
VALUES ARE WITH RESPECT TO GROUND LINE - DENOTED BY P(PRIME)AND TO AIRPLANE DATUM          CG  5 3230.000  LBS  XCG=  85.100 ZCG=  93.000
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-----
CASE CONDITION         CG POINT < - - - - - - - RELATIVE TO GROUND LINE - - - - - - - ><- - - - - - RELATIVE TO AIRPLANE DATUM - - - -
- ->
```

NO	DESCRIPTION	NO LOAD	EQUATIONS	PITCHING	ROLLING	YAWING	EQUATIONS	PITCHING	ROLLING	YAWING
		OF								
1	THREE WHEEL LEVEL	5	VN=RCOS(PHIN)	0	0	0		0	0	0
2	LANDING	5	DN=RSIN(PHIN)	0	0	0		0	0	0
3	INCLINED REACTIONS	7	SN=SNP	0	0	0		0	0	0
4	TWO WHEEL LEVEL	5 CENTER	NOSE	-168058	0	0		-168058	0	0
5	LANDING	5	WHEEL	-242054	0	0		-242054	0	0
6	INCLINED REACTIONS	7 OF	CLEAR	-232918	0	0		-232918	0	0
7	TAIL DOWN LANDING	5 EACH	NOSE	-9827	0	0		-9827	0	0
8	VERTICAL	5	WHEEL	-79870	0	0		-79870	0	0
9	REACTIONS	7 WHEEL	CLEAR	-94579	0	0		-94579	0	0
10	ONE WHEEL LEVEL	5	NOSE	-84029	231147	-75000	PMOM=PMOMP	-84029	235873	-58460
11	LANDING	5	WHEEL	-121027	231147	-75000		-121027	235873	-58460
12	INCLINED REACTIONS	7	CLEAR	-116459	200375	-65016	RMOM=RMOMP*COS(GA)	-116459	204472	-50678
	LEFT WHEEL LOADED						+YMOMP*SIN(GA)			
13	NOSE	5	VN=RCOS(PHIN)	0	0	0	YMOM=YMOMP*COS9GA)	0	0	0
14	GROUND GEAR	5	DN=RSIN(PHIN)	0	0	0	-RMOMP*SIN(GA)	0	0	0
15	ROLL- DOWN	7	SN=SNP	0	0	0		0	0	0
--	MAIN									
16	WHEELS NOSE	5 GROUND	NOSE	-192652	0	0		-192652	0	0
17	BRAKED GEAR	5	WHEEL	-235797	0	0		-235797	0	0

```
18            CLEAR      7 CONTACT CLEAR                  -205293         0          0                          -205293          0
0
---------------------           ------------------------------------------------------                        -------------------------
-----
19           LT DRIFT   5 POINT   NOSE WHEEL             -39838    -119206     -24861                        -39838    -116754     -
34594
20 GROUND    RT DRIFT   5          CLEAR                    -39838     119206      24861                           -39838     116754
34594
---------------------           ------------------------------------------------------                        -------------------------
-----
21 SIDE LOAD LT DRIFT   5         NOSE WHEEL             -80308    -121293     -50117                        -80308    -116754     -
59936
22 LEFT      RT DRIFT   5          CLEAR                    -80308     121293      50117                           -80308     116754
59936
---------------------           ------------------------------------------------------                        -------------------------
-----
23 VALUES    LT DRIFT   7         NOSE WHEEL             -79358     -98238     -49524                        -79358     -93826     -
57447
24            RT DRIFT   7          CLEAR                    -79358      98238      49524                           -79358      93826
57447
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A.40 Landing Load Factor

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ESTIMATED AIRPLANE LANDING LOAD FACTOR

INPUT

LANDING WEIGHT  =   3230.000
WING AREA       =    184.121
STROKE OF AXLE  =      7.000
OD OF TIRE      =     19.000
HUB DIAMETER    =      7.000
LIFT FACTOR     =      0.667
MAIN GEAR IS OLEO

