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STUDY FOR THE OPTIMIZATION OF A TRANSPORT AIRCRAFT WING FOR MAXIMUM FUEL EFFICIENCY

VOLUME I - METHODOLOGY, CRITERIA, AEROELASTIC MODEL DEFINITION, AND RESULTS

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Burbank, California

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VOLUME I - METHODOLOGY, CRITERIA, AEROELASTIC MODEL DEFINITION, AND RESULTS

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Burbank, California

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NATIONAL AERONAUTICS AND SPACE ADMINISTARTION

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PREFACE

This report is a documentation of work performed at Lockheed California Company in the design of a transport aircraft wing for maximum fuel efficiency. It forms the basis for transmitting to NASA-LaRC material associated with Lockheed's design criteria, design methodology, and three design configurations. The design database includes complete finite element model description, sizing data, geometry data, loads data, and inertial data. This report illustrates a design process which satisfied the economics and practical aspects of a real design.

The report also dicusses the cooperative study relationship between Lockheed and NASA during the course of the contract.

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1.0 INTRODUCTION

Volume I of the final report for contract NAS1-16794, "Study for Optimization of a Transport Aircraft Wing for Maximum Fuel Efficiency" is the main body of the final report. Volume II describes the database which was provided to NASA on a VAX compatible tape and includes selected material transmitted to NASA during the contract period.

The two volumes are in a working paper format and provide the bases for documentation of a cooperative study between NASA-LaRC and Lockheed-California Company for this contract. The volumes are especially directed to a description of Lockheed's design methodology and to the wing design effort performed at Lockheed.

The Lockheed wing optimization effort involves three processes. The first process selects the best airplane configuration based on the design constraints, mission and performance requirements, and the best data available for the wing weight as a function of aspect ratio, thickness to chord ratio, wing area, and sweep. This process also includes parametric models for aerodynamics, and propulsion. The process uses small models for development, production, and procurement costs determination.

The second process generates aeroelastic designs near the airplane configuration which was formed in the first process. The aeroelastic point designs form a bases for validating/updating the weight representation over the immediate design space.

The third process in this wing optimization effort is the repeat of the first process with the updated aeroelastic weight data.

The aeroelastic design effort involves the wing cover sizings for the Baseline airplane, and two perturbations from the Baseline, namely, aspect ratio 12 for 35 and 25 degrees of sweep. The Baseline airplane has an aspect ratio of 7.63 and a sweep of 35 degrees.

Weight data generated from these specific wing design studies formed a bases to update the statistical weight equation used in the performance/airplane parameter sizing program. A new optimized airplane configuration was derived for fuel efficiency with the following fixed constraints: weight to thrust ratio, range, wing loading, and payload.

The initial wing geometry design variables for the aeroelastic point designs included thickness to chord ratio (t/c), and wing area together with aspect ratio and sweep. While the detailed aeroelastic study was limited to changes in aspect ratio and sweep, the configuration study based on the parametric weight equation which was normalized to the aeroelastic point design results did include wing design variables t/c, area, sweep, and aspect ratio.

Lockheed's perspective of the cooperative study will be illustrated by repeating some of the ideas presented in the proposal for this contract, with some minor updating.

1.1 Formulation Of The Cooperative Study Relationship

The Interdisciplinary Research Office (IRO) at NASA-LaRC formulated a task to optimize a transport aircraft wing for maximum fuel efficiency subject to constraints dictated by structures, aerodynamics, aircraft performance, and active controls. This task is intended to be a focus of the IRO activity over the next five to ten years which will promote the development of new analysis and synthesis techniques in the participating disciplines. The task was presented to the NASA-LaRC Director on July 17, 1979 and to Lockheed-California Company (Calac), Douglas (Long Beach), and the Boeing Commercial Aircraft Division (Seattle) in September of 1979. Copies of that presentation entitled "A Proposed Airframe Configuration Optimization Methodology" were distributed to the IRO team members, the IRO Steering Committee, the NASA-LaRC management, and the above companies.

The Lockheed-California Company responded to the presentation with a willingness to consider participation in the proposed task as a cooperative effort with NASA-LaRC.

A meeting was held at LaRC on February 28 and 29, 1980 to define the technical substance and organizational arrangements for the proposed transport aircraft wing optimization task.

A proposal was submitted and accepted by NASA. The contract was initiated September 1981. The contract was completed in February 1985 and this report which includes two volumes is the documentation of the material sent to NASA during the contract, and in particular the optimization procedure and results from the Lockheed effort.

1.2 Program Objectives

The general program objectives are stated for the participating organizations and together they comprise the objectives of the program.

Lockheed objectives:

1. Participate in the optimization study to get first hand information on the advantages and limitations of the LaRC proposed optimization methodology.

2. Define the preliminary design characteristics resulting from the Lockheed-California Company optimization methods and compare with results obtained by LaRC using their upgraded methods and nonlinear programming optimization techniques.

3. Exercise preliminary design tools through a complete wing design cycle.

NASA-LaRC objectives:

1. Methodology for airframe optimization:

    a. Development

    b. Demonstration and evaluation of effectiveness.

2. Advancement of methods in participating disciplines (aerodynamics, structures, active controls, performance, optimization)

    a. New analysis techniques tailored for optimization applications

    b. Techniques for disciplinary suboptimizations

    c. New computer programs for 2a and 2b.

3. Set of publications for 1a, 1b, 2a, 2b, and 2c.

4. Understanding of the associated physical phenomena separately and collectively (synergistically).

1.3 Background

1.3.1 General methodology - Lockheed-California Company is actively developing a preliminary design methodology which automates the structural sizing for static and dynamic loads, strength, fatigue, flutter, and active controls. The structural sizing of aircraft to satisfy aeroelastic constraints uses the Preliminary Aeroelastic Design of Structures (PADS) and Advanced System Synthesis and Evaluation Technique (ASSET) computing systems. The ASSET system generates an aircraft configuration which satisfies design objectives including performance and cost. PADS sizes an aircraft to satisfy aeroelastic constraints and performs the suboptimization task data for use in updating ASSET's parametric structural weight database. ASSET then reevaluates the aircraft configuration using the new structural database provided by PADS. An iteration cycle between PADS and ASSET continues until ASSET has sufficient information to extrapolate structural data during the synthesis of an optimal aircraft configuration.

Other disciplines such as aerodynamics, propulsion, stability and control, performance, and cost are represented in ASSET by relatively small computing routines and parametric data formats. ASSET computer runs consume 2 to 4 cpu minutes (IBM 3081) for one configuration definition whereas one aeroelastic design definition consumes 2 to 5 cpu hours. However, each small computing module or parametric representation has a large computing program which support the numerical definition in ASSET. The ASSET team updates any small model in ASSET when the design goes outside the applicable design space for that small model. PADS, therefore, is just another large computing model which supports one of the small computing routines in ASSET.

The NASA-LaRC methodology under development within IRO is experimental and its verification is one of the objectives of the cooperative effort. The methodology is based on nonlinear mathematical programming with all the engineering disciplines concurrently contributing their design variables and constraints to the algorithm that decides how to change the design to improve a figure of merit (objective function). The optimization may be organized as a two, or more, level scheme to separate the gross design variables (e.g., sweep angle) from the detailed ones (e.g., spar cap cross-section area). It uses sensitivity analysis in many contributing disciplines, and retains the man in the loop by providing the engineer with means to monitor and to override the optimization algorithm decisions during the optimization process. The NASA-LaRC methodology development is based on an integrated system of computer programs, called multilevel multidisciplinary optimization system (MMOPS), operated as part of IRO.

1.3.2 Analysis methods - It was found that both Lockheed and NASA-LaRC use similar analyses in most cases. The objective of the exercise, however, may be compromised if careful attention is not paid to possible differences in either the analytical programs used by the different disciplines and/or the discipline's model representation. A conclusion based on optimization results using different methodologies may reflect only that the analysis procedures were different between Lockheed and NASA and not the method of optimization (suboptimization versus nonlinear programming). It is therefore vital to perform the initial sizing of the structure and the analysis of the structure for static and dynamic loads, and flutter under carefully controlled conditions by both organizations. The results of these efforts should be understood before proceeding to the more complex optimization task.

1.3.2.1 Weight calculations - The weight calculations are divided into two major categories, variable weight and fixed weight. Variable weight consists of that portion of the structure which is influenced by the design or structural concept being considered. Variable weight can be further separated into idealized structural weight which would include skins, spars and ribs and non-optimum weights such as rivets, backup plates, and sealants. Non-optimum weights are in the range of 25 percent of the idealized structural weight.

Fixed weights consist of aircraft components which are unaffected by the structural design process. Items such as the avionics system, the aircraft APU, landing gear, and engine would be possible candidates for fixed weights.

1.3.2.2 Gust loads and flutter - Gust Loads and Flutter may affect high aspect ratio wing design. Lockheed wing optimizations will include gust loads and flutter for the Baseline design and at least one design perturbation. Differences were noted in the methods used to resize the airframe structure for gust, and in the use of mission analysis versus a design envelope as sources for the loads data. Lockheed resizing is based on the equal probability combinations of internal loads on individual structural components. This is in agreement with LaRC approach. The design envelope is used in the LaRC work as the only source of airframe loads, while both design envelope and mission analysis are used as such sources in the Lockheed work.

1.3.2.3 Performance calculations - The issue of the drag calculations was an open item at the start of the contract. This is a difficult area to make compatible between NASA and Lockheed. There will be exchange of data to calibrate the performance modules with respect to aerodynamic representations.

1.3.3 Object of optimizations - An L-1011 derivative aircraft known as the Baseline aircraft will be the starting point for the optimization. The optimization included in the cooperative task will be carried out only for the aircraft wing with fuel efficiency as the objective function. The task of designing the aircraft wing for maximum fuel efficiency corresponds also to one of the objectives of the NASA-LaRC managed Aircraft Energy Efficiency (ACEE) project in which Lockheed was one of the contractors.

1.3.4 The Initial Scope of the optimization - The scope of the effort is defined in terms of loads, design constraints, design variables, and the way to include active controls.

- o Loads and Design Constraints - The number of loading cases will initially be limited to 15, drawn from a design envelope defined by Lockheed. The loading cases will pertain to maneuver, dynamic gust, landing, and taxiing. These loads will be reflected in strength constraints.

- o Design Variables - The proposed target set of design variables is wing box cover thickness, thickness to chord ratio, aspect ratio, wing area, sweep, and chordwise translation of an engine on the wing. The optimization procedure will begin with the three most potentially effective variables, aspect ratio (AR), wing sweep, and thickness to chord ratio. Subsequently, the pool of free variables will be extended by a few new variables at a time and optimization will continue away from the previous "best" design. This "sequential inclusion" approach should help to control computer cost and should yield additional information as to the importance of variables of various types.

- o Inclusion of Active Controls - The active control effects will be included for maneuver and dynamic gust loads as well as for flutter. It is an iterative approach which calls first for optimization of the airframe with some constraints relaxed in anticipation of the use of active controls. Next, a synthesis of active controls is carried out to achieve the requirements associated with the previously relaxed constraints. These two basic operations of the airframe optimization and active control synthesis will continue alternating in an iterative loop until acceptable results are established in both.

- Vehicle Synthesis - The aeroelastic suboptimization results will be used to update the wing weight parametric data in ASSET. A new configuration, based on new wing weight parametric data, will be derived for minimum weight while conforming to the airplane performance requirement. The new configuration may then be optimized for aeroelastic effects if the new configuration differs significantly from the previous iteration.

1.3.5 Proprietary information - All Lockheed data to be transmitted under this contract to NASA will be of non-proprietary nature. Data associated with the engine performance may be restricted under the federal guide lines for exposure to a foreign country. If data are required that are Lockheed proprietary, the data will be transmitted outside of this contract at no cost to NASA. Accordingly, the dissemination of Lockheed proprietary data will be negotiated separately.

1.3.6 Travel and communication - Alternating travel, using teleconferences, and transferring data directly from computer to computer are some of the means of assuring communication effectiveness while holding down its cost.

1.4 Reassessment Of The Contract Scope And Objectives

The contract was initiated with the understanding that a comparison of different optimized designs between Lockheed and LaRC was the appropriate tool to show the strengths and limitations of the multilevel optimization procedures which are under development at LaRC relative to Lockheed's sub-optimization procedures.

Since then, a consensus has been formed at Lockheed that the sizing details of the respective designs may not be nearly as important as the design path taken in arriving at the respective designs. The intention of the contract and consequently this report is to provide Lockheed's wing design data, design criteria, and details of the design path to NASA-LaRC which will facilitate the demonstration of the multilevel optimization development. A comparison of the design path between Lockheed and NASA-LaRC would have a significant impact on providing industry a basis for possible integration of the multilevel process into their respective design efforts. The design process defined in this report is representative of current state of the art. The full implementation of the multilevel approach is a significant departure from the traditional approach and a comparison of the path of design would be most interesting.

The study of the design paths, as one objective of the method comparisons, would relieve the requirement that the analysis modules between LaRC and Lockheed need to be almost identical.

2.0 DEFINITION OF DESIGN METHODOLOGY AT CALAC

Preliminary design for an optimal aircraft configuration requires the integration of aeroelastic considerations into the configuration selection and design process. Aeroelastic design incorporates the effects of aircraft flexibility on static and dynamic loads, control effectiveness, and aeroelastic stability into the sizing of the structure. Configurations of aircraft in the early design stage are usually based on statistical and analytical weight methods computed from approximate loads and stress analyses. This often leads to the freezing of external geometry before strength and flutter analyses are sufficiently advanced, thereby decoupling the powerful, but time-consuming, process of structural design-to-minimum-weight from the configuration optimization process.

In the past, the level of effort required for an accurate aeroelastic design was not justifiable relative to the answers provided by statistical methods which were supported by historical databases. Today, however, there are many combinations of advanced technologies and configurations, such as supercritical airfoils, high aspect ratio wings, forward swept wings, active controls, aeroelastic tailoring, and new materials, that have no historical database from which to derive the statistical or parametric weight equations.

Airplane design involves complete interactions between the conceptual designer, the customer with design specifications, and the engineers with final design and manufacturing requirements. Since many facets of the engineering process defy quantification, the computer methodology used to improve the flow of design information must be: 1) flexible, and 2) highly modular. Flexibility will permit inputs into the design process from many sources, and modularity will deter obsolescence when new engineering design processes become available.

The goal of Preliminary Aeroelastic Design of Structures (PADS) is to develop computer program operating system architecture and design methodology to generate an accurate aeroelastic design within the conceptual and early preliminary design phases. This aeroelastic design database will permit more accurate weights to be established during the configuration trade-off studies. The long-term goal is to define an accurate aeroelastic design within an elapsed time which is measured in weeks and to perform a design perturbation within days. Design perturbations include changes to any variable which does not require significant data preparation. For the wing, these variables will include sweep, planform definition, taper, airfoil sections, t/c, and aspect ratio.

The work to achieve these goals is in progress, PADS capabilities currently include a structural finite element model generator, weight distribution, grid transformations, steady maneuver loads for symmetric conditions, flutter analysis, and structural sizing.

Aeroelastic analysis of an aircraft structure is a substantial undertaking involving many disciplines and complex data paths. A short time ago, preliminary aeroelastic analysis was reserved for projects on the verge of achieving go-ahead status while preliminary aeroelastic design was not even attempted.

There are two options available for acquiring a rapid aeroelastic analysis and design capability: generate or acquire special programs tailored to rapid analysis procedures; or adapt existing engineering methodology and the associated computer tools to requirements of rapid analyses. Software maintenance is a major part of any proposed computer-aided design system. Lockheed-California Company (Lockheed) has an extensive library of computer programs which support airplane design through final and production design phases. It would be convenient to extend the application of that software into the preliminary design phase instead of creating specialized software.

Computer programs used to perform final aeroelastic analyses are well established in any major aircraft design company. At Lockheed, the list of computer programs include:

- A user-friendly matrix algebra based computing system
- Grid transformation programs
- A finite element based structural analysis program
- Steady and unsteady aerodynamic programs
- Weight estimation and distribution programs
- Steady maneuver aeroelastic load programs
- Transient maneuver aeroelastic load programs
- Ground handling load programs
- Dynamic loads (gust, taxi, landing) programs
- Flutter analysis programs

- Structural resizing program with stiffness (flutter) and deflection
- Structural sizing programs for stress and fatigue
- Feedback control functions synthesis programs for load relief and flutter suppression
- Database management (DBM) programs for card image data, matrices, and tables
- Structural finite element model generator program
- Plotting programs
- General utility programs known as pre- and postprocessors

When PADS development was initiated in 1976, computer programs and computing systems were generally available at Lockheed for preliminary aeroelastic design, but certain computer access and job preparation problems prevented these programs from being used on quick design studies. In particular:

- The overwhelming number of details associated with computer job designation made the most trivial computer job setup a challenge. Misplacing a single comma or being off one space in the input format field caused the job to fail.
- The computing tasks were serially related to each other. Job "A" had to run successfully before Job "B" could be submitted. Many computer jobs required one day turnaround and, with setup difficulties, one week could pass before the results were available.
- Analytical modeling requirements for each discipline were different.
- Data flow between disciplines involved paper flow and manual transcription.
- The trail on "How was it done last time?" was not always protected from unfavorable personnel transfers.
- Many preparations for computer job submittals required transcribing of input data by hand.
- When the process required hundreds of computer submittals and hundreds of individual input deck setups, it was impractical to mobilize engineering personnel just to feed the computer for a quick project.

Against the background of existing data management systems, existing computing systems of great sophistication, and high-level languages oriented to the user-friendly atmosphere, Calac decided to use the production design computing tools and to attack directly their known deficiencies with respect to preliminary design applications. A computing system was postulated which would act as a bare tree from which existing computer programs could be hung as needed in a user-friendly and highly modular environment.

2.1 Calac's ASSET/PADS Design Approach

Lockheed's Advanced Systems Synthesis and Evaluation Technique (ASSET) computer program provides a rapid and cost-effective solution to configuration selection for any aircraft mission, but only within the limitation that the structural weight must be based on semi-analytical and statistical data. An ASSET study usually requires that a baseline aircraft model be created and exercised in ASSET to represent an actual known aircraft database. This model then is modified through adjustments to parametric coefficients to simulate changes to baseline aircraft systems, structural arrangement, material usage, design parameters, and mission requirements. Once complete, the model is passed into the ASSET design cycle for sizing, configuration trade-off analysis, and performance evaluation.

The PADS goal is to update the weight database during configuration trade-off studies as well as to perform general aeroelastic analysis and design in a highly computerized environment. Figure 2.1-1 shows the possible interaction between PADS and ASSET during a typical configuration trade-off study. Aeroelastic inputs to ASSET should lead to significant improvements in the configuration selection process, especially when advanced designs combine advanced structural materials, such as composites, with unusual geometry.

PADS development consists of three distinct efforts:

- The development of the computer program operating system which will permit continuous computing capability in a user-friendly and engineering-defined environment.

- The definition and mechanization of basic engineering processes for use in aeroelastic design and analysis.

- The validation and upgrading of these procedures by exercising the system through an aeroelastic design cycle

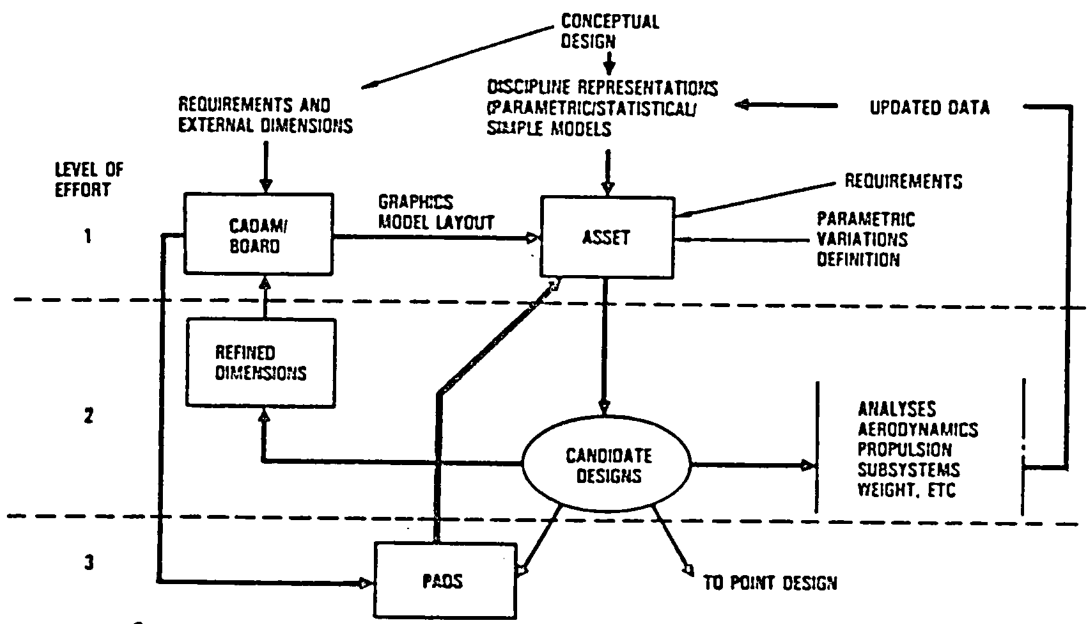


FIGURE 2.1-1 PADS and ASSET Interface

2.2 Modeling Criteria

Each engineering discipline will define the modeling requirements according to its functions and responsibilities. There will be no outside constraints except where more than one discipline will be affected. For the modeling tasks which affect many disciplines, such as structural finite element modeling, unsteady aerodynamics, and weight distributions, one discipline will be designated as "prime", usually by tradition. This discipline will then integrate the requirements of the other cognizant disciplines into the modeling definition.

Since existing computer programs will be used in the PADS system, the conditioning of data from various sources will be accomplished by the aggressive use of pre- and postprocessors. If the flow of data is interrupted in an existing engineering process with a large number of subjective interpretations, and if the engineering process is not amenable to conversion to a more continuous data flow, a new or modified process will be defined, where possible, to facilitate this data flow.

2.3 The Design Process

The first step in the design process is to define the objectives of the task and the necessary level of design detail required to satisfy those objectives. The design team must review the requirements, cost out the project, and define a schedule. This is an iterative process between the customer and the design team. This phase is labelled DESIGN OBJECTIVES in figure 2.3-1.

The next step is the generation of the structural finite element model, initial weight distribution, and initial entries into the various modules to generate geometry tables for each grid system to be used in the design. This provides the necessary database for the computation of all transformation matrices between the various grid systems.