OUTPUT
SINK RATE            =      9.005
AIRPLANE LOAD FACTOR =      3.095
LANDING GEAR FACTOR  =      2.428
```

A.41 Chordwise Distribution of 13 Critical Horizontal Tail Loads

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CHORDWISE DISTRIBUTION OF CRITICAL LOADS ON AVE CHORD OF HORIZONTAL TAIL

INPUT

AREA OF LH HORIZONTAL TAIL (SQR INCHES)          2660.000
AREA OF LH ELEVATOR (SQR INCHES)                 1181.000
AREA OF LH ELEVATOR FWD OF HL (SQR INCHES)        118.000
AREA OF LH ELEVATOR AFT OF HL (SQR INCHES)       1065.000
SEMI SPAN OF TAIL (INCHES)
TOTAL TAIL LIFT AT 25 & 50 PERCENT MAC        SHOWN BELOW

CHORDWISE STATIONS FROM THE LEADING EDGE DEFINING THE LOAD DISTRIBUTION
X1 =      0.000
X2 =      9.097
X3 =     36.389
X4 =     14.569
X5 =     21.819

DISTRIBUTIONS FOR THE CRITICAL LOAD FOR THE FOLLOWING CONDITIONS:
 1. UP BALANCING TAIL LOAD FLAPS RETRACTED
 2. DOWN BALANCING TAIL LOAD FLAPS RETRACTED
 3. UP BALANCING TAIL LOAD FLAPS EXTENDED
 4. DOWN BALANCING TAIL LOAD FLAPS EXTENDED
 5. UNCHECKED MANEUVER DOWN TAIL LOAD (ELEV TE UP)
 6. UNCHECKED MANEUVER UP TAIL LOAD (ELEV TE DN)
 7. DOWN LOAD CHECKED MANEUVER TAIL LOAD
 8. UP LOAD CHECKED MANEUVER TAIL LOAD
 9. UPGUST TAIL LOAD FLAPS RETRACTED
10. DOWN GUST TAIL LOAD FLAPS RETRACTED
11. UP GUST TAIL LOAD FLAPS EXTENDED
12. DOWN GUST TAIL LOAD FLAPS EXTENDED
13. UNSYMMETRICAL TAIL LOAD

OUTPUT
```

COND	LT25	LT50	PSI(X1)	PSI(X2)	PSI(X3)	PSI(X4)	PSI(X5)
1	907.620	-387.770	0.682	0.095	0.000	0.015	-0.030
2	217.580	-831.500	0.164	-0.122	0.000	-0.228	-0.239
3	-34.760	-62.090	-0.026	-0.019	0.000	-0.025	-0.023
4	-532.850	-496.120	-0.401	-0.197	0.000	-0.236	-0.209
5	-51.600	-1227.790	-0.039	-0.250	0.000	-0.393	-0.390
6	65.040	1072.700	0.049	0.222	0.000	0.346	0.343
7	-458.460	-218.340	-0.345	-0.129	0.000	-0.137	-0.114
8	700.300	87.480	0.527	0.149	0.000	0.133	0.098
9	843.460	65.040	0.634	0.171	0.000	0.147	0.105
10	-1186.700	-106.000	-0.892	-0.244	0.000	-0.212	-0.152
11	-478.670	3.520	-0.360	-0.089	0.000	-0.071	-0.047
12	-1087.520	-161.300	-0.818	-0.236	0.000	-0.214	-0.160
13	-1186.810	-106.000	-0.892	-0.244	0.000	-0.212	-0.152

A.42 Chordwise Distribution of 4 Critical Vertical Tail Loads

```
CHORDWISE DISTRIBUTION OF CRITICAL LOADS ON AVE CHORD OF VERTICAL TAIL

INPUT

AREA OF VERTICAL TAIL (SQR INCHES)           2137.392
AREA OF RUDDER (SQR INCHES)                   754.000
AREA OF RUDDER FWD OF HL (SQR INCHES)          82.000
AREA OF RUDDER AFT OF HL (SQR INCHES)         667.000
SPAN OF VERTICAL TAIL (INCHES)                 57.000
TOTAL VERTICAL TAIL LOAD AT 25 & 50 % MAC    SHOWN BELOW

CHORDWISE STATIONS FROM THE LEADING EDGE DEFINING THE LOAD DISTRIBUTION
X1 =      0.000
X2 =      9.375
X3 =     37.498
X4 =     11.702
X5 =     25.796

DISTRIBUTIONS FOR THE CRITICAL LOAD FOR THE FOLLOWING CONDITIONS:
1. MANEUVER LOAD FOR SUDDEN FULL RUDDER DEFLECTION
2. MANEUVER LOAD FOR YAW TO SIDESLIP OF 19.5 DEG WITH RUDDER MAINTAINED AT FULL DEFLECTION
3. MANEUVER LOAD FOR YAW OF 15 DEG WITH RUDDER IN NEUTRAL
4. SIDE GUST LOAD AT VC

OUTPUT

         LT25       LT50   PSI(X1)   PSI(X2)   PSI(X3)    PSI(X4)     PSI(X5)
   1     0.000    679.000     0.000     0.370     0.000      0.462       0.462
   2 -1076.000    679.000    -2.014    -0.133     0.000      0.000       0.252
   3  -827.000      0.000    -1.548    -0.387     0.000     -0.355      -0.161
   4   950.000      0.000     1.778     0.444     0.000      0.408       0.185
```

A.43 Chordwise Distribution of Loads on Station Chords of Horizontal Tail

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CHORDWISE DISTRIBUTION OF LOADS ON STATION CHORDS OF HORIZONTAL TAIL

INPUT

FOR 1. UP BALANCING TAIL LOAD FLAPS RETRACTED
ANGLE OF ATTACK LOAD (LH+RH) AT 25% MAC                 907.620
CAMBER LOAD (LH+RH) AT 50% MAC                         -387.770
AREA OF HORIZONTAL TAIL LH (SQR INCHES)                2660.000
AREA OF LH ELEVATOR AFT OF HL(SQR INCHES)              1065.000
NUMBER OF STATIONS                                        4.000

BUTT LINE STATION                                        11.500
CHORD OF TAIL AT ABOVE BL                                43.316
CHORD OF ELEVATOR (INCHES)                               19.376
CHORD OF ELEVATOR AFT OF HL (INCHES)                     18.193

BUTT LINE STATION                                        27.500
CHORD OF TAIL AT ABOVE BL                                40.409
CHORD OF ELEVATOR (INCHES)                               18.154
CHORD OF ELEVATOR AFT OF HL (INCHES)                     17.064

BUTT LINE STATION                                        42.000
CHORD OF TAIL AT ABOVE BL                                35.769
CHORD OF ELEVATOR (INCHES)                               15.041
CHORD OF ELEVATOR AFT OF HL (INCHES)                     14.036

BUTT LINE STATION                                        59.500
CHORD OF TAIL AT ABOVE BL                                32.590
CHORD OF ELEVATOR (INCHES)                               13.704
CHORD OF ELEVATOR AFT OF HL (INCHES)                     12.801


CHORDWISE STATIONS FROM LEADING EDGE AND LOAD PSI DISTRIBUTION
BL STATION      X1      X2      X3      X4      X5 PSI(1) PSI(2)  PSI(3) PSI(4)  PSI(5)
    11.500   0.000  10.829  43.316  18.193  25.123  0.682  0.098  0.000  0.010  -0.026
    27.500   0.000  10.102  40.409  17.064  23.345  0.682  0.099  0.000  0.010  -0.026
    42.000   0.000   8.942  35.769  14.036  21.733  0.682  0.093  0.000  0.017  -0.032
    59.500   0.000   8.148  32.590  12.801  19.789  0.682  0.093  0.000  0.017  -0.032
```

A.44 Chordwise Distribution of Loads on Station Chords of Vertical Tail

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CHORDWISE DISTRIBUTION OF LOADS ON STATION CHORDS OF VERTICAL TAIL

INPUT

FOR 2. MANEUVER LOAD FOR YAW TO SIDESLIP OF 19.5 DEG WITH RUDDER MAINTAINED AT FULL DEFLECTION

ANGLE OF ATTACK LOAD AT 25% MAC                    -1076.000
CAMBER LOAD AT 50% MAC                               679.000
AREA OF VERTICAL TAIL (SQR INCHES)                  2137.392
AREA OF RUDDER AFT OF HL(SQR INCHES)                 667.000
NUMBER OF WL STATIONS                                  3.000

WATER LINE STATION                                   120.000
CHORD OF TAIL AT ABOVE WL                             51.318
CHORD OF RUDDER (INCHES)                              16.724
CHORD OF RUDDER AFT OF HL (INCHES)                    15.726

WATER LINE STATION                                   140.000
CHORD OF TAIL AT ABOVE WL                             40.638
CHORD OF RUDDER (INCHES)                              13.244
CHORD OF RUDDER AFT OF HL (INCHES)                    12.454

WATER LINE STATION                                   160.000
CHORD OF TAIL AT ABOVE WL                             29.956
CHORD OF RUDDER (INCHES)                               9.763
CHORD OF RUDDER AFT OF HL (INCHES)                     9.180

CHORDWISE STATIONS FROM THE LEADING EDGE DEFINING THE LOAD DISTRIBUTION
BL STA      X1     X2     X3     X4     X5 PSI(1) PSI(2) PSI(3) PSI(4) PSI(5)
120.000  0.000 12.830 51.318 15.726 35.592 -2.014 -0.127  0.000 -0.004  0.256
140.000  0.000 10.160 40.638 12.454 28.184 -2.014 -0.127  0.000 -0.004  0.256
160.000  0.000  7.489 29.956  9.180 20.776 -2.014 -0.127  0.000 -0.004  0.256
```