The initial internal loads are generated from static loads for a rigid airplane and uniform properties (starting properties) for those structural finite elements to be sized. A panel sizing and stress allowable (PSASA) process then generates initial sizing and the associated stress allowables for the specified margins of safety. There may be multiple entries into the internal load and sizing procedures until a converged rigid airplane sizing is obtained.

The computations for static loads and internal loads are repeated using the sizing derived from the rigid airplane

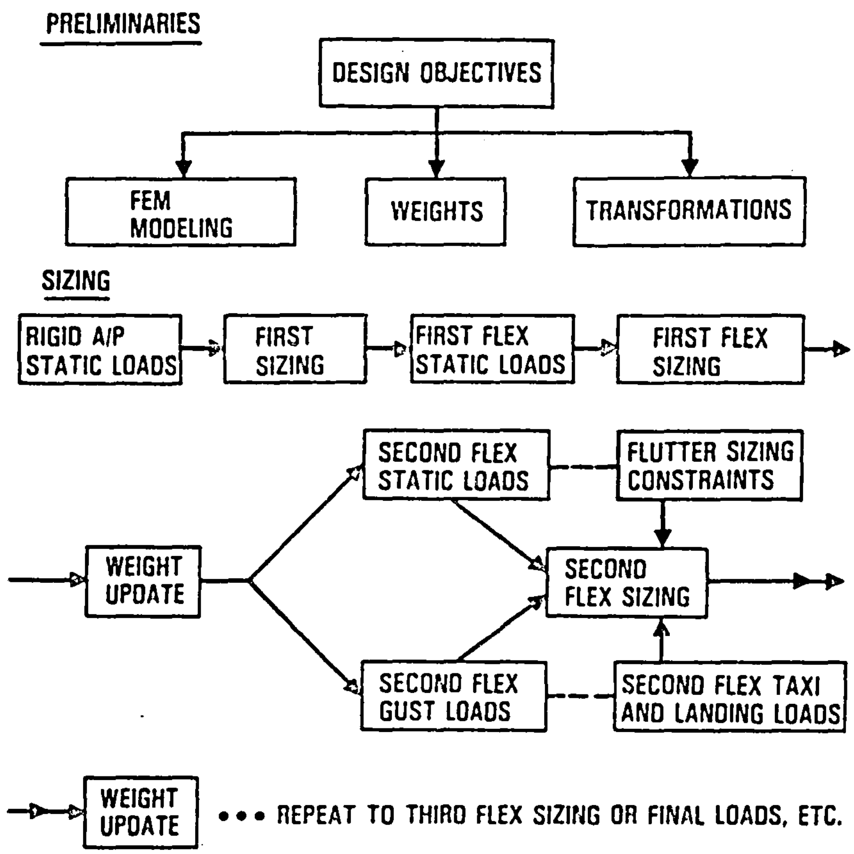


FIGURE 2.3-1 Design Process Overview

loads. The term "first flex sizing" refers to the sizing produced from the static loads formed with the design's first flexibility matrix.

The first flex sizing provides a basis for updating the weight data and for generating dynamic loads input along with flutter minimum sizing constraints. The second flex sizing should be close to the sizing which satisfies strength and stiffness requirements. If the loads and stiffness are highly coupled, another iteration may be required.

2.4 The Application Of PADS Against A Known Design

The general theme behind PADS is the use of known analytical tools in the application of aeroelastic sizing in preliminary design. The effort to automate the overall design process, however, required many preprocessors to generate the data that were traditionally input from worksheets. The automation effort also required the generation of a high level operating system which would permit the execution of design processes rather than separate programs. The PADS system and concepts required a known design for a benchmark as proof that the overall PADS approach did perform as desired.

Other benefits of passing a known design through a design tool such as PADS is the determination of weight factors required to convert finite element weight data generated by a design tool into actual hardware weight.

The application of a known design to PADS in the context of supporting the cooperative study with NASA-LaRC had another constraint. NASA-LaRC had decided to delay the dynamic gust loads and flutter constraints inclusion into the design to some later date. Lockheed performed the design on the baseline aircraft with and without gust loads. The gust loads weight effect was less than a 2 percent increase on the cover weights. The gust loads at the higher aspect ratio had a similar effect on cover weight. Flutter was not an active constraint in the baseline design, and flutter speed deficiencies above dive velocity could be solved on high aspect ratio designs (AR=12) with active controls.

There are two sources of weight differences between the PADS finite element model (FEM) sizing and the actual hardware weight. The first source is the difference between the PADS FEM sizing and the airplane production FEM sizing. The PADS sizings are based on limited design conditions and no analytical correction factors. The production sizings come from actual airplane production drawings which reflect windtunnel pressure data corrections, strength and fatique test data, and flight test data.

The second source is the difference between production FEM model weight and hardware weight. Hardware weight include sealants, stringer runout structure, access holes support structure, etc.

The PADS FEM sizing did not include gust loads effects when the factors were computed to account for the differences between the two FEM sizings. The factors established for the FEM differences are sightly larger than if the PADS FEM sizing had included the gust loads effects.

Additional discussion of these two weight factors and their derivations are found in section 3.3. The application of the weight factors to a new design are found in section 5.

3.0 DEFINITION OF BASELINE AIRPLANE

The Baseline aircraft selected for this design process was a wide-body, three- engine, long- range transport airplane with active controls. The airplane 3-view is shown in figure 3.0-1. This configuration has a maximum gross takeoff weight of 504,000 pounds and a payload of 40,000 pounds at a range of 5200 nautical miles. The cruise mach number is 0.83 and the cruise altitude is 39,000 feet. Active control technology was used to minimize structural weight and improve fuel economy. The maximum design zero fuel weight is 338,000 pounds and the typical operating empty weight is 252,000 pounds. The Baseline airplane has an aspect ratio of 7.64. Definition of aspect ratio as well as some other geometric parameters can be found in APPENDIX F.

Table 3.0-1 contains wing, elevator, and horizontal stabilizer primary dimensions. Reference dimensions are dimensions which have been fixed between designs and may not be the actual aircraft dimensions. All reference dimensions within this report will be noted. Figure 3.0-2 displays the wing bones drawing containing keypoint and coordinate system information for description of the wing leading and trailing edge, ailerons, and wing box substructure. The information on the wing bones drawing is used for generation of the finite element model. This figure is described in greater detail in volume II of this report.

Figure 3.0-3 contains keypoint information for the fuselage and depicts the origin for the basic coordinate system. This figure depicts a side view of the finite element model. Distance from the origin on the X0, Y0, or Z0 axes are sometimes referred to as fuselage (FS), buttline (BL), or waterline (WL) stations respectively.

The general geometry of the Baseline airplane is identical to that of the L-1011-500 airplane. That is, the external geometric description is the same. Internal structural definitions are unique for the Baseline airplane.

The term L-1011-500 airplane as used herein refers to the existing production L-1011-500 airplane with Active Control System (ACS). This configuration was used to assist in the selection of symmetric flight conditions to be considered for the Baseline airplane analyses.

The term Baseline airplane as used herein refers to the the basic airplane with external geometric shapes and dimensions identical to that of the L-1011-500 airplane. Structural properties of the Baseline airplane are unique.

The term AR12 airplane as used herein refers to the Baseline airplane with an increased aspect ratio wing, relocated wing engine and pylon, and a relocated main landing gear. AR12 designs of 35 and 25 degree sweep were formulated for the PADS design study. Structural properties of the increased aspect ratio wings are different from those of the Baseline airplane wing. Structural properties for all other airplane components are the same.

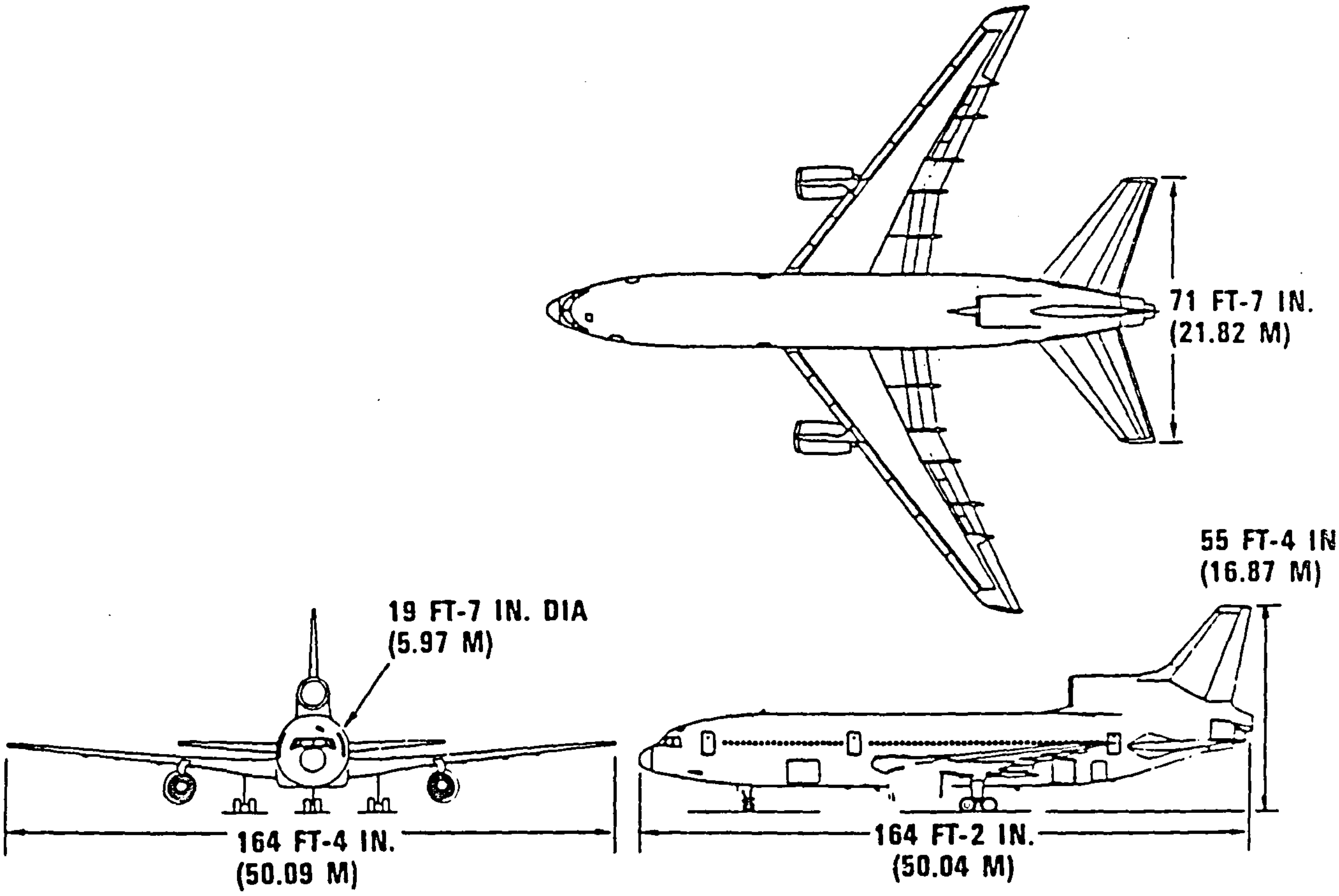


FIGURE 3.0-1 Baseline 3-VIEW

TABLE 3.0-1 Baseline Airplane Dimensions

Wing:

Area*	3552 ft.
Span*	164 ft. 4 in.
Aspect Ratio*	7.64
Mean Aerodynamic Chord*	289.20 in.
Fuselage Station of 0.25% Chord on MAC*	1225.60 in.
Root Chord*	34.33 ft.
Sweep (25% Chord Line)*	35 deg.
Area (reference)	3456 ft.
Span (reference)	155 ft.
Aspect Ratio (reference)	6.95
Mean Aerodynamic Chord (reference)	293.50 in.
Fuselage Station of 0.25% Chord on MAC (reference)	1216.40 in.

Horizontal Stabilizer: (Primary Longitudinal Control)

Area (gross)	1282 ft.
(exposed)	961 ft.
Span (perpendicular to airplane centerline)	859 in.
Root Chord	323 in.
Tip Chord	107 in.
Aspect Ratio	4
Mean Aerodynamic Chord (streamwise)	233 in.
Sweep (25% chord line)	35 deg.
Dihedral @ Trailing Edge	3 deg.
Fuselage Station of 0.25 Chord on MAC	1881.3
Deflection limits relative to FRL	+1 deg. to -14 deg.

Elevator (Geared to Horizontal Stabilizer)

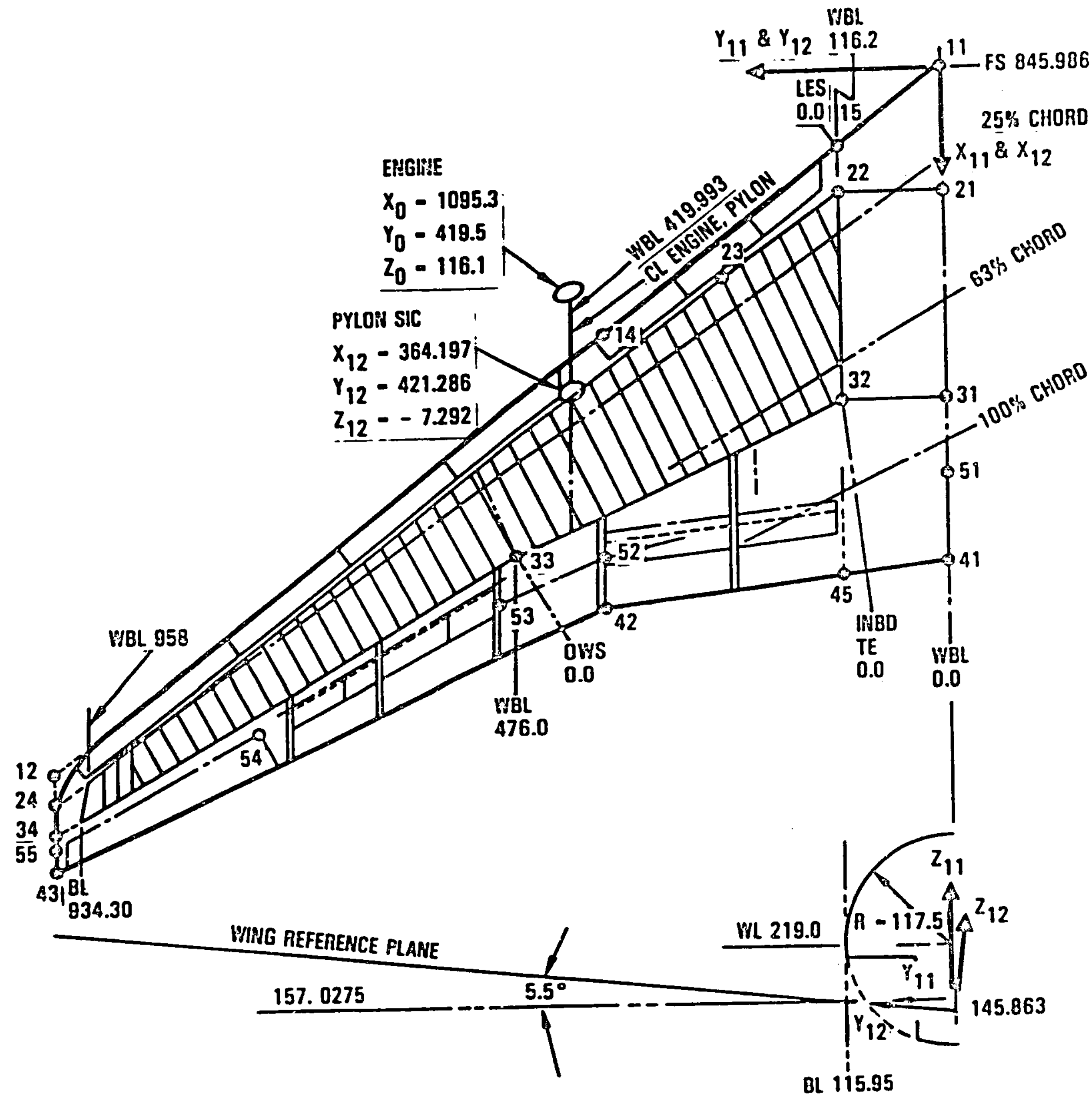
Area (aft of hinge line per side)	127.5 sq.ft./side
Span (perpendicular to airplane centerline)	368 in.
Mean Aerodynamic Chord (streamwise)	52.0 in.
Deflection Limits (perpendicular to hinge line)	0 deg. to -25 deg. as stab. goes from +1 deg . to -14 deg.

Note:

* - Computed for Baseline design wing, applied when defining AR12 wings

reference - Not actual, used as a reference in selected calculations or descriptions (originated from L-1011 wing prior to tip extension)

FRL - Fuselage reference line



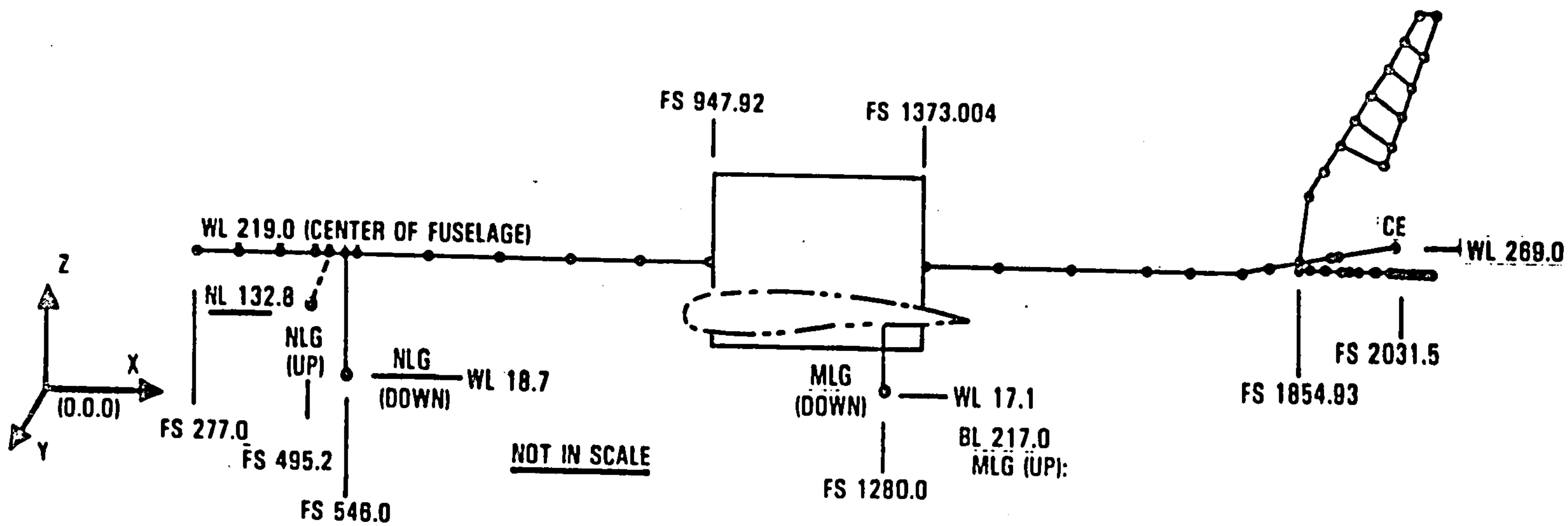
*** WING BONE PARAMETERS ***

POINT,11,, 0.0 , 0.0 , 0.0
POINT,12,,765.256,988.301,0.0
POINT,14,,299.493,386.780,0.0
POINT,15,,90.1969,116.486,0.0
POINT,21,,137.01 , 0.0 , 0.0
POINT,22,,137.014,116.486,0.0
POINT,23,,232.402,251.063,0.0
POINT,24,,803.256,988.301,0.0
POINT,31,,366.014, 0.0 , 0.0
POINT,32,,366.014,116.486,0.0
POINT,33,,535.374,476.0 , 0.0
POINT,34,,832.223,988.301,0.0
POINT,41,,539.697, 0.0 , 0.0
POINT,42,,591.849,386.775,0.0
POINT,43,,871.553,988.301,0.0
POINT,45,,555.404,116.486,0.0
POINT,51,,447.381, 0.0 , 0.0
POINT,52,,536.710,386.780,0.0
POINT,53,,586.193,493.197,0.0
POINT,54,,724.961,766.947,0.0
POINT,55,,845.437,988.301,0.0

*** COORDINATE SYSTEMS ***

FUSELAGE	0 (GLOBAL)
EMPENNAGE	0 (GLOBAL)
CENTER FUSELAGE	11 (LOCAL)
WING CENTER BOX	11 (LOCAL)
WING	12 (LOCAL)

FIGURE 3.0-2. BASELINE FINITE ELEMENT WING BONES DRAWING



MLG – MAIN LANDING GEAR
NLG – NOSE LANDING GEAR
CE – CENTER ENGINE
FS – FUSELAGE STATION (X COORDINATE)
BL – BUTTLINE STATION (Y COORDINATE)
WL – WATERLINE STATION (Z COORDINATE)

X_0 - 1277.0
Y_0 - 112.2
Z_0 - 150.2

FIGURE 3.0-3. FUSELAGE KEYPOINT INFORMATION

3.1 Design Criteria

3.1.1 Introduction - The flight environment, airplane configuration, and maneuver conditions used for external loads analysis were selected from existing load envelopes for the L-1011-500 airplane. The selected conditions represent positive and negative steady maneuvers for both the clean and flapped configuration, ground handling, dynamic taxi, and dynamic landing conditions. No transient pitch maneuvers, rolling maneuvers, or lateral-directional maneuvers were considered. All external load conditions analysed for this task are in full compliance with current FAR-25 regulations.

An active control system (ACS) is included in the external loads analysis by symmetric deflection of the wing outboard aileron as a function of load factor. External load distributions were generated for both system on and system off conditions.

Initial ACS control laws for the Baseline airplane were considered to be the same as those for the existing L-1011-500 airplane since the definition of control laws to determine the required aileron deflection angle is not part of this criteria definition. However, the magnitude of the active control gains could be used as a variable in the design with the understanding that the definition of the hardware to be compatible with the established gains is a completely different task.

With the system on, the outboard ailerons are biased down two degrees at one-g with the flaps retracted, and biased up 8 degrees with the flaps down. The ACS aileron deflection due to maneuver excursions was -11.33 degrees per "delta g", where delta g is measured relative to the one-g trim flight position. ACS system off external load distributions were generated with the outboard ailerons in the neutral position. Resulting external load distributions were then factored prior to inclusion in the internal stress analysis. The system off factors are defined in section 3.4.2.3.

The primary objective was to generate net balanced external load distributions for a minimum number of conditions sufficient to determine a first level structural sizing.

Structural deflections due to applied external load distributions are measured from the wing jig shape. The wing jig shape was obtained by deflecting the wing from the prescribed aerodynamic midcruise shape while under a balanced

1-g trim load condition at the midcruise flight condition. For the Baseline analysis, the midcruise flight condition was defined as: gross weight of 350,300 lb; c.g. at 12.5% MAC (reference); altitude of 39,000 ft; and a mach of 0.83.

3.1.2 Design philosophy and methods - The basic design approach employed the same methodology as that used for a prospective preliminary design or production phase design cycle at Lockheed. That is, loads were obtained as panel loads, using two and three dimensional grid systems. Basic data, such as inertia distributions, aerodynamic load distributions, structural model stiffness characteristics, etc., were created for a number of different grid systems. Balanced net external load distributions required for internal stress analysis at the finite element model were generated at a basic loads grid system and then transformed to the SIC grid system as equivalent net load distributions.

Extensive use was made of matrix concepts and matrix algebra both in data handling and in solving the aeroelastic equations to obtain the net balanced panel load distributions. External loads analysis as done at Lockheed utilizes a high level matrix algebra system called Flutter And Matrix Algebra System, (FAMAS). Thus, FAMAS formed the foundation upon which all external load distributions were generated. The FAMAS system was used to carry out the closed-form aeroelastic solutions in which panel loads were defined over the entire airplane for the various flight maneuver and ground handling conditions. It was also used to generate elastic vibration modes and related input data used in the dynamic loads analyses and in the final processing of dynamic loads data into panel loads distributions.

A number of different grid systems were required to accommodate the various analyses. Loads analysis as done at Lockheed does not require that all disciplines use a common grid. Each discipline is allowed to define grids which best reflect their needs. Transformation matrices are then created which will transform data at one grid to the required equivalent data type at another grid. These matrices are of the following general types:

1. Grid transformation matrices which when multiplied by panel loads or deflections in one grid yields loads or deflections in another grid.

2. Differentiating matrices which when multiplied by structural deflections yield angles of attack at aerodynamic collocation points.

3. The alpha (angle-of-attack) distribution due to airplane camber and twist at the aerodynamic collocation points.

4. Alpha-delta distributions which define the effective angle of attack at each aerodynamic collocation point due to unit deflections of control surfaces (stabilizer,

elevator, ailerons, flaps, slats, etc.).

5. Load integration matrices which when multiplied by panel loads in a given grid system yield load quantities such as shear, bending moments, torsions, hinge moments, empennage loads, etc.. Although internal stress analysis requires panel load distributions, computer plots of integrated load quantities provide a high degree of data visibility, and facilitates parametric studies and comparisons.

Generation of these type transformations is a highly automated process at Lockheed. It encompassed the use of a series of pre- and post-processors to established program modules. All of the programs which generate basic data required for panel loads methods automatically generate the data files necessary to execute these pre- and post-processors.

3.1.2.1 Aerodynamic load distributions - Aerodynamic data for this analysis were generated using the theoretical vortex lattice method of VORLAX, Reference 1. This method, developed at the Lockheed-California Company under Contract NAS1-12972, is described by NASA Contract Report 2865, "A Generalized Vortex Lattice for Subsonic and Supersonic Flow Applications". It is Lockheed's understanding that VORLAX, per Report 2865, is an in-house theoretical method at NASA Langley.

The VORLAX Fortran algorithm listed in Report 2865 was implemented as a FAMAS subroutine, referred to as P-130, and is available as a module within FAMAS. Solutions generated by either the basic program of Report 2865 or P-130 are identical. In addition to the basic algorithm, P-130 includes routines designed to output geometric and downwash arrays not available as output when using the Fortran version of Report 2865. The advantage of operating within the FAMAS system is that all output is in matrix format, and thus readily available for other matrix operations. Since most of the external loads programs used for production analysis at Lockheed reside in FAMAS, the incorporation of P-130 into the FAMAS system greatly enhanced completion of the analysis.

Theoretical aerodynamic load distributions were generated for the Baseline and the AR12 airplanes at mach = 0.50, 0.80, and 0.88. Downwash matrices for each configuration were obtained as output from P-130. These matrices were converted to aerodynamic influence coefficient (AIC) matrices by inversion and application of local panel geometry characteristics. AIC matrices were then used to generate airload distributions for both the rigid and the flexible airplane. Flexible load distributions are generated as a fully closed solution, not as an iterative process.

3.1.2.2 Weight distributions - Weight distributions required to generate net balanced load distributions on the rigid or flexible airplane were generated for designated weight and center-of-gravity combinations required for both flight and ground handling conditions.

Weight distribution data required for external loads analysis are provided in the form of inertia matrices. These matrices, provided by the Weight Division at Lockheed, consisted of the following three sets of distributed data:

{ Pz/Nz } - vertical load at each load panel due to a unit inertia load factor acting at the A/P cg. Units = lb

{ Pz/ 0 } - vertical load at each load panel due to a unit pitching acceleration acting at the A/P cg. Units = lb-sec-sec

{ Pz/ 0 } - vertical load at each load panel due to a unit rolling acceleration acting at the A/P cg. Units = lb-sec-sec

These matrices represent the inertia load distributions on the left side of the airplane.

Inertia distributions were generated for flight conditions and ground handling conditions, as follows:

	Weight	c.g.- %MAC @	Fuel	Gear	Type Conditions
1.)	350300	12.5	12500	UP	SYM. FLIGHT MAN.
2.)	504000	17.13	165000	UP	SYM. FLIGHT MAN
3.)	368000	14.1	30000	DN	FAR-25 GRD. HAND.
4.)	506000	26.9	168000	DN	FAR-25 GRD. HAND. LL-2, DYN. TAXI

@ - reference MAC

3.1.3 Loads general data -

3.1.3.1 Configuration data - Figure 3.0-1 presents a three view drawing of the Baseline airplane.

All reference areas and lengths are the same for all configurations considered.

3.1.3.2 Sign convention - The aerodynamic sign convention for the Baseline and the AR12 airplanes are the same. It corresponds to a right hand rule. Table 3.1-1 presents a summary of this sign convention.

The sign convention for integrated loads, moments, accelerations, and deflections is the same for the Baseline and the AR12 airplanes. Table 3.1-2 presents a summary of this sign convention.

3.1.3.3 Load axes - Integrated shears, bending moments and torsions about a wing reference load axis are generated for both the Baseline and the AR12 airplanes. Table 3.1-3 defines the load axes and wing stations at which these data are generated.

ITEM	POSITIVE WHEN
Lift, L, $C_L = L/qS$	Up
Drag, D, $C_D = D/qS$	Aft
Thrust, T, $C = T/qS$	Forward
Pitching Moment, M, $C_m = M/qS\bar{c}$	Nose Up
Side Force, Y, $C_Y = Y/qS$	Right
Yawing Moment, N, $C_n = N/qSb$	Nose Right
Rolling Moment, L, $C_\ell = L/qSb$	Right Wing Down
Hinge Moment, H, $C_h = H/qS^* \bar{c}^*$	Moment Which Moves Trailing Edge Down or to the Left
Angle of Attack, αFRL	Above Relative Velocity
Angle of Yaw, ψ	Nose to Right
Angle of Sideslip, β	Nose to Left
Elevator Angle, δ_e	Trailing Edge Down
Rudder Angle, δ_R	Trailing Edge to Left
Aileron Angle, δ_a	Trailing Edge Down

S^* = area of control surface or tab aft of hinge center line

$\bar{c}^*$ = mean chord of control surface or tab

TABLE 3.1-1 Load Aero Sign Convention

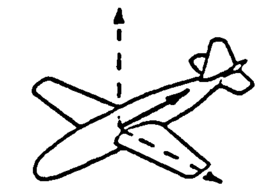
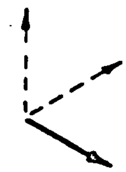
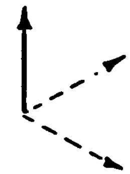
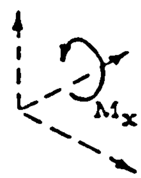
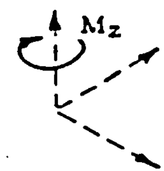
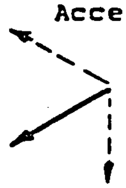
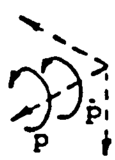
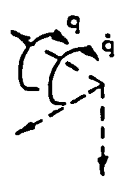
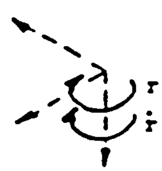
Axis	X	Y	Z
Coordinate External Forces Load Factor			
External Moments			
Translational Velocities and Accelerations	Vel. = u Accel. = a_x 	Vel. = v Accel. = a_y 	Vel. = w Accel. = a_z 
Angular Velocities and Accelerations			
Angular Displacements	The sign of an angle of incidence, angle of attack, angle of yaw, and control surface displacement is determined from the left hand rule in accordance with axis system shown for coordinates above. Exception: Negative sideslip angle results from application of this rule.		
Internal Loads	Internal loads and moments are positive if the loads or moments applied by the part with the greater algebraic coordinate are positive in accordance with axis system shown for coordinates above.		

TABLE 3.1-2 Loads Sign Convention

TABLE 3.1-3   WING LOAD AXES SYSTEMS

Component	Baseline	AR12	AR12	Comment
Wing Sweep - deg	35	25	35	
FS @ BL = 0	1052.98	1325.5	1014.3	
BL @ CUT 1	37.1	37.1	37.1	
BL @ CUT 2	117.0	117.0	117.0	
BL @ CUT 3	190.6	190.6	212.0	
BL @ CUT 4	242.6-	242.6-	247.5-	Main Land. Gr. Pick up
BL @ CUT 5	242.6+	242.6+	247.5+	
BL @ CUT 6	291.2	291.2	341.7	
BL @ CUT 7	339.8	339.8	404.2	
BL @ CUT 8	387.5	387.5	465.6	
BL @ CUT 9	418.0-	471.2-	486.5-	Wing Eng/ Pylon Pick up
BL @ CUT 10	418.0+	471.2+	486.5+	
BL @ CUT 11	453.4	550.4	550.4	
BL @ CUT 12	500.8	611.4	611.4	
BL @ CUT 13	567.3	697.0	697.0	
BL @ CUT 14	633.8	782.4	782.4	
BL @ CUT 15	700.2	868.0	868.0	
BL @ CUT 16	788.8	982.0	982.0	
BL @ CUT 17	833.0	1038.9	1038.9	
BL @ CUT 18	877.3	1095.9	1095.9	
BL @ CUT 19	921.6	1152.7	1152.7	
BL @ CUT 20	930.0	1163.7	1163.7	

3.1.4 Design speed and load factors -

3.1.4.1 Design speed altitudes - The design speed altitudes used for the Baseline and AR12 airplanes are the same. The maximum operational ceiling is 40,000 feet. VD/MD is defined as the lesser of 435 CAS or mach = 0.95. VC/MC is defined as the lesser of 365 KCAS or mach = 0.90. The midcruise point is defined as a velocity of 360 KEAS at mach = 0.80.

Definitions of VD/MD and VC/MC are in accordance with FAR-25 regulations.

3.1.4.2 Flight maneuver design load factors - Design maneuver load factors used for the Baseline and AR12 analyses comply with FAR-25 requirements.

For the basic clean airplane configuration, the maximum positive maneuver load factor is 2.5 gs, with the maximum negative load factor varying linearly from 0.0 gs at VD to -1.0 gs at VC. The maximum design load factors at VA are +2.5 and -1.0 gs.

For the flap extended airplane configuration the maximum positive load factor is 2.0 gs, and the maximum negative load factor is 0.0 gs.

A typical design speed envelope for the baseline airplane is shown in figure 3.1-1.

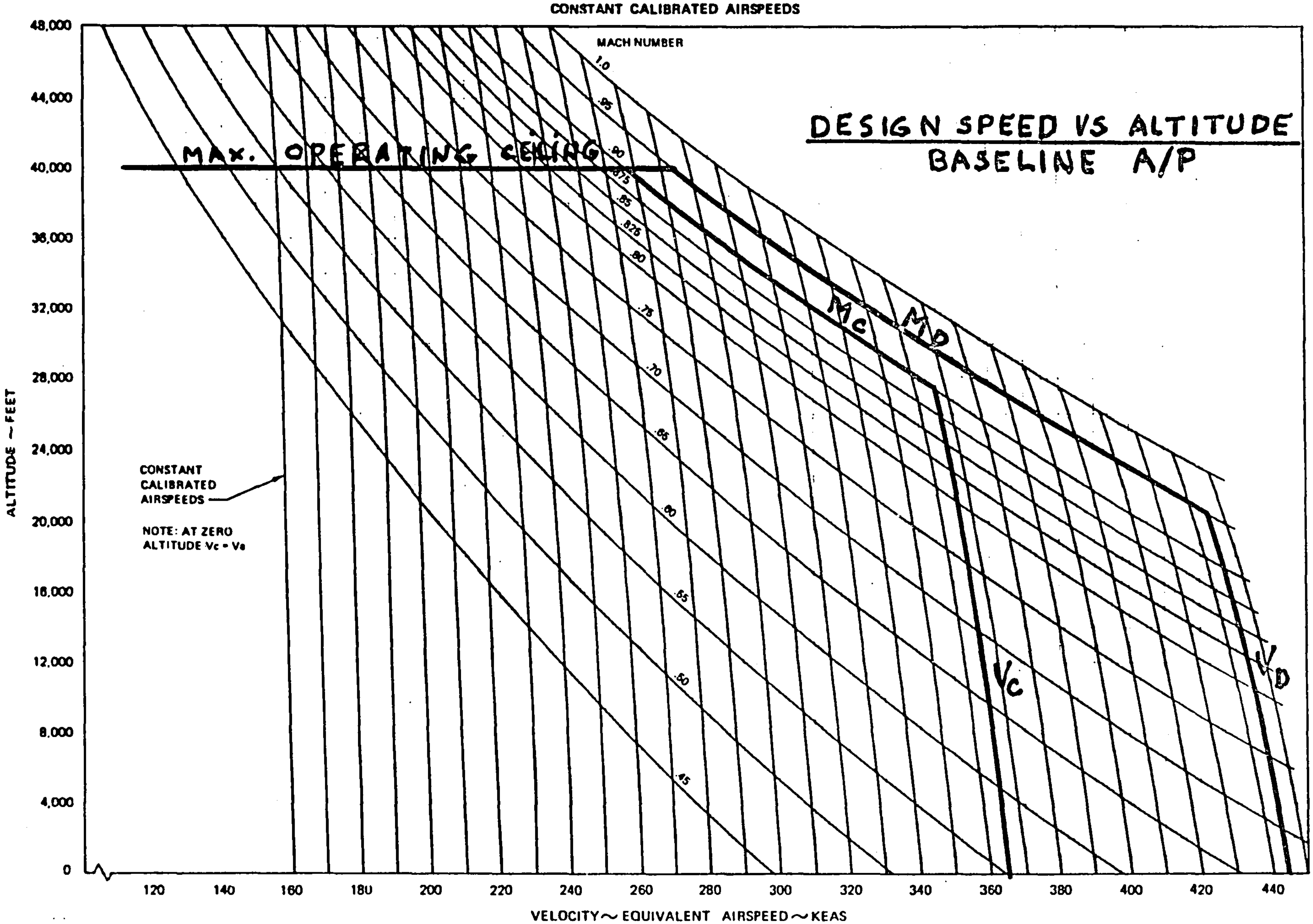


FIGURE 3.1-1 Design Speed Envelope

3.2 Aerodynamics

3.2.1 Wing design introduction - This study addresses the optimization of wings for transport aircraft. Optimization for maximum high subsonic or transonic cruise is considered. The emphasis is on structural design and fuel efficiency. The aerodynamic configuration is taken as constant for a first approximation. Development of the aerodynamic configuration is discussed in this section. Geometry and airfoil selection are reviewed first. Then airfoil design methods used are explained briefly. Next the transfer of aerodynamic design data to the structural design optimization is covered. Finally, the question of how perturbations to the baseline geometry effect the aerodynamic performance is considered.

3.2.2 Geometry Selection. - Life cycle cost is the driving factor in wing geometry selection for transport aircraft. A major factor is aspect ratio. High aspect ratio is important for attaining good aerodynamic efficiency. However high aspect ratio wings tend to be heavier and thus have a higher initial cost. The trade-off is lower fuel consumption and hence lower operating cost.

Given a desired type of operation, such as payload, operating altitude and range; a wing can be designed for the lowest life cycle cost. The resulting aerodynamic configuration is the input to the structural optimization procedure explained in this report. The following information is supplied:

1) Airfoil type and nondimensional ordinates.

2) Wing incidence distribution.

3) Operating mach number.

4) Operating lift coefficient.

5) Operating Reynolds number.

Aspects of airfoil selection are discussed in the following subsections.

3.2.3 Airfoil type selection. - For transport aircraft the available airfoil choices can be grouped into two categories. They are:

1) Peaky - Conventional.

2) Advanced Technology - Supercritical.

The peaky technology airfoils represent an earlier and mature technology. These airfoils are characterized by a rapid development of the upper surface pressure distribution suction at the leading edge. There is a gradual recompression to the trailing edge. These airfoils have a relatively low pitching moment about the quarter chord point. Consequently the total configuration will have minimal trim drag.

The advanced technology - supercritical airfoils represent a newer and developing type. Higher efficiency is possible with these airfoils compared to the conventional technology. The design for a given application using advanced technology airfoils is exacting. More of the chord length is made to contribute to the section lift than is the case with the conventional technology airfoils. This is done by using a rooftop upper surface pressure distribution. As much of the upper surface is operated slightly supersonic as is possible. Limiting factors are geometry constraints and the ability to recompress the upper surface flow at the trailing edge without flow separation. The airfoils are highly aft-chord loaded, and consequently they exhibit a high nose-down pitching moment. Additional tail loads and resultant trim drag are needed to counter this pitching moment. However the airfoil section drag reduction can be substantial, and thus more than makeup for the trim drag penalty. Additional discussion on conventional and supercritical technology airfoils may be found in reference 8.

For the PADS wing design study, a conventional type airfoil was selected because currrent analysis procedures for supercritical wings are not as mature as for conventional wings and sufficient data to benchmark the analysis techniques are not available. Also, the baseline airplane has a conventional airfoil.

3.2.4 Airfoil design - The range factor, M(L/D), is the most significant figure of merit for assessing the aerodynamics of wings designed for cruise efficiency. The importance of the range factor is demonstrated by the specific-air-range (miles flown per pound of fuel consumed) formula.

$$SAR = (a/SFC) * M(L/D) * (1/GW)$$

This formula also points out the impact of gross weight, GW, and propulsion specific fuel consumption divided by the speed of sound (SFC/a).

Two airfoil design methods have been found to work well in the wing three-dimensional flow environment. They are:

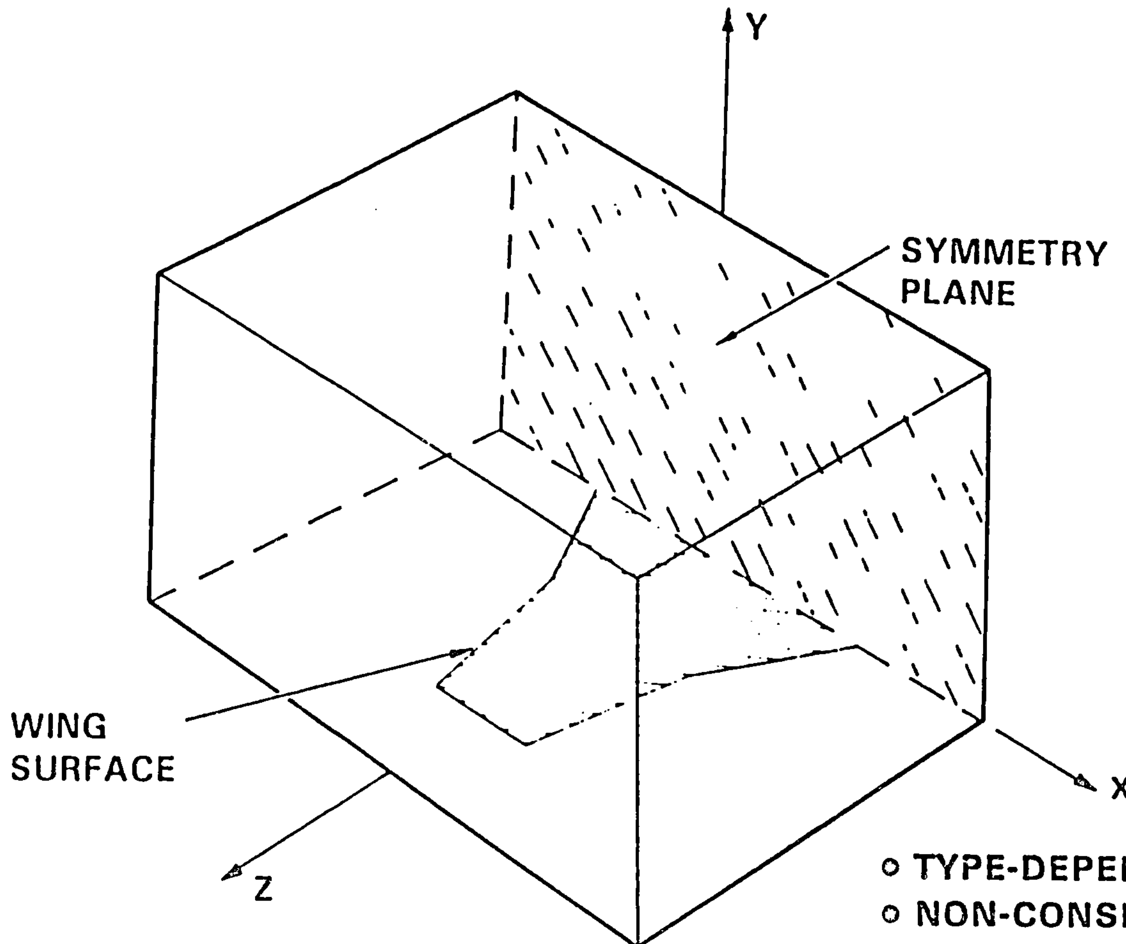
1) Design-by-iteration

2) Perturbation redesign.

The design-by-iteration process is a coupling of an airfoil ordinate generator and a wing pressure distribution analysis program. Iterations are made to the airfoils, the configuration is analyzed, and the resulting pressure distribution changes are used as a guide for further iterations. Engineering experience and intuition are important ingredients in making this method work successfully. Interactive computer graphics is used to configure the airfoils.

The FLO-22 code is used to compute the pressure distributions. This code solves a nonconservative finite-difference approximation to the potential equation for an isolated wing. The wing and computational domain are shown in figure 3.2-1. Flow equations are solved in a set of sheared parabolic coordinates. The mapping to obtain these coordinates is shown in figure 3.2-2. FLO-22 is a cost effective analysis tool.

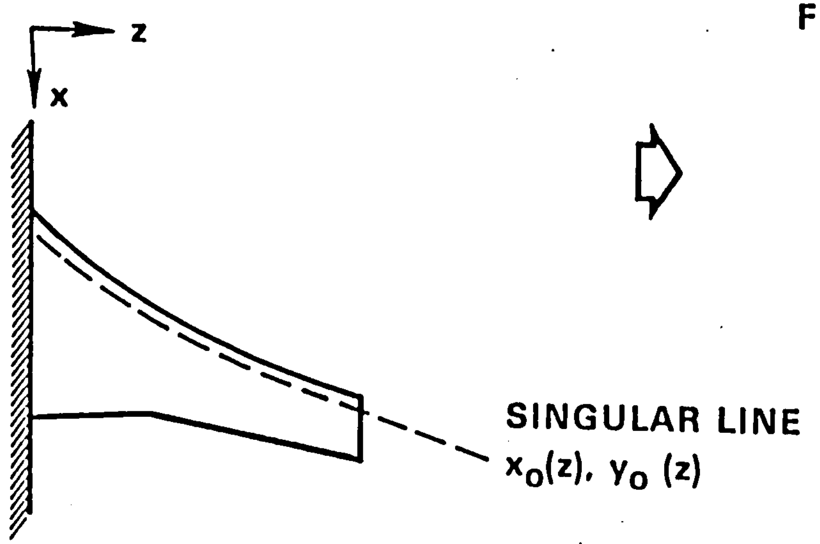
The second design procedure mentioned, perturbation redesign, is a design option added to the FLO-22 code. Being a perturbation procedure, some starting point design is required. Typically it comes from the first method. Given a desired target pressure distribution, the starting geometry is perturbed by the program. The amount of change is directly related to the difference between the initial pressure distribution and the desired or target pressure distribution. If the pressure distribution computed for the perturbed geometry differs from the target distribution, then the process is repeated until convergence is achieved. The iterative procedure is schematically shown in figure 3.2-3.



- TYPE-DEPENDENT FINITE DIFFERENCE
- NON-CONSERVATIVE.
- SOLUTION BY RELAXATION (SLOR)

FIGURE 3.2-1 Computational Domain

JAMESON-CAUGHEY FULL-POTENTIAL FINITE-DIFFERENCE CODE
FLO 22



$G = \phi - x \cos \alpha - y \sin \alpha$

- SHEARED PARABOLIC COORDINATES
- TYPE DEPENDENT FINITE DIFFERENCE
- NON-CONSERVATIVE
- SOLUTION BY RELAXATION

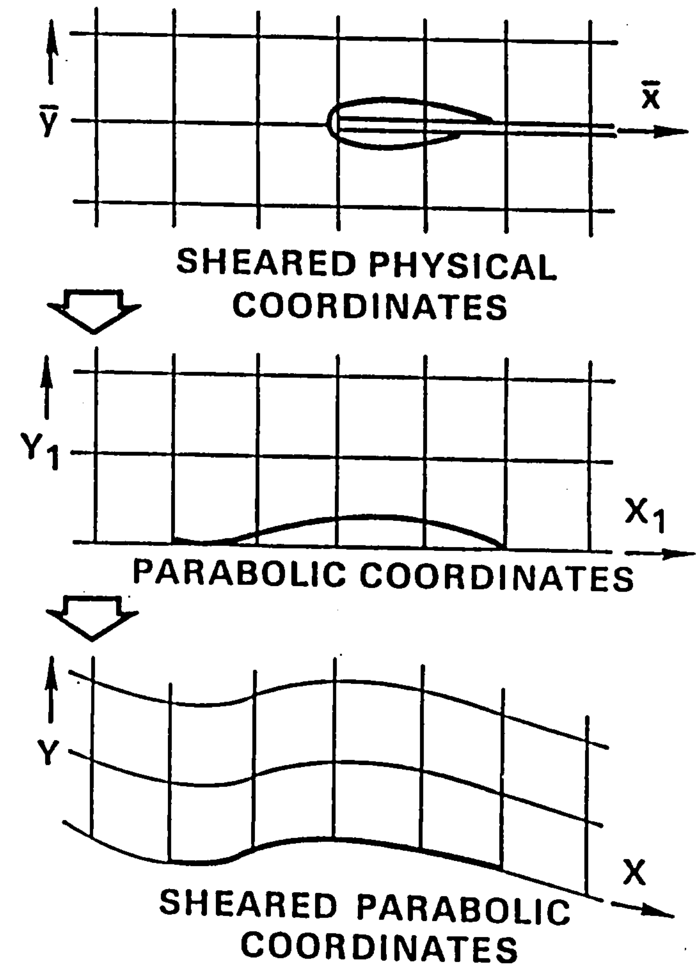


FIGURE 3.2-2 Sheared Parabolic Coordinates for Flo-22

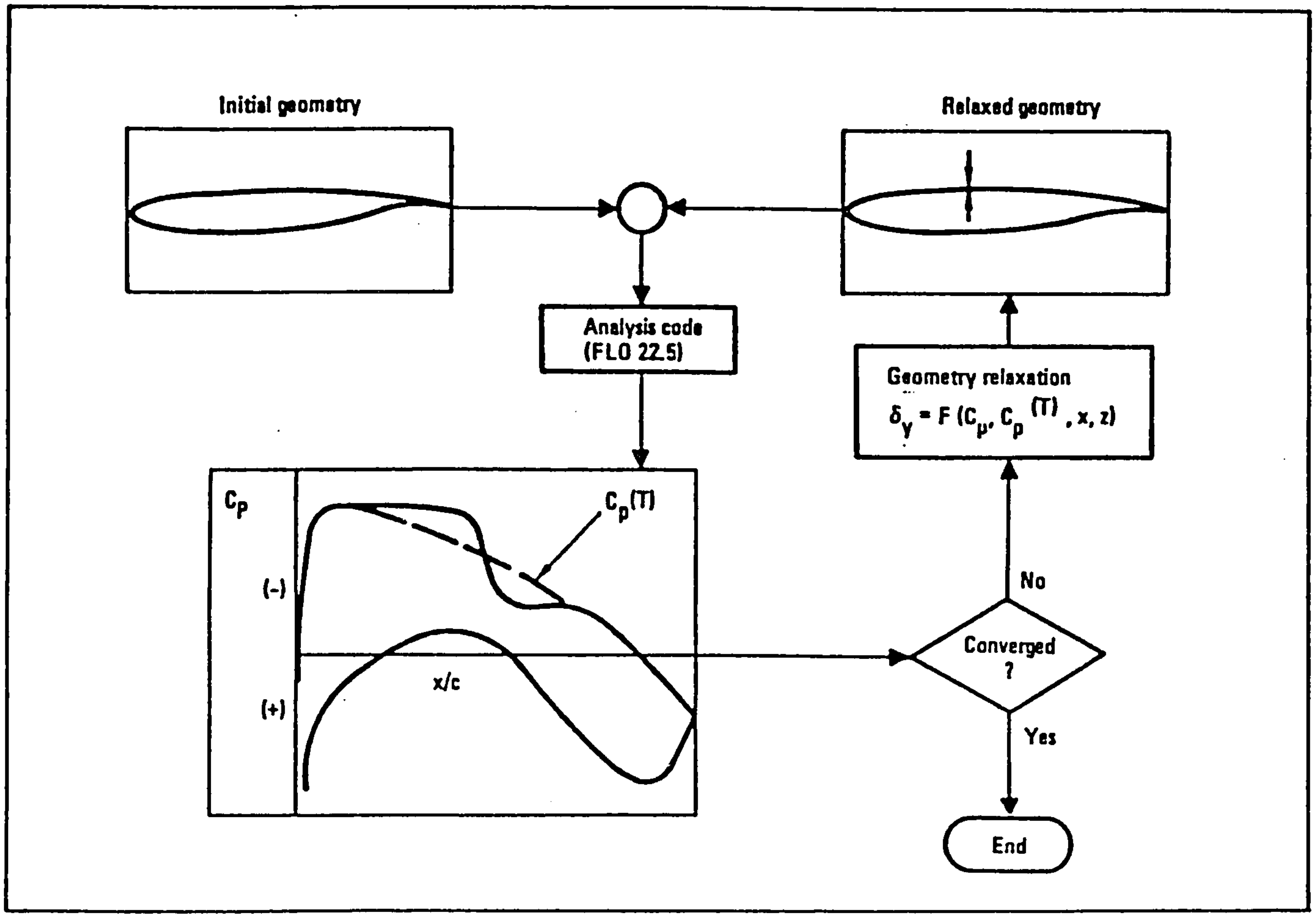


FIGURE 3.2-3 Perturbation Redesign by Geometry Relaxation

3.2.5 Wing airfoil definition - Wing designs used as baseline configurations for the structural optimization procedures are described in terms of FLO-22 input data. FLO-22 input data consists of air flow and geometry parameters. Only the geometry portion is used in the PADS design process. Location of the airfoil streamwise sections are specified. The airfoil section ordinates can be directly specified, or alternately, the parameters for an analytic airfoil generation code can be specified. Use of the FLO-22 input data to form the finite element model is described in Section 3.5.1.3.

Normally airfoil sections for structural modeling are at locations and angles other then those specified by the FLO-22 data set. The required interpolation to obtain the desired cuts is done with SLICE. (Surface Lofting by Interpolated Cubic Elements). A set of spanwise cubic fits of the original data is interpolated to yield the desired new cut. This is shown in figure 3.2-4.

3.2.6 Validity of Perturbations to Baseline Wing Configuration - As a result of the structural optimization, there may be changes made to the original aerodynamic performance. Generally conventional technology airfoils do not radically change pressure distribution characteristics with small perturbations to the section geometry. Therefore resulting configuration for small parametric changes can still be considered viable. For larger parametric changes to section geometry, airfoil performance must be reevaluated.

3.2.7 Airfoil Definition for PADS Designs - The airfoil technology was held fixed during the PADS design cycles. In addition, the airfoil definition was derived from parametric equations which were consistent with ASSET aerodynamic performance definitions.

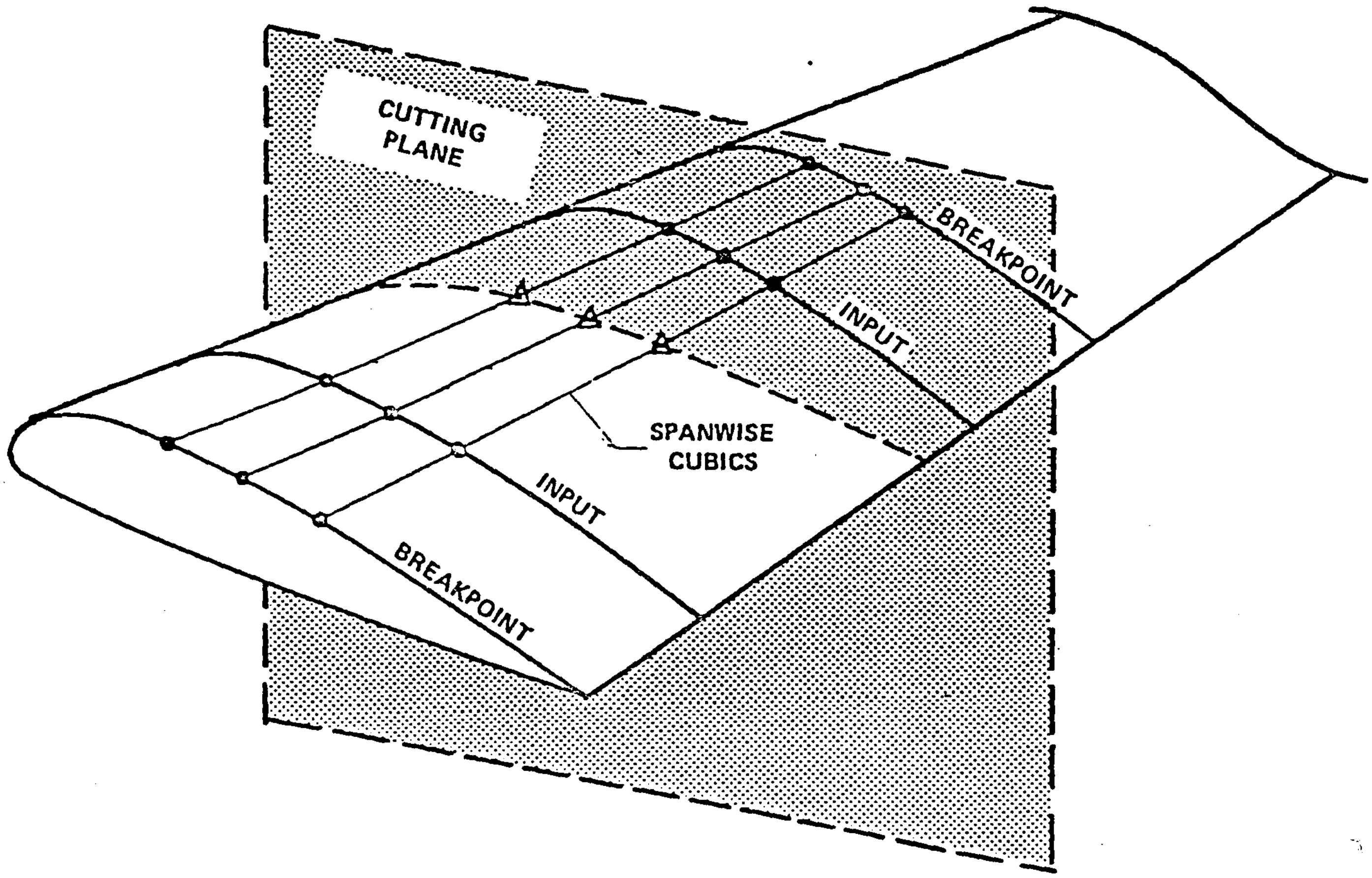


FIGURE 3.2-4 Schematic of Wing Slice

3.3 WEIGHT

3.3.1 ASSET weight estimation philosophy - Statistical weight estimating equations have been used for many years to derive aircraft component weights in the preliminary design phase. These equations have been developed primarily by using the least squares multiple regression mathematical techniques to get a best fit equation. This method results in an acceptable equation for estimating preliminary weights.

The major design variables which derive the structural weight have been determined and appropriate trends have been developed for wing, tail, body, landing gear and nacelles for purposes of estimating aircraft weight. The effect on a transport aircraft's empty weight for various types of stretching has also been explored. Validity of the trends and relationships has been substantiated through weight correlation plots for conventional transport aircraft. Weight trends versus design variables in relation to a standard wide-body transport were applied in the PADS study program.

The design variables directly affecting wing weight are:

- o Aircraft gross weight
- o Load factor
- o Stress level allowables
- o Taper ratio
- o Aspect ratio
- o Wing area
- o Thickness to chord ratio
- o Sweep
- o Bending relief (wing mounted concentrated weights, such as engines , etc., and wing fuel)

Correlation between estimated and actual weights for the wing weight equation used for the PADS program is shown in figure 3.3-1.

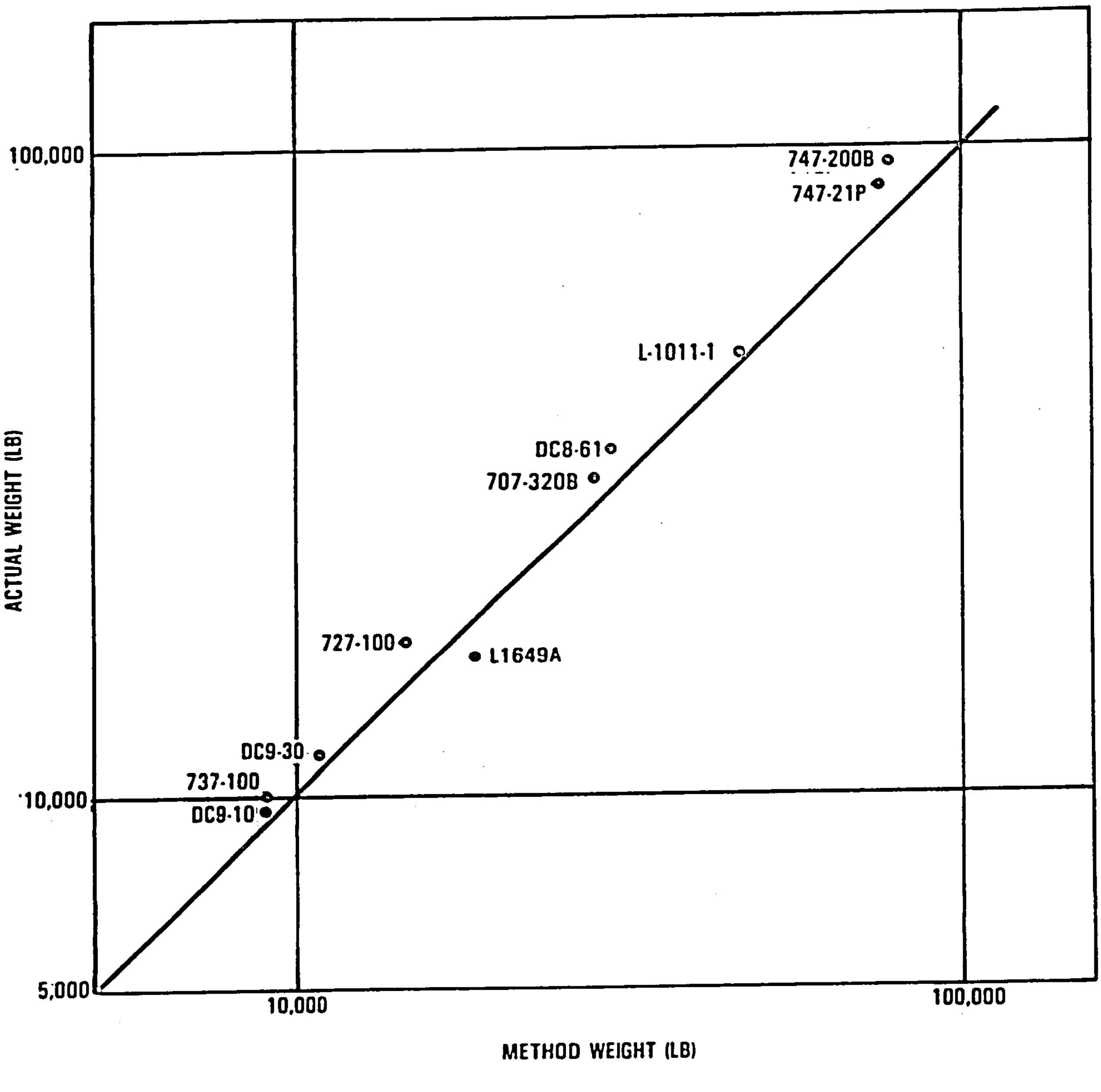


FIGURE 3.3-1. COMPARISON OF ASSET PREDICTED VS ACTUAL WING WEIGHT

3.3.2 Mass Distribution Module - Accurate mass distribution is necessary to perform loads, stress, and dynamic analyses. Specific data requirements include weight distribution by panel points, and an accurate representation of mass moment of inertia data, especially for aircraft components such as the engine pods, pylons, landing gears, and control surfaces.

To achieve an accurate representation of an aircraft's mass properties at a preliminary design level, analysis must begin at the drawing board. Mass properties analysis of an aircraft begins as early as the preliminary general arrangement drawing. The present basis for automated parametric vehicle sizing capability is the Lockheed developed Advanced Systems Synthesis and Evaluation Technique (ASSET) computer program. The output from these analyses would include a group weight statement, a weight and balance Summary, center-of-gravity travel analysis, and mass moment of inertia data.

In the past, mass matrices were generated manually, based upon the latest aircraft weight statement. This method has been improved to achieve the shorter design cycle times and reduced engineering man-hours associated with an automated, integrated design system.

A stand-alone mass distribution module (MDM) was developed independently of the PADS effort to generate data similar in detail to that provided for production analyses. Panel point data are presented in pounds and pound-inches squared (moments of inertia). The total airplane mass matrix for any desired loading consists of the addition of separate matrices for:

- o Wing and contents
- o Body and contents
- o Tail - horizontal and vertical
- o Landing gear - up or down
- o Engine/pods
- o Payload - passengers, cargo/baggage, stores
- o Fuel - quantities from empty to full

The Mass Distribution Module depends only upon a group weight statement generated by ASSET and geometry data from a general arrangement drawing for input. The output consists of mass and geometry matrices suitable for use in loads, flutter, and structural analysis. The mass distribution module is designed to fulfill the need for a realistic mass distribution early in the preliminary design phase when major structural arrangement decisions are made. It also ensures that structural and dynamic analyses are based on the same mass data.

The MDM is extremely useful in that it is flexible enough to handle a great variety of configurations. It accepts basic ASSET parametric output, interfaces with the other technologies through the creation of mass and inertia matrices, and can be easily revised for rapid response to weight data from NASTRAN or any other source.

3.3.2.1 Data flow diagram - A data flow diagram for the MDM is shown in figure 3.3-2. Functional groups are combined into the major components which comprise the operational empty weight.

Each of the major components or load items is distributed into appropriate panels based on typical distributions from comtemporary aircraft and preliminary basic loads. This distribution is accomplished by utilizing unity distributions for major components and their contents and multiplying each of them by predicted component and content weight. Typical input data are shown in table 3.3-1.

Five types of output are obtained from the Mass Distribution Module:

Inertial geometry matrix

Component and total aircraft summary matrices

Full inertial matrix (diagonal)

Load inertial matrices (6)

Printed table

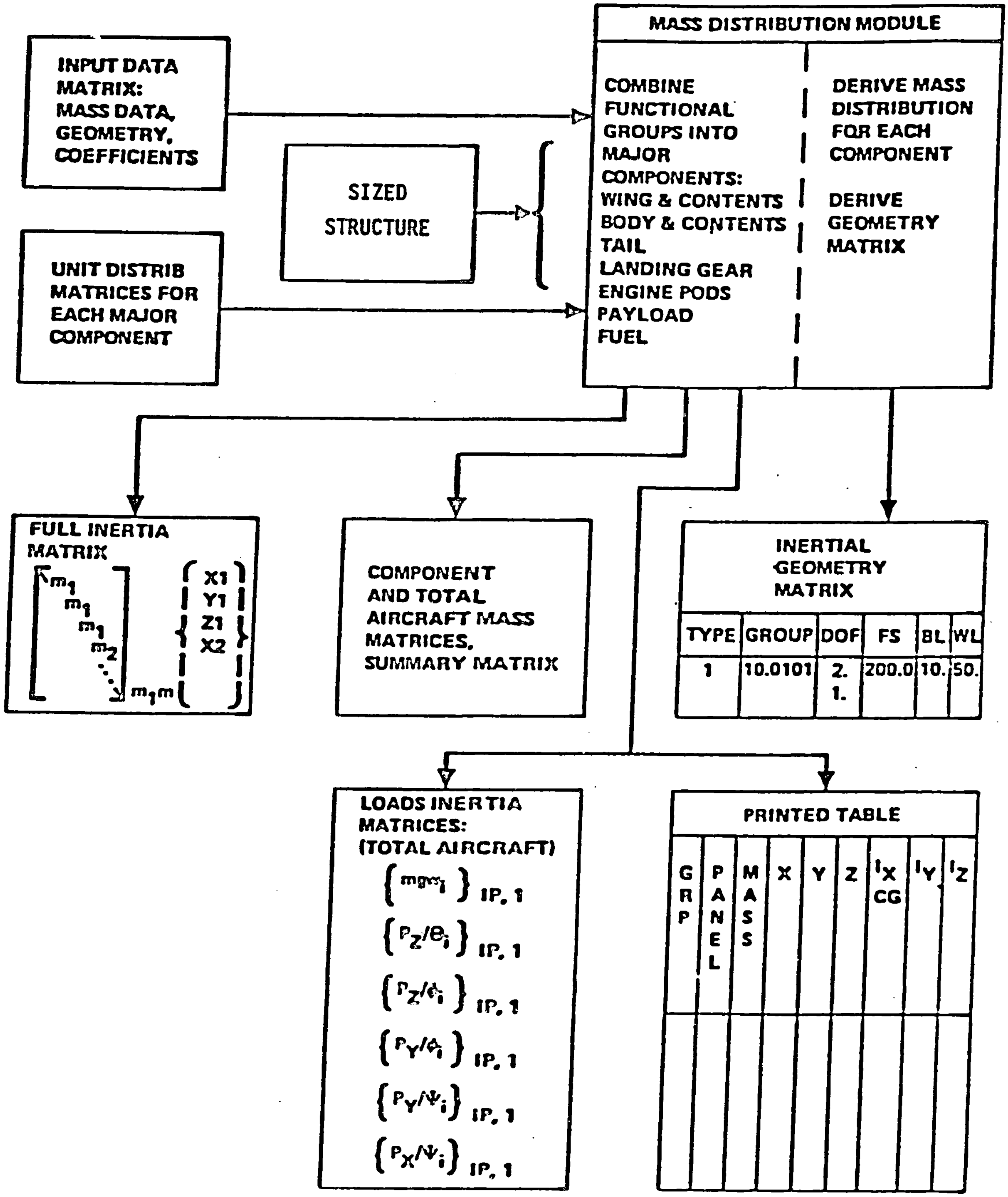


FIGURE 3.3-2 Mass Distribution Module Flow Diagram

TABLE 3.3-1 TYPICAL MASS DISTRIBUTION MODULE INPUT DATA

WFUS = 54762.
CHYDW = .2233576
CELECW = .186821
CECSW = .326966
CLGB = 0.
CICEB = .328982
CSCB = .6571702
WAPU = 1202.
WFURN = 21170.4
WINST = 998.
WAMO = 0.
WSK = 0.
WGUN = 0.
WWING = 75556.
CSCW = .3050576
WTAIL = 9042.
CSCHT1 = 0.
CSCHT2 = 0.
CSCVT1 = .037772
CSCVT2 = 0.
WES = 2449.
CPASS = 1.
WCARGO = 35330.6
WFA = 156276.

WHYD = 2740.
WELEC = 5797.
WECS = 5811.
WEC = 214.
WLG = 0.
WANICE = 383.
WSCE = 6248.
WAV = 2827.
WAG = 0.
WARM = 0.
WSTD = 8647.
WOPEQ = 8599.
WPYL = 0.
CICEW = .671018
WFS = 1518.
CHT1 = .803030
CHT2 = 0.
CVT1 = .196970
CVT2 = 0.
WPROP = 40963.
XNOENG = 3.
XNPASS = 59129.1
XKSTOR = 0.

The inertial geometry matrix (table 3.3-2) gives the location of each inertial panel point and contains all the information required to create transformation matrices. All three types of analyses, static loads, dynamic loads, and flutter depend on the same inertial data. Table 3.3-3 shows the correlation between the airplane group number (column 2 of the geometry matrix) and the airplane component which the group number identifies. In this way, each panel point mass is associated to a component of the airplane.

Typical form of the full inertia matrix is shown in table 3.3-4. Generally, each mass in the full inertia matrix has three degrees of freedom (x, y, z). Concentrated masses representing airplane components such as landing gear or engines at a single location have three additional degrees of freedom (Ix, Iy, Iz).

The loads inertia matrices are in the same form as presently supplied for the L-1011 and S-3A projects. Format of the printed table output is shown on the flow diagram, figure 3.3-2.

DESCRIPTION OF THE COLUMNS OF A GEOMETRY MATRIX						
TYPE MATRIX	AIRPLANE GROUP	DEGREE OF FREEDOM	LOCATION IN SPACE			...
			FS	BL	WL	

	1	2	3	4	5	6	...
1	1.	10.0101	2.	20.	30.	40.	
2	1.	10.0101	1.	20.	30.	40.	
⋮	EXAMPLE						

MATRIX SIZE = m x n

$m \leq 1000$

$n \leq 20$, AT PRESENT $n = 6$

TABLE 3.3-2 Format of Inertia Geometry Matrix

The number in column 2 is composed of 6 digits.

AIRPLANE GROUP	
Column 2 of Geometry Matrices	DESCRIPTION
10.00 CO	WING
10.01 yy	BOX
10.02 yy	FIN
10.03 yy	TE CONTROL SURFACE
10.04 yy	LE CONTROL SURFACE
10.05 yy	SPOILER
10.06 yy	ENGINE
10.07 yy	STORE
10.08 yy	LANDING GEAR
10.XX yy	
20.00 00	FUSELAGE
20.01 yy	BOX
20.02 yy	CONTROL SURFACE
20.03 yy	ENGINE
20.04 yy	STORE
20.05 yy	LANDING GEAR
20.XX yy	
30.00 00	HORIZONTAL STABILIZER
30.01 yy	BOX
30.02 yy	ELEVATOR
30.XX yy	
40.00 CO	VERTICAL TAIL
40.01 yy	BOX
40.02 yy	RUDDER
40.XX yy	

An example of yy are the TE control surfaces.

The 1st surface would be identified as 10.0301.
The 2nd surface as 10.0302, etc.

TABLE 3.3-3 Airplane Group

Summary Matrix:

$$\begin{Bmatrix} MT\emptyset \\ XTO \\ YTO \\ ZTO \\ IXCGTO \\ IYCGTO \\ IZCGTO \\ IXZTO \\ PATO \end{Bmatrix}_{9,1}$$

Full Inertia Matrix:

$$\begin{bmatrix} m1 & & & & & & & & & & & \\ & m1 & & & & & & & & & & \\ & & m1 & & & & & & & & & \\ & & & m2 & & & & & & & & \\ & & & & \ddots & & & & & & & \\ & & & & & meng & & & & & & \\ & & & & & & meng & & & & & \\ & & & & & & & meng & & & & \\ & & & & & & & & I_x eng & & & \\ & & & & & & & & & I_y eng & & \\ & & & & & & & & & & I_z eng & \\ & & & & & & & & & & & \ddots \end{bmatrix} \begin{Bmatrix} x1 \\ y1 \\ z1 \\ x2 \\ \vdots \\ xeng \\ yeng \\ zeng \\ \phi eng \\ \theta eng \\ \Psi eng \\ \vdots \end{Bmatrix}$$

TABLE 3.3-4 Summary and Full Inertia Matrices

3.3.2.2 Fuel tank loading procedure - A program called TANK is used to compute fuel tank weight distributions. TANK receives input data defining desired flight orientations and fuel loadings and constructs a lumped fuel distribution which reflects the flight configurations necessary for design. TANK computes definition of the tank boundaries from airfoil definition contained in the grid cards for the finite element model and corner point inputs. With tank boundaries known, the fuel tank total volume can be computed. Basic data input such as fuel weight, allow TANK to form elementary fuel boxes and distribute the fuel weight as lumped fuel masses to the finite element grid locations. Balance and center-of-gravity data, as well as the fuel distribution, are then supplied as output.

TANK is used several times for aircraft with multiple tanks. Then, a postprocessor combines the various fuel tank weight distributions and performs a transformation to the weight panel points. The fuel distribution at the weight panel points is used as direct input to MDM.

3.3.3 Initial distribution of ASSET estimated Baseline aircraft. -

ASSET was used to derive parametrically the weight of the Baseline aircraft in a manner similar to that used in the estimation for advanced configurations. The ASSET weight statement is presented in table 3.3-5. Mass moment of inertia data were generated for the pylons, engines, and landing gears by hand. Coefficients were then developed to be used for the advanced configurations to scale inertia by geometry and weight.

Individual group weight as determined by ASSET was distributed to major aircraft components based on conventional transport distribution fractions.

Fuel weight distributions were formed as described in the previous section for input to the Mass Distribution Module.

Payload was distributed to meet specified weight and center-of-gravity requirements for the establishment of loads data. Maximum passenger compartment load criterion was used which requires a uniform loading of 300 pounds/foot above the floor to account for approximately an 80 percent passenger load. The remainder of the payload is block loaded above the floor at 350 pounds/foot and in the cargo compartments to maximum capacity from either or both extremities. A typical example of this loading configuration is presented in figure 3.3-3.

The design center-of-gravity envelope for the Baseline aircraft is presented in figure 3.3-4. The loading conditions used in the PADS study are presented in table 3.3-6.

This aircraft data was input to the Mass Distribution Module (MDM) as explained in the previous section. Weight was distributed to the panel points. The printed table output and full inertia output matrix are included in Appendix C.

Table 3.3-7 is the table of contents for the MDM output matrices. The first panel point and the total number of panels used for each component are shown, as well the panel degrees of freedom. The column labeled "FIRST ELEM LOCATION" presents the beginning of each component's weight and inertia data in the full inertia output matrix.

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(504000.)		
FUEL AVAILABLE	166001.	FUEL	32.94
EXTERNAL	0.		
INTERNAL	165998.		
ZERO FUEL WEIGHT	337999.		
PAYLOAD	85963.	PAYLOAD	17.06
PASSENGERS	42500.		
BAGGAGE	13750.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	252036.		
OPERATIONAL ITEMS	8599.	OPERATIONAL ITEMS	3.42
STANDARD ITEMS	8662.		
EMPTY WEIGHT	234775.		
STRUCTURE	138639.	STRUCTURE	27.51
WING	50087.		
ROTOR	0.		
TAIL	9042.		
BODY	48353.		
ALIGHTING GEAR	21473.		
ENGINE SECTION AND NACELLE	9684.		
PROPULSION	40552.	PROPULSION	8.05
CRUISE ENGINES	32693.		
LIFT ENGINES	0.		
THRUST REVERSER	4923.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	214.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2184.		
DRIVE SYSTEM	0.		
SYSTEMS	55586.		
FLIGHT CONTROLS	6220.		
AUXILIARY POWER PLANT	1202.		
INSTUMENTS	934.		
HYDRAULIC AND PNEUMATIC	2663.		
ELECTRICAL	5308.		
AVIONICS	2827.	SYSTEMS	11.03
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	30267.		
AIR-CONDITIONING	5811.		
ANTI-ICING	354.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

TABLE 3.3-5 BASELINE ASSET WEIGHT STATEMENT

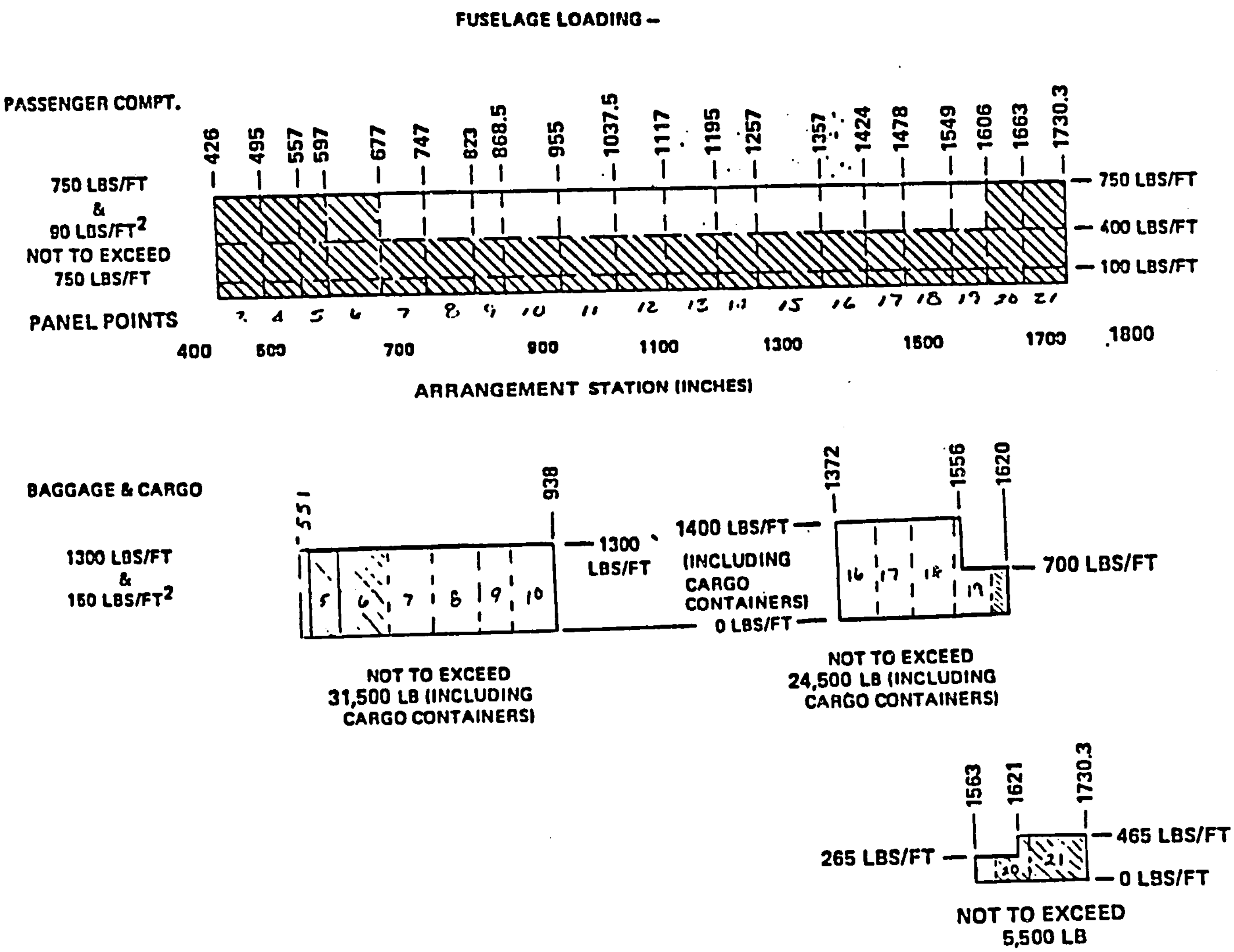


FIGURE 3.3-3 Fuselage Loading

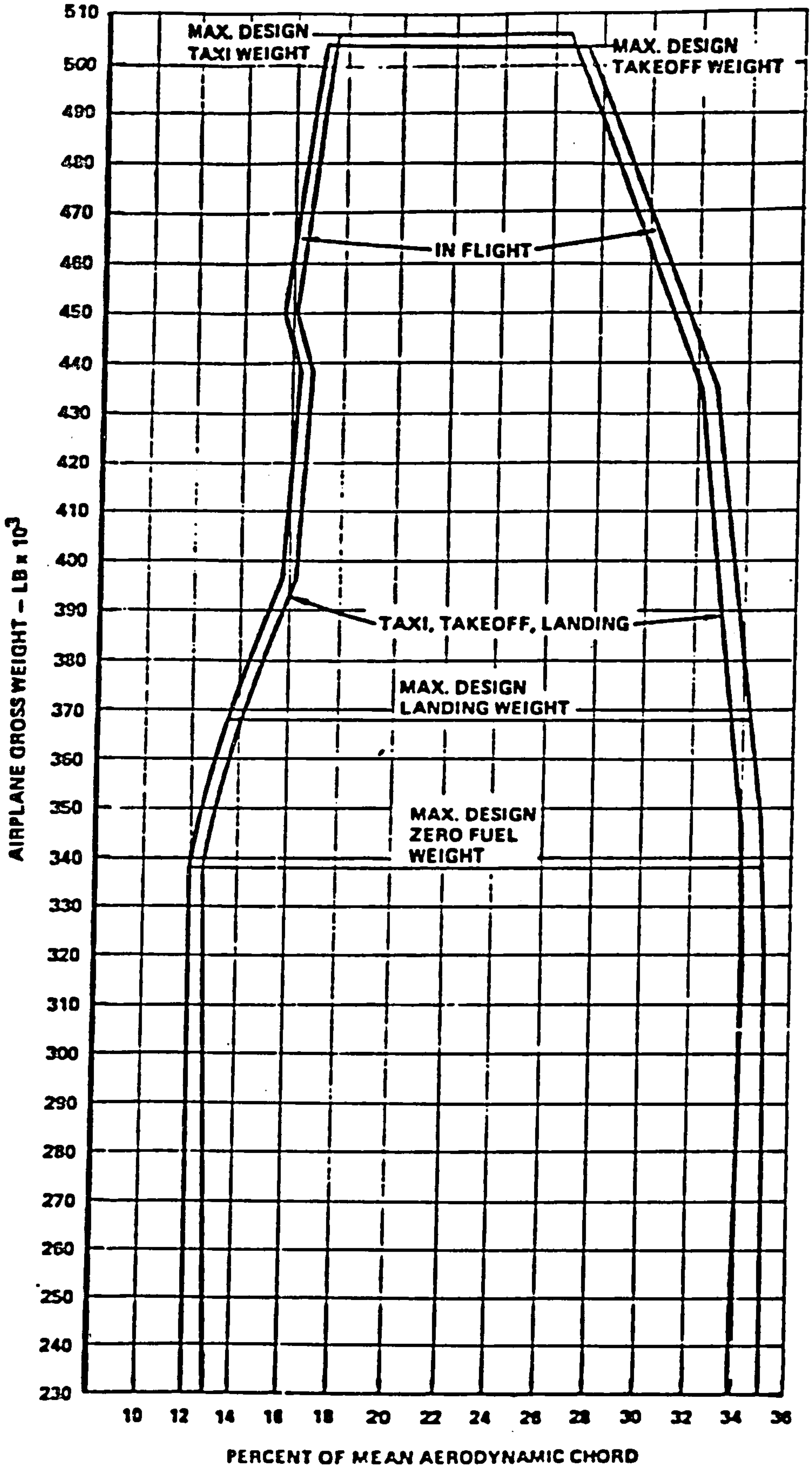


FIGURE 3.3-4 Design Center-of-Gravity Envelope

Table 3.3-6 Inertia Conditions for Baseline Design

Condition	Name	Description
1	Full fuel	Gear up, fwd cg limit, max takeoff weight, 166,000 lb. fuel
2	Low fuel	Gear up, fwd cg limit, 12,300 lb. fuel
3	WDA-963	Gear down, fwd cg limit, max landing wt (368k), 30,000 lb. fuel
4	WDA-965	Gear down, aft cg limit, max taxi wt.

Aircraft Loading Condition
(all weight shown in pounds)

Condition	1	2	3	4
Op. empty wt.	252037.5	252037.5	242940.3	242940.3
Payload	85961.1	85961.1	95055.7	95059.6
Zero fuel wt.	337998.6	337998.6	337996.0	337999.9
Fuel wt.	166000.0	12299.5	30000.6	167998.3
Gross wt.	503998.6	350298.1	367996.6	505998.2
XCG	1193.6	1182.6	1184.4	1222.0
YCG (half a/p)	188.0	136.6	149.7	195.0
ZCG	193.0	192.2	193.8	190.6

TABLE 3.3-7 Table of Contents for Output Matrices

TABLE OF CONTENTS FOR OUTPUT MATRICES

COMPONENT NUMBER	COMPONENT TITLE	TYPE	FIRST PANEL	PANEL COUNT	DOF	FIRST ELEM LOCATION
1	**BODY	1	1	25	3	1
2	**CENTER BOX STRUCTURE **	13	11	4	3	31
3	**CENTER BOX SYSTEMS**	13	11	4	3	31
4	**WING BOX STRUCTURE**	2	26	150	3	76
5	**WING BOX SYSTEMS**	2	26	150	3	76
6	**LEADING EDGE**	2	26	150	3	76
7	**TRAILING EDGE**	2	26	150	3	76
8	**INBOARD AILERON**	2	176	150	3	526
9	**OUTBOARD AILERON**	2	386	150	3	976
10	**OUTBOARD FLAP**	2	476	150	3	1426
11	**INBOARD FLAP**	2	799	150	3	1876
12	**INBOARD SLAT**	2	949	150	3	2326
13	**OUTBOARD SLAT**	2	1099	150	3	2776
14	**HORIONTAL TAIL**	3	626	80	3	3226
15	**VERTICAL TAIL**	5	706	80	3	3466
16	**S-DUCT**	5	786	6	3	3706
17	**NOSE GEAR UP**	8	792	1	6	3724
18	**MAIN GEAR UP**	8	794	1	6	3730
19	**NOSE GEAR DOWN**	8	793	1	6	3636
20	**MAIN GEAR DOWN**	8	795	1	6	3742
21	**ENGINES**	8	797	2	6	3748
22	**PYLON**	8	796	1	6	3760
23	**PASSENGERS**	10	3	19	3	7
24	**CARGO AND BAGGAGE**	11	4	18	3	10
25	**FUEL**	12	12	139	3	34

3.3.4 Calculation of weight utilizing NASTRAN - The NASTRAN program has the capability of computing the total weight, center of gravity, and moment of inertia for a specified set of structural elements. This has the potential advantage of faster weight and inertia determination. When several different structural representations are under consideration, this ability has significant impact on the design cycle. These data are used to form a weight data update loop to modify appropriate elements of the Mass Distribution Module (MDM). Care must be exercised in defining the weight sets to assure that there is no duplication or exclusion of any structural members.

3.3.4.1 Elements - A grid system is set up to model the structure, and individual elements are used between the grid points to represent the structure. Some of the elements used are described in table 3.3-8.

A spar web, as shown in figure 3.3-5, is selected to illustrate the bulk data deck for the NASTRAN model. The bulk data for the spar web is shown in figure 3.3-6. Line 1 identifies this web as a shear element, assigns an identification (ID) number to it, and defines the connecting grid points. The aircraft three-axis coordinates are listed under line 2. The material ID number and panel thickness are given in line 3. The material properties are presented in line 4. In a similar fashion, each structural element is specified.

3.3.4.2 NASTRAN weight computation example - NASTRAN computes the weight for a finite element model with a routine called Grid Point Weight Generator, (GPWG). The spar web of figure 3.3-5 is used as an example of the GPWG subroutine logic. The process is repeated for each of the elements specified in a set and then the total weight, c.g., and moment of inertia for the set are calculated.

TABLE 3.3-8 NASTRAN Elements for Weight Calculations

ELEMENT	LOAD CAPABILITY	APPLICATION	CARD ID: CONNECTING GRID POINTS	CARD ID: PROPERTIES
Bar	Bending	Frames	CBAR	PBAR
Quadri-lateral	Inplane	Honeycomb Panels	CQDMEM	PQDMEM
Rod	Axial	Spar Caps	CROD	PROD
Shear	Shear	Webs	CSHEAR	PSHEAR
Triangular	Inplane	Honeycomb Panels	CTRMEM	PTRMEM

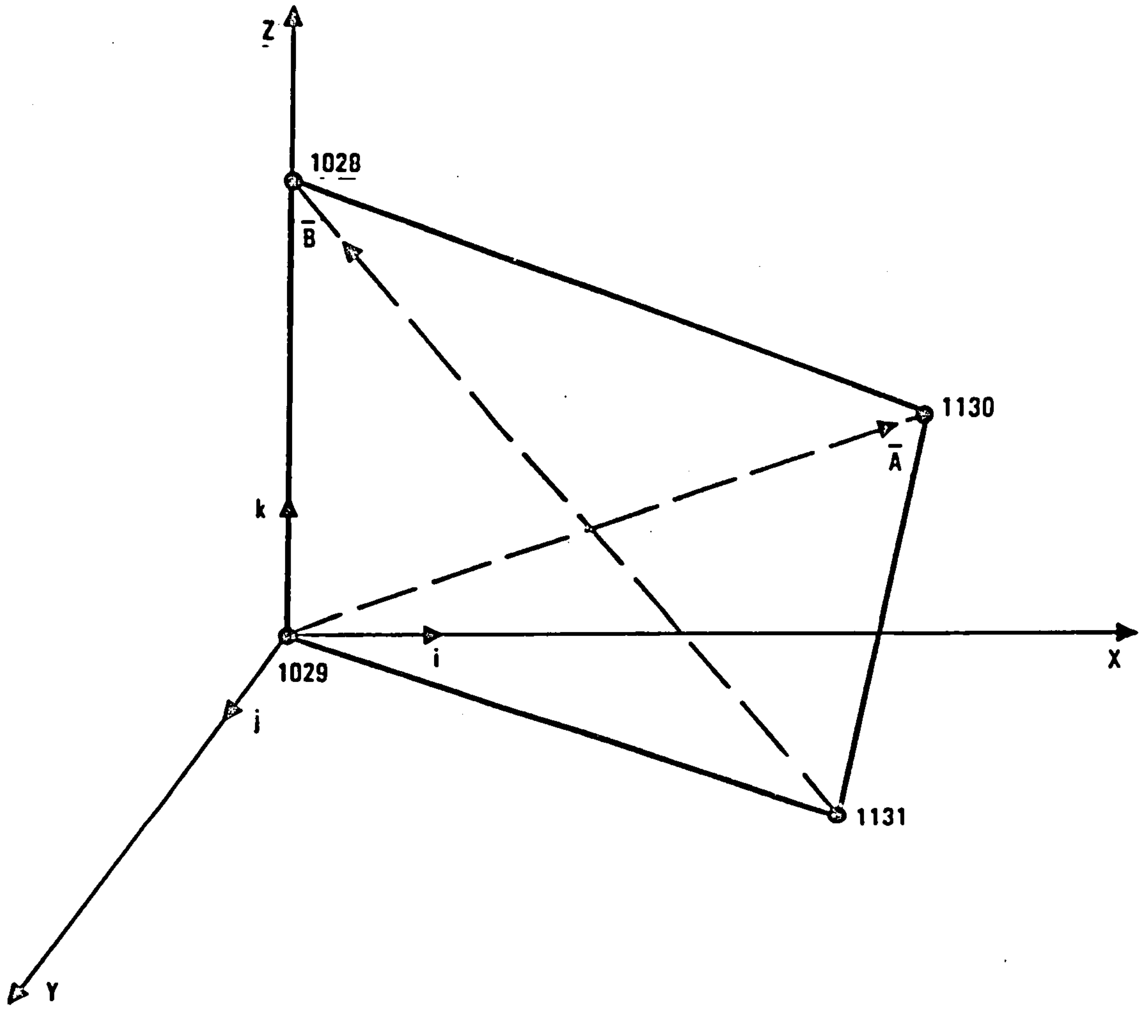


FIGURE 3.3-5. SPAR RIB

1) Shear Panel Element Connection Cards

	Element Id	Property Id	Connecting Grid Point Id No.			
CSHEAR	50044	50044	944	1044	1045	945
CSHEAR	50044	50044	944	1044	1045	945

2) Grid Cards

	Grid Id	X	Y	Z	Permenant Single Pt. Constraint
GRID	946	2868.0	365.0	326.0	456
GRID	1028	2145.0	402.0	282.0	6
GRID	1029	2145.0	402.0	268.0	456
GRID	1030	2235.0	406.0	241.0	456
GRID	1045	2821.5	406.0	322.5	456
GRID	1046	2821.5	406.0	331.0	456
GRID	1130	2235.5	435.0	285.5	6
GRID	1131	2235.5	435.0	273.0	456
GRID	1152	2417.5	438.0	317.0	

3) Shear Panel Property Cards

	Prop. Id	Material Id	Thickness
PSHEAR	50044	501	.015
PSHEAR	51028	501	.050
PSHEAR	51030	501	.027

4) Material Cards

	Mat. Id	E	G	Nu	Density	Texpand	Tref
MAT1	501	16.0E6	6.10E6	.31	.16	5.30E-6	70.0

FIGURE 3.3-6 Spar Rib Bulk Data

3.3.4.3 Input set definition and GPWG output - Generally the set definition consists of lumping similar elements, covers, ribs, webs, etc. together. These may also be subdivided into location, such as outboard, mid, and inboard sections. Figure 3.3-7 illustrates the input for several sets of sample spar webs. The first field identifies the input card DELK, followed by an assigned set number in the next field. Fields 3 and 4 specify the starting and stopping element ID numbers. The fifth field indicates the increment applied to the element ID, for inclusion in the set. The example element 51028 is seen to be the first included in the set number 510, which also includes element numbers 51030, 51032, 51034, 51036, 51038, 51040, and 51130.

The output for each set is typically illustrated in table 3.3-9 for set 510. The first matrix contains most of the weight and inertia data in the arrangement shown in figure 3.3-8.

If the structural model weight consists of only element weights, the three weights (Wx = Wy = Wz) printed out will be equal and the c.g. will be unique.

3.3.4.4 Mass properties comparison - A hand calculation was made of the sample test set 510 spar webs, and the total is compared with the output of table 3.3-9. These data are summarized in table 3.3-10.

The two sets of data are within 1 to 2 percent of each other. The GPWG reduces the tedious recalculation of elements as they are resized.

It must be emphasized that these element weights are for the NASTRAN represented structural components. Weight due to items such as joints, fasteners, tolerances, and discontinuities are not included. Weight allowances for these items must be derived and added to the NASTRAN weight before comparisons with the MDM are made.

FIGURE 3.3-7 NASTRAN DELK INPUT - SPAR WEB

	DELK Set Id	Element Id 1	Element Id 2	Incr.
DELK	501	50112	50140	2
DELK	502	50212	50240	2
DELK	502	50312	50340	2
DELK	504	50116	50440	2
DELK	504	50518	50540	2
DELK	504	50620		
DELK	507	50722	50740	2
DELK	507	50926	50940	2
DELK	507	50824		
DELK	510	51028	51040	2
DELK	510	51130		
DELK	513	51300	51340	2
DELK	513	51344	51352	2
DELK	513	51236	51240	2
DELK	513	51232		
DELK	513	51360		
DELK	515	51502	51682	20
DELK	515	51504	51704	20
DELK	515	51506	51706	20
DELK	515	51508	51708	20
DELK	515	51748		

M O - RIGID BODY MASS MATRIX IN BASIC COORDINATE SYSTEM

6.676253E 01	0.0	0.0	0.0	1.974553E 04	-2.906565E 04
0.0	6.676253E 01	0.0	-1.974553E 04	0.0	1.621928E 05
0.0	0.0	6.676253E 01	2.906565E 04	-1.621928E 05	0.0
0.0	-1.974553E 04	2.906565E 04	1.856976E 07	-7.067066E 07	-4.812538E 07
1.974553E 04	0.0	-1.621928E 05	-7.067064E 07	4.018258E 08	-8.598170E 06
-2.905565E04	1.621928E 05	0.0	-4.812534E 07	-8.598170E 06	4.036802E 08

S - TRANSFORMATION MATRIX FOR SCALAR MASS PARTITION

1.000000E 0.0	0.0	0.0
0.0	1.000000E	0.0
0.0	0.0	1.000000E 00

DIRECTION MASS AXIS SYSTEM (S)	MASS	X-C.S.	Y-C.G.	Z-C.G.
X	6.676253E 01	0.0	4.353586E 02	2.957576E 02
Y	6.676253E 01	2.4293995 03	0.0	2.957576E 02
Z	6.676253E 01	2.429399E 03	4.353586E 02	0.0

I(S) - INERTIAS

7.589700E 04	5.862400E 04	1.556320E 05
5.862400E 04	1.955058E 06	1.787000E 03
1.556320E 05	1.787000E 03	1.995264E 06

I(Q) - PRINCIPLE INERTIAS

2.0086129 06		
	1.056030E 06	
		6.156493E 04

Q - TRANSFORMATION MATRIX

-9.339642E-02	2.044180E-02	9.963079E-01
-1.242722E-01	9.917727E-01	-3.074978E-02
-9.837382E-01	-1.2637785-01	-3.015996E-02

TABLE 3.3-9 Output From Grid Point Weight Generator

$$
\begin{bmatrix}
Wx & 0 & 0 & 0 & Sy(x) & Sz(x) \\
0 & Wx & 0 & Sx(y) & 0 & Sz(y) \\
0 & 0 & Wz & Sx(z) & Sy(z) & 0 \\
0 & - & - & Ixx & -Ixy & -Ixz \\
- & 0 & - & -Iyx & Iyy & -Iyz \\
- & - & 0 & -Izx & -Izy & Izz
\end{bmatrix}
$$

FIGURE 3.3-8 GPWG OUTPUT MATRIX ARRANGEMENT

TABLE 3.3-10 Weight Calculation Comparison

Panel I.D.	Wt(lbs)	X FS	Y WS	Z WL	I_{xx} (10^6 Lb-In^2)	I_{yy} (10^6 Lb-In^2)	I_{zz} (10^6 Lb-In^2)
51028	10.160	2189	428	277	2.56	49.48	50.50
51030	1.910	2235	420	281	.487	9.70	9.88
51032	5.898	2330	435	288	1.609	32.51	33.14
51034	9.294	2417	436	295	2.584	55.12	56.08
51036	11.610	2514	436	303	3.29	74.43	75.58
51038	12.888	2614	437	310	3.724	89.29	90.52
51040	7.112	2716	440	316	2.093	53.17	53.85
51130	9.518	2282	452	282	2.705	50.32	51.51
Calc Total	68.39	2436	436	297	19.05	414.0	421.1
NASTRAN Total	66.76	2429	435	296	18.57	401.8	408.7

3.3.5 NASTRAN weight factoring process -

3.3.5.1 Introduction and objective - Two finite element models were set up to represent the Baseline aircraft. The first, a production sized model, to represent the actual member sizing of the L-1011-500 aircraft as derived from engineering drawings and reports. The second was a finite element model whose covers were sized through the PADS procedures, but with ribs, and spars remaining as in the production sized model. The decision was made early in the program that finite element sizing would be used to generate cover sizing only, to simplify the design cycle. Cover weight represents approximately half of the structural weight of a typical aspect ratio seven wing, and thus would be sufficient to indicate significant weight trends.

The structural sizing for the covers contained on NASTRAN property cards was used as a basis for weight estimation. The overall goal of the process was to provide timely and accurate structural system weight data based on the finite element analysis. Deriving structural weight estimates from a finite element model involves three fundamental steps.

3.3.5.1.1 Determination of finite element structural weight -

The finite element analysis process is an idealization that by its nature requires judgment on the part of the analyst. Beams, plates, rods, or other finite elements are used to approximate real structure. The analyst's judgment is key to the realistic representation of the structure. Whether or not a finite element should be included in the structural system or how it should be factored are important issues to be understood before a weight estimation is performed. One example is a hydraulic actuator. The analyst may decide to model the actuator with a very large, stiff beam. This beam is not intended to represent the actuator for weight purposes and should not be considered as part of the finite element structural weight. Another example is the engine/pylon structure. In this case, the purpose of the model may be to provide a reasonable stiffness model to determine accurate deflections. The engine/pylon structure may be represented by five or six beam elements instead of 2000 to 3000 elements. It would be unwise to predict engine weight from a representation of this nature.

3.3.5.1.2 Finite element weights factoring process - A complete weight estimate can be made by multiplying the finite element weight by factors. These factors account for the differences between what is represented by the finite element model and what actually exists. For example, a spar may be represented by a combination of rods and plates. The rods would represent the spar caps, and the web may be represented by a single, constant thickness plate. The actual web represented by that plate may be chem-milled with edge margin and thickness for fasteners, access holes, system penetrations, etc. The actual plate may be painted. Stiffeners may not have been included in the model. The effects of these examples can be accounted for with factors.

In the PADS sizing weight calculation process, two sets of factors will be applied. Figure 3.3-9 illustrates this process. The first set of factors, called nodal factors, were produced as ratios of finite element nodal weights for two different sizing data sets. These nodal factors compensate for weight which was not accounted for during the PADS sizing of the baseline airplane. This form of factoring adjusts each nodal weight individually. The numerator of each ratio was the nodal weight from a sizing data set which represents section properties of the existing L-1011-500 ircraft (entry 1 in table 6.0-2). These section properties were derived from drawings. The NASTRAN model with this sizing shall be referred to as the production finite element model. Finite element nodal weights for this model will be called production FEM weights.

The denominator of each ratio was the nodal weight for a final sizing data set produced by the PADS sizing process (2nd flexible aircraft loads sizing with stress margins of safety, entry 9 in table 6.0-2). The NASTRAN model with a sizing data set produced in PADS will be referred to as a PADS finite element model. Finite element nodal weights for a PADS finite element model will be called PADS FEM weights.

PADS FEM weights factored by the production factors will be called production factored PADS FEM weights.

The second set of factors adjusts the regional weights of the finite element representation to approximate the regional weights of a weights representation for the existing L1011-500 aircraft. These factors are called regional factors. They adjust for weights of structure not modeled in the finite element model. Examples are fairings, access panels, control system supports, fasteners, paint, sub-structure, etc. Those items not adaptable to a finite element factoring process, such as secondary structure, must be added to the factored finite element weight totals. The factors are applied to seven regions on the wing. All nodal weights within a region

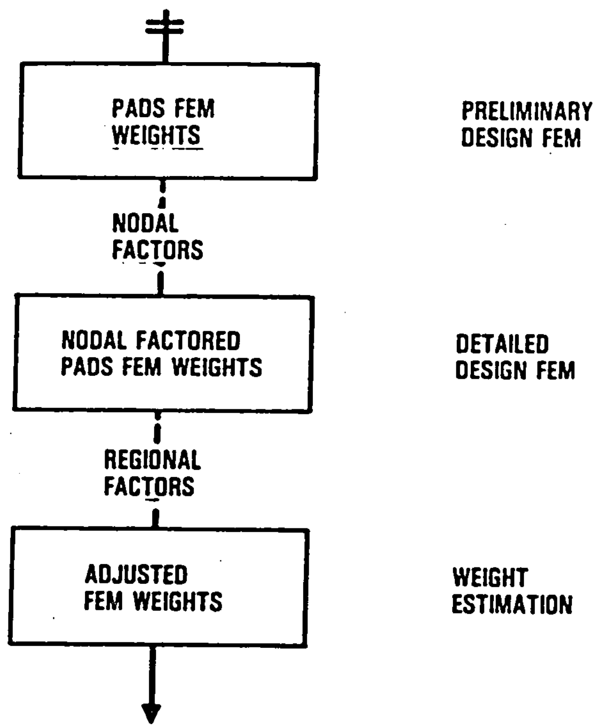


FIGURE 3.3-9. FEM WEIGHT FACTORING PROCESS

are factored with a single factor. The factors retain the distribution of the region.

Production factored PADS FEM weights which have been factored by the regional factors are called adjusted FEM weights.

3.3.5.1.3 Nodal factors - The nodal factors are in seven 3741 by 3741 diagnal matrices. A seperate matrix contains the nodal factors for each of the following components on the finite element model:

Component	Elements	Matrix#
Center box ribs and spars		1511
Center box upper covers		1512
Center box lower covers		1513
Wing box upper covers		1514
Wing box lower covers		1515
Wing box spars		1516
Wing box ribs		1517

The total weight of the production finite element sizing for the designed elements is 10,961 pounds and the total weight of the PADS sizing used in formation of the factors is 9,718 pounds.

3.3.5.1.4 Regional factors - The design regions for which the regional factors apply are defined in figure 3.3-10. Tables 3.3-11 through 3.3-14 present the regional factors for each of the major wing components by design region.

These factors are applicable to derivative design configurations. It was assumed that the weight factors would not be dependent upon aspect ratio, sweep, or thickness ratio.

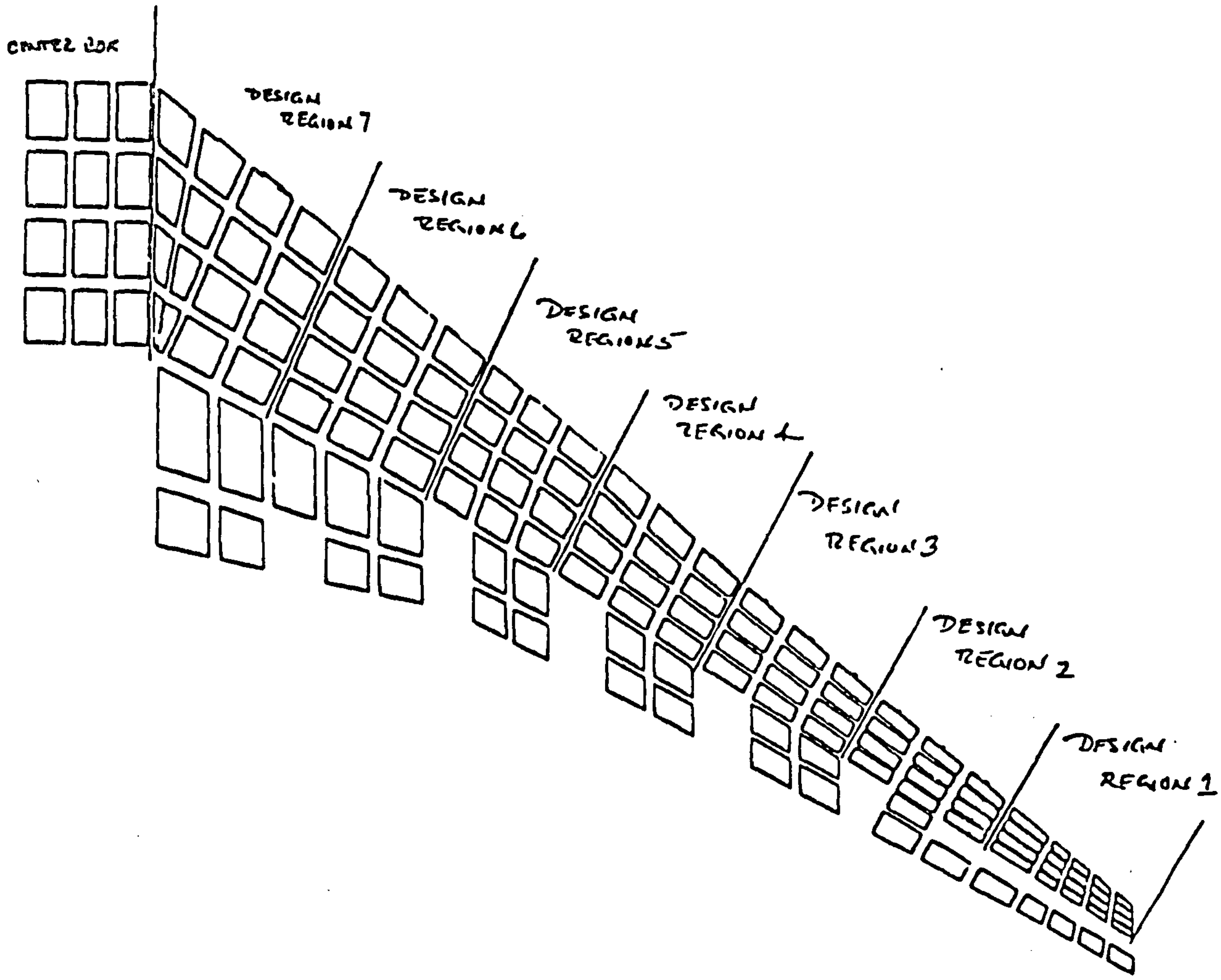


FIGURE 3.3-10 Design regions for regional factoring

Table 3.3-11 Upper Cover Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	1.30000
2	1.22260
3	1.10119
4	1.08630
5	1.07580
6	1.18520
7	1.29705

Table 3.3-12 Lower Cover Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	1.32790
2	1.12800
3	1.05980
4	1.14470
5	1.01660
6	1.10940
7	1.26930

Table 3.3-13 Spar Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	0.64430
2	0.68060
3	0.83680
4	0.92640
5	0.74260
6	0.91520
7	1.19445

Table 3.3-14 Rib Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	1.66090
2	1.28800
3	1.32650
4	1.05880
5	0.75800
6	0.63720
7	0.87100

3.3.5.2 Update to the Mass Distribution Module - Weight matrices are output from the Grid Point Weight Generator (GPWG) by component (7 components) and are factored immediately by the nodal factors for the structural node distribution. The matrices are then broken down into design regions, and the regional factors are applied for each design region of each component. A single matrix is then formed by recombining the pieces.

Nonmodeled weight such as trailing and leading edge, flaps, control surfaces, etc., are added, and a new wing weight is determined. This new weight and distribution is then converted to Mass Distribution Module input and ready to be recycled for the updated weight production.

A general overview of the initial weight data determination and the subsequent weight update which reflect sizing data from the aeroelastic design process is shown in figure 3.3-11.

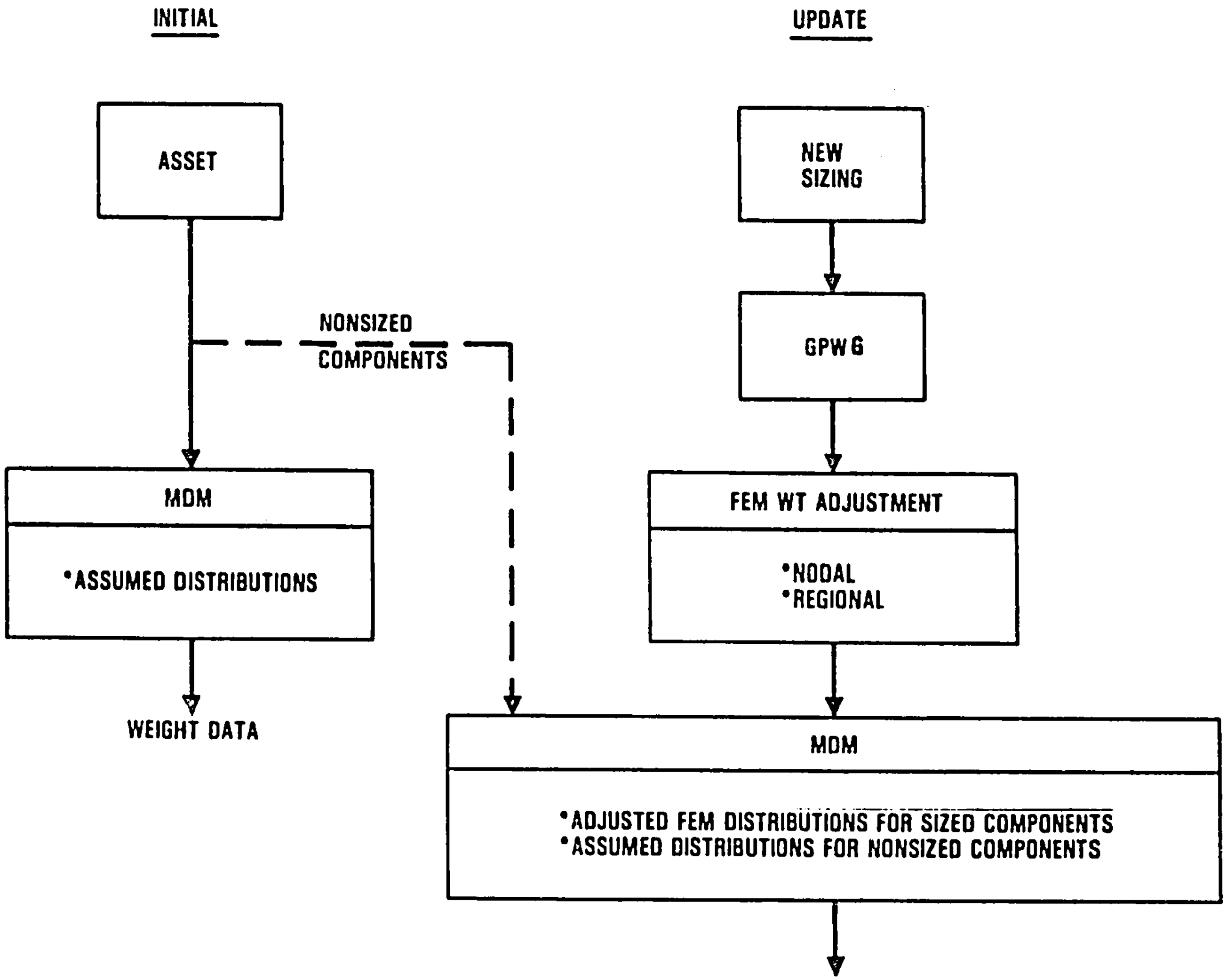


FIGURE 3.3-11. WEIGHT DATA PROCESS FLOW CHART

3.4 Static Loads

3.4.1 Discussion of loads methodology - Net balanced load distributions, together with integrated vertical shears, bending moments, and torsions acting on the wing, were generated for appropriate flight and ground conditions. These conditions were considered as a minimum set to be used for sizing studies associated with this task. They do not, nor were they intended to, represent a complete set of load conditions normally associated with a total analysis cycle.

3.4.1.1 Aerodynamic stability derivatives - Aerodynamic stability derivatives were generated using the theoretical vortex lattice method of reference 1. Stability derivatives were generated for mach = 0.5, 0.80 and 0.88. Aerodynamic influence coefficient matrices were generated from downwash matrices output by VORLAX. Figure 3.4-1 illustrates the aerodynamic load points used to generate these data for the Baseline airplane. Fuselage and empennage panels are identical for all derivatives. The total number of panels on the wing are the same for all derivatives. AR12 derivative wing panels have been scaled such that the semispan locations of the load points match, where possible.

Aileron effectiveness ratios, used for active control on loads runs were obtained from L-1011-500 data. These ratios reflect flight test correlation factors. Use of such data is considered to be normal practice during analysis cycles. Table 3.4-1 summarizes the airplane stability derivatives used in this analysis.

The input data files (as catalogued in PANVALET) used to generate the VORLAX output airload distributions are listed in Appendix E. NASA is referred to Report 2865 for a definition of the contents of these data files. It should be noted that airload distribution data and stability derivatives generated through use of these files will represent rigid airplane data with the horizontal stabilizer/elevator at zero deflection angles.

The airplane is put into balance by using the following horizontal stabilizer/elevator gearing curve:

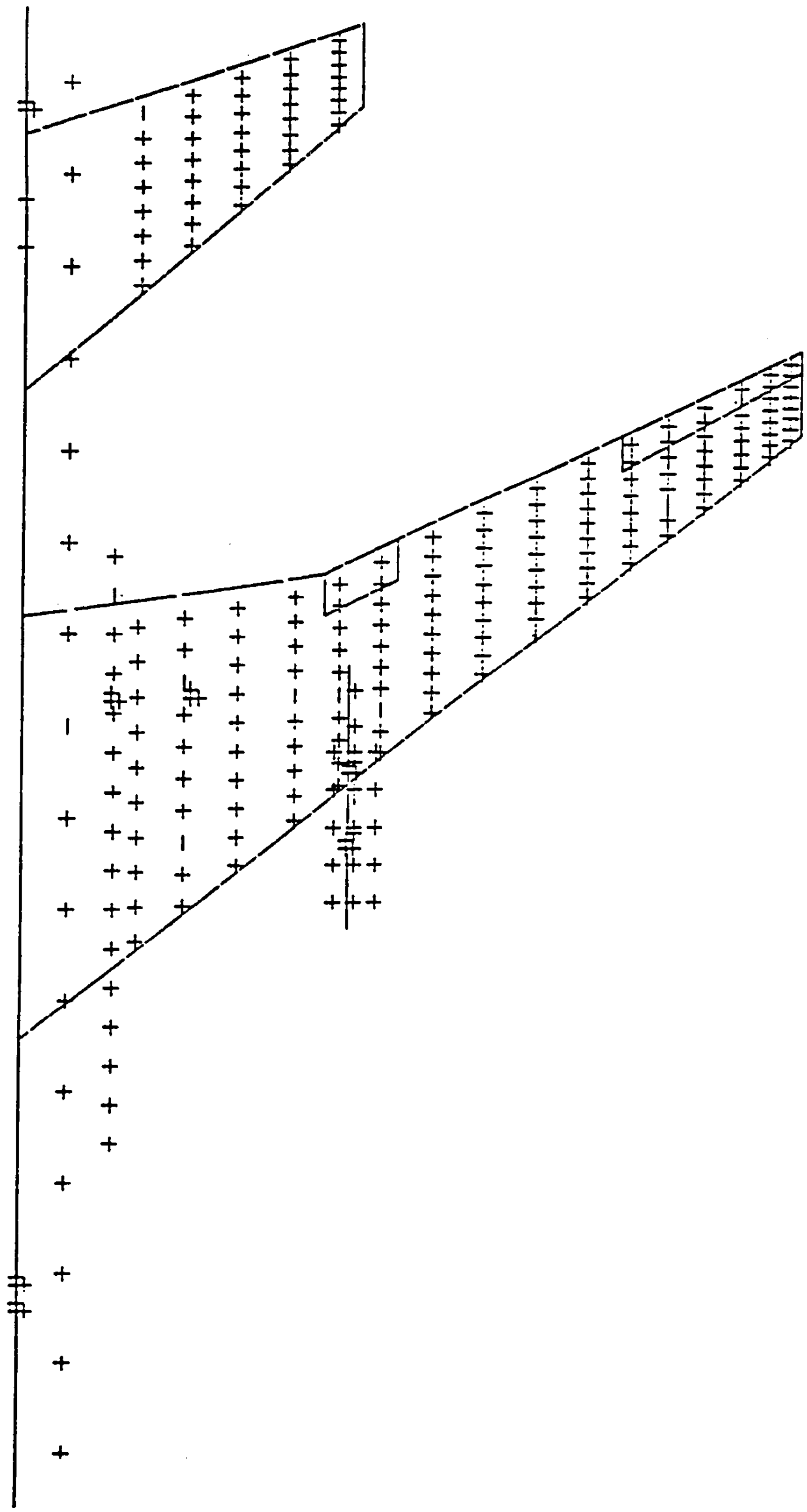


FIGURE 3.4-1 LOADS AERODYNAMIC LOAD POINT GRID FOR BASELINE

TABLE 3.4-1 AERODYNAMIC STABILITY DERIVATIVES

DERIVATIVE	BASELINE A/P SWEEP = 35			AR12 A/P SWEEP = 25			AR12 A/P SWEEP= 35		
	MACH NUMBER			MACH NUMBER			MACH NUMBER		
	0.5	0.8	0.88	0.5	0.8	0.88	0.5	0.8	0.88
C_L @ alpha=0	.0508	.0622	.0713	.0702	.0909	.1093	.0651	.0809	.0932
C_m @ alpha=0	.0418	.0451	.0468	.0372	.0416	.0430	.0411	.0438	.0445
$C_{L\alpha}$	.1074	.1241	.1363	.1275	.1540	.1756	.1187	.1388	.1534
$C_{m\alpha}$	-.0290	-.0371	-.0433	-.0296	-.0391	-.0470	-.0380	-.0491	-.0575
$C_{L_{wing}}$ @ alpha=0	.0614	.0746	.0852	.0716	.0928	.0117	.0661	.0824	.0952
$C_{m_{wing}}$ @ alpha=0	-.0016	-.0092	-.0125	.0040	-.0009	-.0069	.0098	.0052	.0003
$C_{L\alpha_{wing}}$	.0767	.0905	.1007	.0973	.1200	.1387	.0886	.1051	.1171
$C_{m\alpha_{wing}}$	-.0196	-.0249	-.0293	-.0169	-.0227	-.0287	-.0237	-.0307	-.0363

Horizontal Stab. Defl. Angle (deg)	Elevator Defl. Angle (deg)
-14.0	-25.0
-10.0	-14.0
-6.0	-5.3
-4.0	-2.4
-2.0	-0.4
1.0	0.0

Note that the elevator is geared to the horizontal stabilizer. That is, a pilot induced stick force drives the horizontal stabilizer. The elevator follows the horizontal stabilizer. The elevator can not move independently, nor can it drive the horizontal stabilizer.

3.4.1.2 Wing loads calculations - FAMAS Master Deck programs PSRL and GDHL were used to generate net balanced load distributions and integrated shears, bending moments, and torsions for the specified conditions of section 3.4.2.1. PSRL was used for flight maneuver conditions, and GDHL was used for ground handling conditions. All conditions satisfy FAR-25 specifications.

Both programs are matrix algebra panel loads methods that use D'Alembert's principle. That is, aerodynamic and externally applied forces are placed in equilibrium with inertia forces. Programs PSRL and GDHL can generate balanced net load distributions for either the rigid airplane or a flexible airplane. Structural flexibilty is introduced through input of an appropriate structural influence coefficient (SIC) matrix. SIC matrices are obtained from finite element modeling methods as described in section 3.5.

The panel loads grid consists of the aerodynamic panels generated when using VORLAX (see section 3.4) plus additional panels required to represent concentrated masses or nonaerodynamic panels. These consist of panel points for the nose and main landing gears, center engine, wing engine and pylon. Two sets of landing gear panel points are required, one for gear up and one for gear down. Note that the panel point used for the aerodynamic grid from VORLAX corresponds to the aerodynamic load point, which is at the quarter chord and semispan of each panel. Panel points for the concentrated masses and nonaerodynamic panels are at the SIC nodes.

PSRL is a steady symmetric-flight, transient pitch maneuver, and uncoupled one degree of freedom maneuver analyses program.

This program was used to compute the loads for steady pitch conditions. Angle of attack and stabilizer angle are treated as variables, with the program calculating the values required to produce the desired load factor together with zero pitch acceleration. Inputs include a number of "measured to theoretical" ratios which are used to adjust the over-all level of the theoretical distributions. For this analysis, only the outboard aileron ratios were used. The program calculates the more critical values of pitch velocity associated with either a steady pull-up, or a steady turn. Incremental airload distributions due to control surface deflections and structural flexibility are a function of the input AIC matrix and the change in local streamwise angle of attack. The change in local streamwise angle of attack is caused by the net balanced load distributions. The flexible loads solution is a closed form solution.

GDHL is set up to run all the FAR-25 ground handling conditions in a single run. Ground loads are reacted by rigid body translational and rotational accelerations for each specified weight and center of gravity combination. GDHL is also used to generate net balanced load distributions for arbitrary ground reactions, such as those associated with dynamic taxi conditions.

3.4.1.2.1 Rigid loads calculations - Net balanced load distributions were generated for the rigid airplane using a null SIC matrix when running PSRL and GDHL. Aerodynamic load distributions, as generated from the P-130 AIC matrices, reflected the midcruise shape for all flight maneuver conditions. Net balanced load distributions acting at the SIC locations of the finite element model were used for the first pass strength sizing.

3.4.1.2.2 Corrections for Jig Shape - Structural deflections associated with the SIC matrix are measured from the jig shape (manufacturing shape). AIC matrices generated by VORLAX are for the midcruise shape. PSRL accepts the input of the jig and midcruise camber distributions and corrects the input AIC matrix to the jig shape.

To make this correction, rigid body net balanced load distributions were generated for a 1-g flight at the midcruise speed and altitude point. The midcruise flight condition was considered to be a gross weight = 350,300 pounds, center-of-gravity at 12.5% MAC (reference), velocity = 360 KEAS, and mach = 0.80.

Structural deflections due to this 1-g condition, as obtained from the SIC matrix, were converted to incremental changes in streamwise angle of attack at the basic loads panels. These incremental changes were subtracted from the midcruise shape obtained as output from P-130, thus defining the jig-shape camber distribution. Under 1-g cruise conditions the airplane would deflect structurally to the optimum performance shape used by P-130 to generate the AIC matrices. Both the midcruise and jig shape camber distributions are input to PSRL.

3.4.1.2.3 Flexible load calculations - Net balanced flexible load distributions were generated using the SIC matrices provided by the PADS sizing process with rigid loads or flexible loads from the previous pass. For each level of SIC matrices, a new jig shape camber and twist was generated for input to PSRL and GDHL. Aerodynamic and inertia distributions acting at the basic loads panels were transformed to an equivalent sets at the SIC. Structural deflections at the SIC are converted to streamwise slopes at the basic loads grid. The closed solution used by PSRL requires transformation matrices from the basic loads grid to the SIC grid, plus a differentiation matrix to convert deflections at the SIC grid to slopes at the AIC control panels. Since these transformations are a function of grid geometry only, they are generated once for an airplane. These transformation matrices need only be changed when there is a change in one of the grid systems. Normally, this is only required when there is a change in the external configuration.

3.4.2 Conditions selected for design analysis - The flight environment, airplane configuration, and maneuver conditions for external loads analysis were selected from existing loads envelopes for the L-1011-500 airplane. Basically this analysis was considered to be a first pass preliminary cycle for a candidate derivative airplane.

The objective was to generate net balanced external load distributions for a minimum number of design conditions sufficient to determine a first level wing structural sizing. These loads are not considered to be a complete set of load conditions normally associated with a full blown analysis.

3.4.2.1 Steady maneuver flight conditions - Net balanced external load distributions were generated for the following five load conditions:

Flight condition No.	1101	1102	1103	1104	1105
Weight - 1000 lb	350.3	504.0	504.0	504.0	504.0
C.G. - % MAC (ref.)	12.5	17.1	17.1	17.1	17.1
Flight condition	Midcruise	VA	VC	VD	Flap Ext.
Mach No.	0.80	0.48	0.82	0.88	0.33
Ve - KEAS	360.0	316.0	356.0	418.0	220.0
Altitude-1000 ft	20.0	0.0	21.3	17.3	0.0

where flight conditions noted above are defined in reference 9 , section 25.333.

Only forward c.g. gross weights were considered for symmetric flight maneuver external loads analyses. For the candidate derivative airplane with an all flying tail, forward c.g. locations require a down stabilizer tail load, thus increasing the total airload on the wing for a given maneuver load factor.

Each load condition (i.e., 1101 through 1105) generated net balanced external load distributions for five unique symmetric maneuvers. The five maneuvers are identical for each load condition, as follows:

	Maneuver Condition	Active Control
1)	1g Level Flight	ON
2)	Positive steady maneuver-(PSM)	ON
3)	Negative steady maneuver-(NSM)	ON
4)	Positive steady maneuver-(PSM)	OFF
5)	Negative steady maneuver-(NSM)	OFF

NOTE: With active control system on, aileron is biased down two degrees at 1g, with flaps retracted, and up eight degrees with flaps down. Aileron returns to neutral with system off.

The incremental airload distributions due to the aileron deflection with the ACS on have correlation factors applied which reflect measured to theoretical data. These factors were determined during the L-1011 airplane project load analysis. These factors are applied to the input aileron deflection angle. The applied factors are as follows:

Flight Cond.#No.	Man.#1	Man.#2	Man.#3
1101	0.88	0.75	0.75
1102	0.94	0.88	0.88
1103	0.78	0.73	0.69
1104	0.73	0.90	0.90
1105	0.97	0.90	0.90

3.4.2.2 Ground Handling load conditions - Net balanced load distributions were generated for the following basic ground handling conditions as specified by FAR 25 (ref. 9):

	Description	Symbol	FAR-25 REG. (ref. 9)
1.)	Two wheel brake roll	BR-2	25.493
2.)	Three wheel brake roll	BR-3	25.493
3.)	Reverse braking	REV. BR.	25.507
4.)	Unsymmetric braking	UNSYM. BR.	25.499
5.)	Turning	TURN	25.495
6.)	Three wheel static	3WS	25.489

Two types of ground handling conditions which are critical for the inboard wing of the L-1011-500 can be generated using GDHL. These are BR-2 from the above table, and a level landing condition, referred to as LL-2. Thus, for this analysis, the above six ground handling conditions and an LL-2 condition were run.

GDNL generates level landing conditions for LL-2 by combing 1g net balanced flight landing load distributions, as gernerated using PSRL, with inertial loads distributions due to incremental ground reactions at the main landing gears. LL-2, therefore, refelects either the rigid or flexible airplane as determined by the availability of an SIC matrix.

In addition to the above, a pseudo dynamic taxi condition was generated using GDHL. Normally this type condition is generated by Dynamic Loads. However, for this analysis, the appropriate load factor and gear reactions were provided by Dynamic Loads, and input to GDHL. It was felt that this pseudo condition would achieve the desired result.

The following table lists the inertia cases used for these conditions:

Condition No.	3301	3303	3304
Weight - 1000 lb	506.0	368.0	506.
c.g. - % MAC (ref.)	26.9	14.1	26.9
Type Condition	Grd. Hnd.	Lev. Ld.	Dyn Taxi
Gear Position	Down	Down	Down

Note: only the BR-2, LL-2 and dynamic taxi conditions were included in the ground handling external load distributions used for internal stress analysis.

3.4.2.3 Design load envelopes - Figures 3.4-2 through 3.4-7 present typical load envelopes of the integrated shears, bending moments, and torsions at the wing root for the 2nd flexible Baseline, AR12, 35 Degree Sweep, and AR12, 25 Degree Sweep airplanes.

The design load envelopes are based on the stacking of the steady maneuver flight conditions, ground handling conditions, landing conditions, and dynamic taxi conditions. The terms, stacking and stacked, are used in the context that each condition is computed separately. Then, each of these conditions are inserted (stacked) into one matrix, where each column of that matrix represents one of the conditions. In a full design effort, this matrix may contain up to 2000 columns, with each column representing a separate load condition.

Note that the steady symmetric flight maneuvers with the ACS system OFF are considered flight restricted dispatch

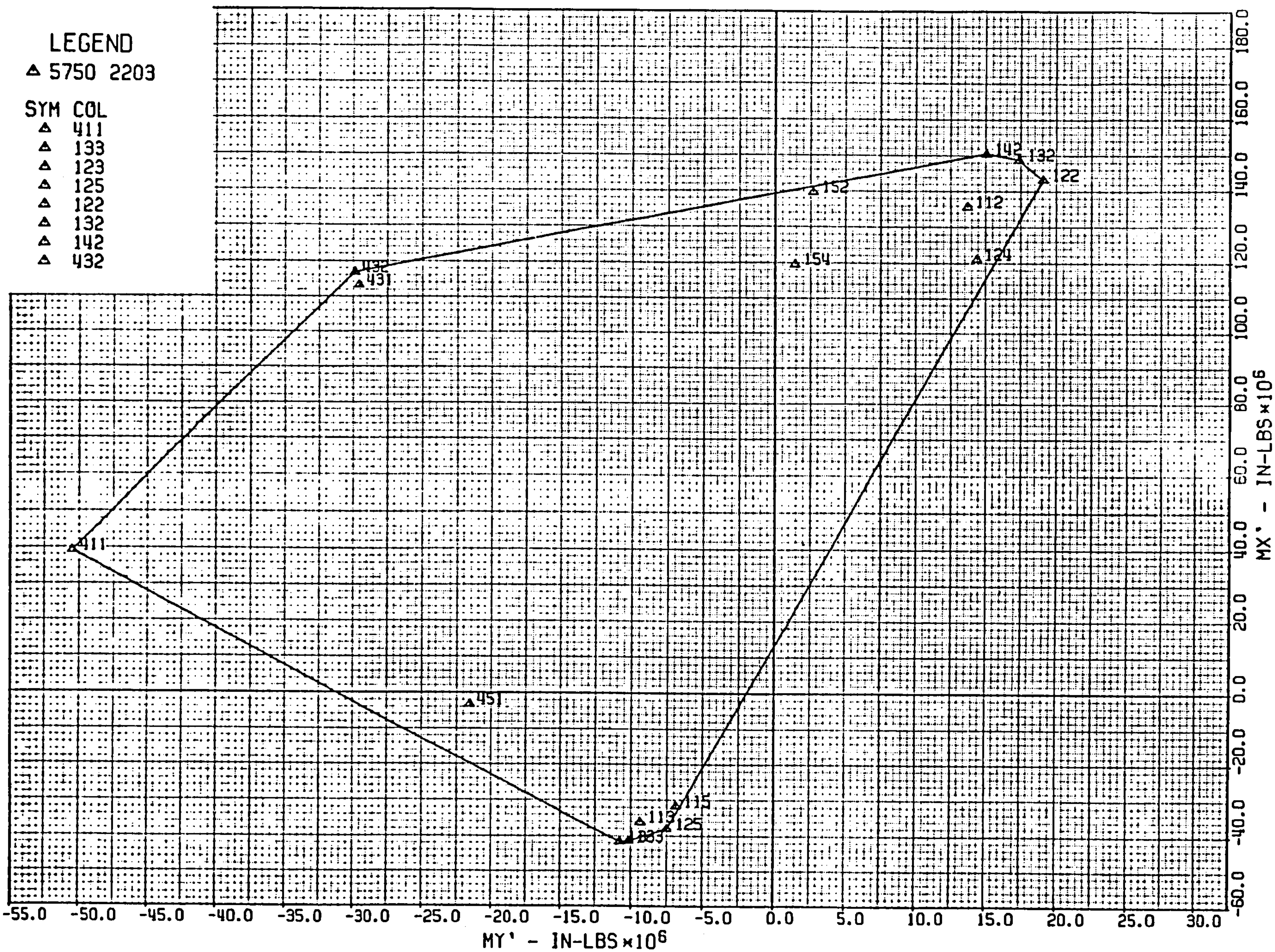


FIGURE 3.4-2 Baseline 2nd flex loads wing root envelope - Mx vs My

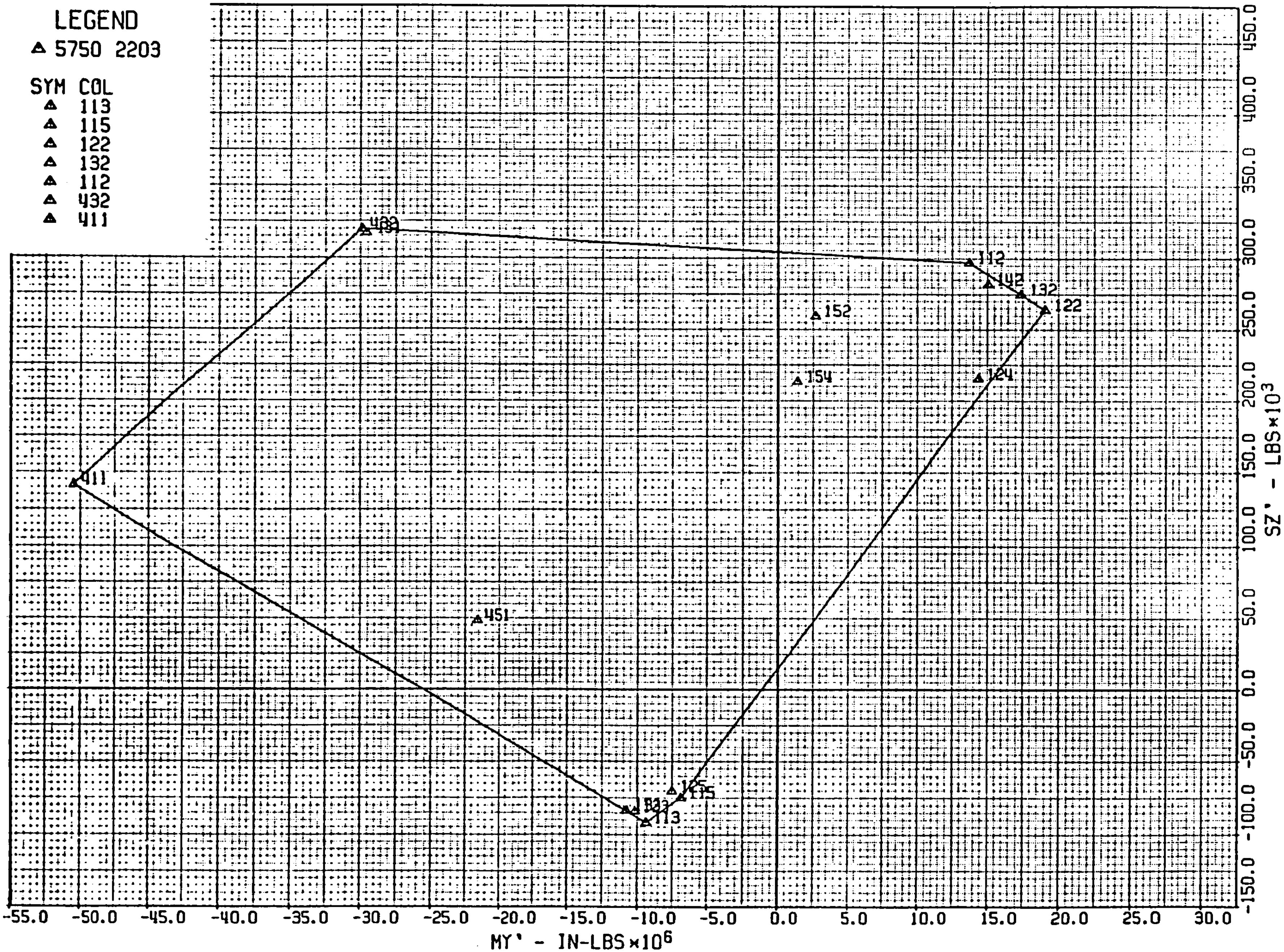


FIGURE 3.4-3 Baseline 2nd flex loads wing root envelope - Sz vs My

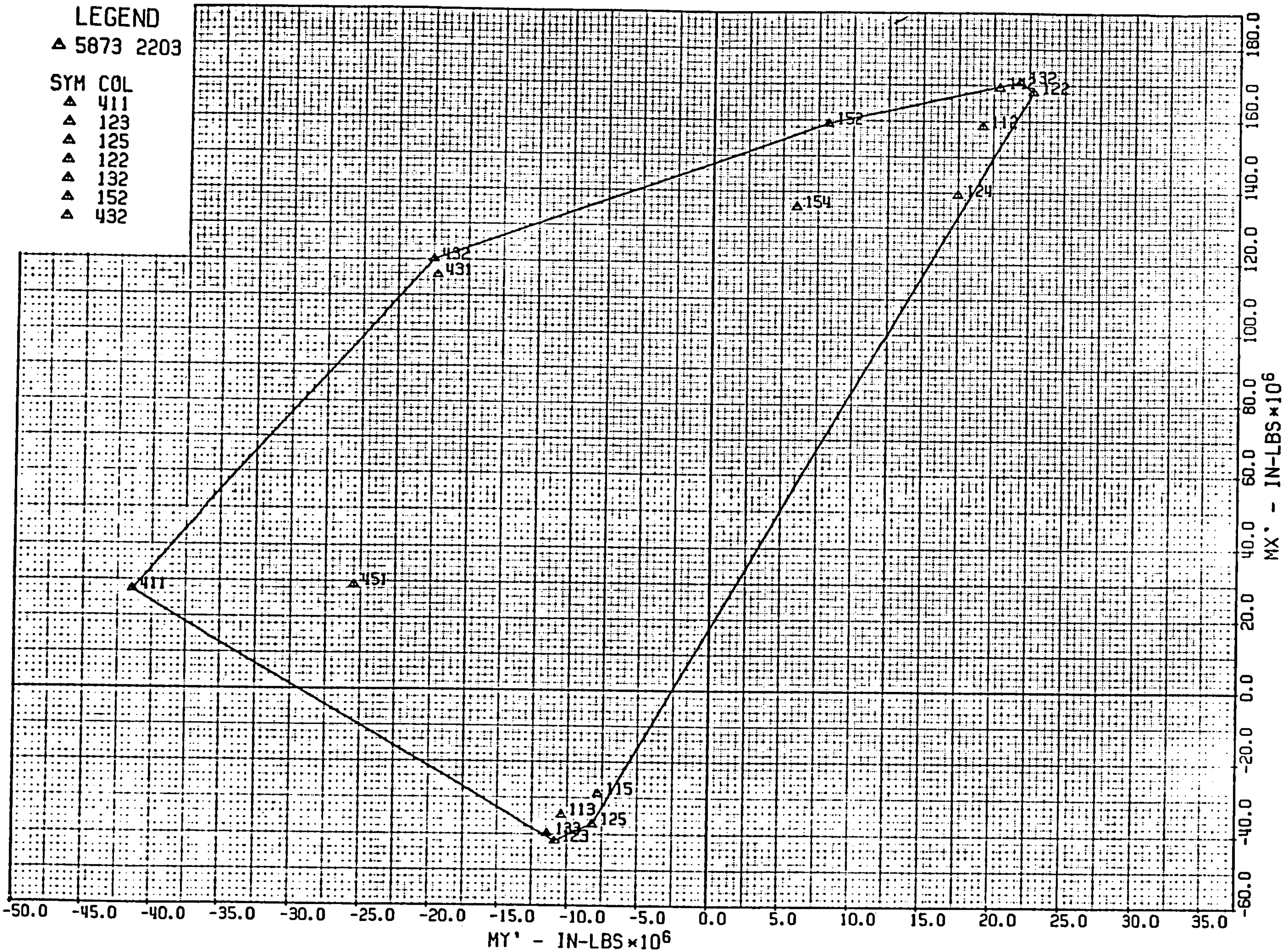


FIGURE 3.4-4 AR12 35 Deg. Sweep 2nd flex loads wing root envelope - Mx vs My

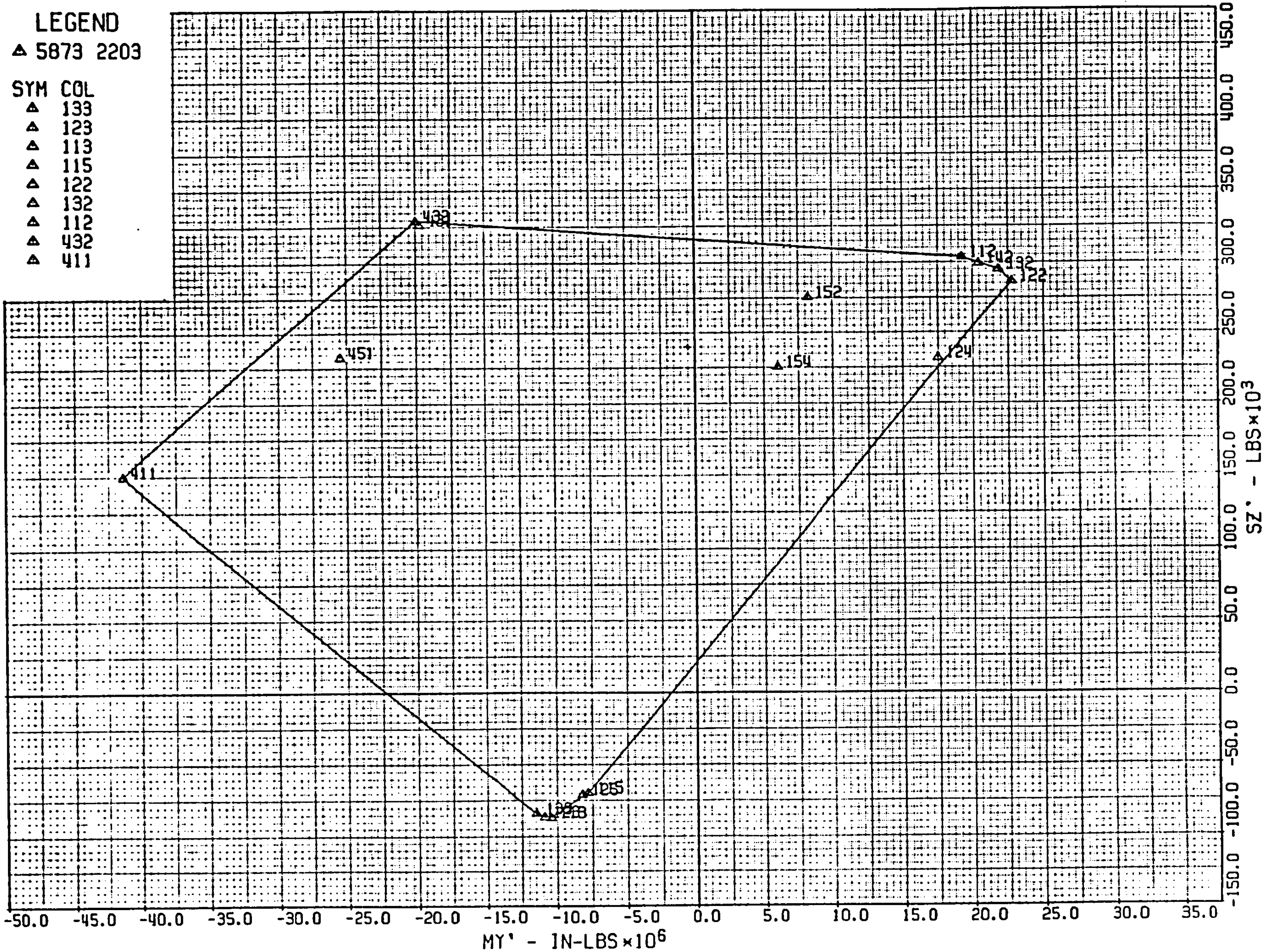


FIGURE 3.4-5 AR12 35 Deg. Sweep 2nd flex loads wing root envelope - Sz vs My

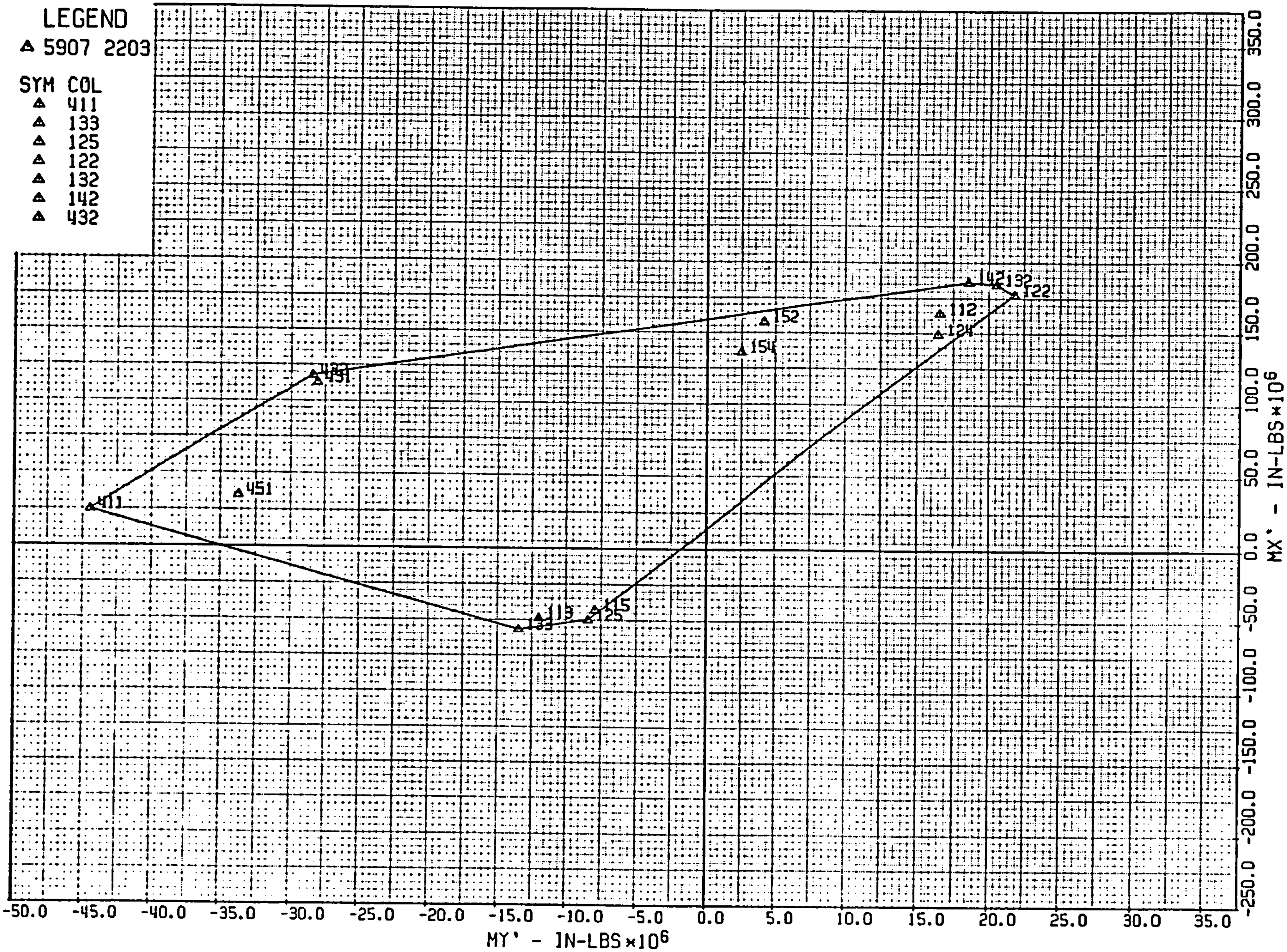


FIGURE 3.4-6 AR12 25 Deg. Sweep 3rd flex loads wing root envelope - Mx vs My

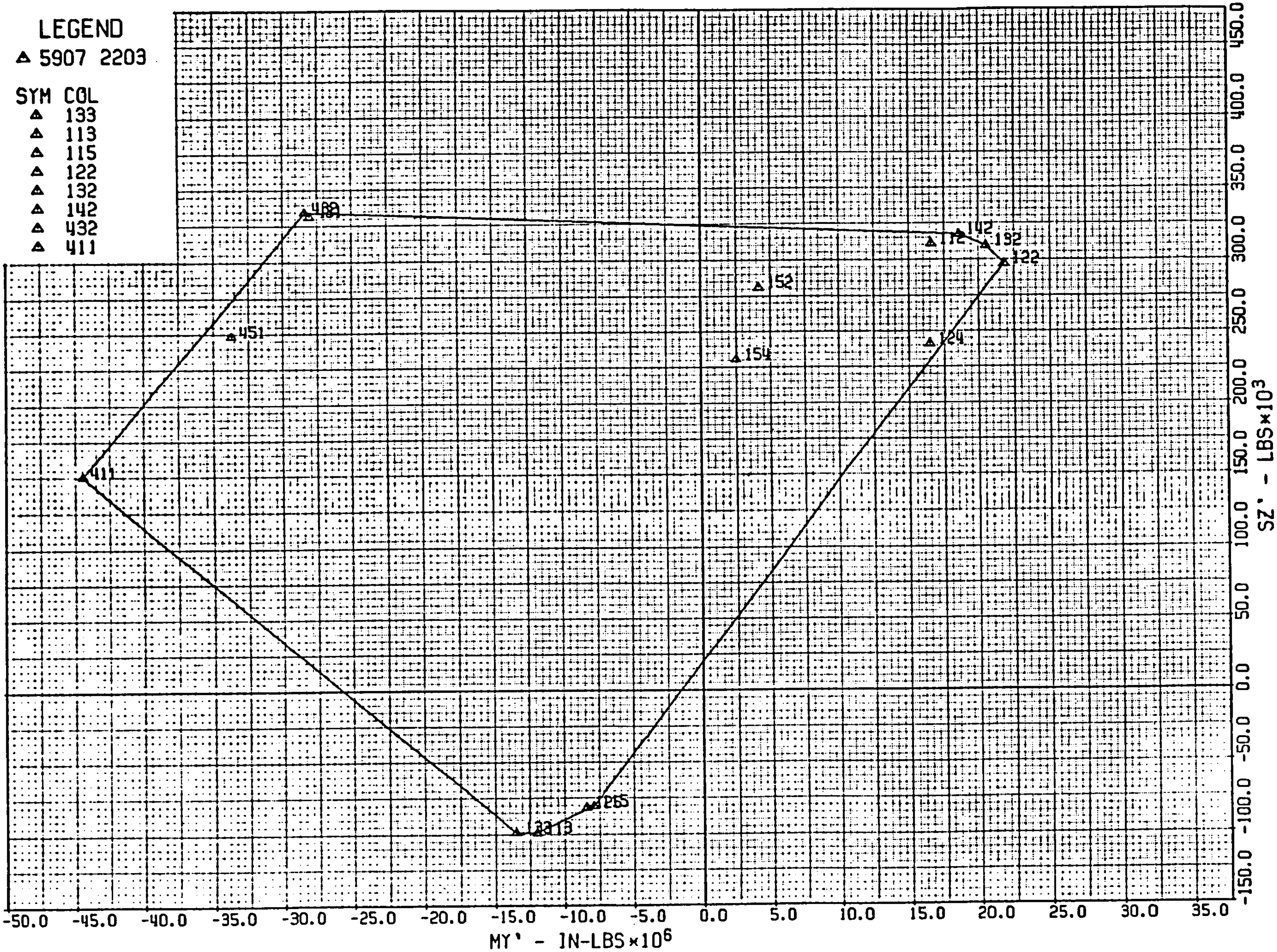


FIGURE 3.4-7 AR12 25 Deg. Sweep 3rd flex loads wing root envelope - Sz vs My

conditions. Thus, when included in the stacked matrix each system OFF condition is factored by 0.80. This is not the equivalent of a true +2.0 or -0.8 g maneuver, in that the airplane loads are generated at the balanced angle of attack and stabilizer setting required for a full steady maneuver. This approach was the result of extensive discussions with the FAA for the active control certification process. The approach has a mathematical formulation based on limit load factor frequency of exceedance level and the ACS inflight availability. Material on this subject may be found in Volume II of this report in a letter to NASA, LAC/081823.

Flight maneuver and ground handling load conditions are stacked into a single matrix to be passed on for the sizing process. Table 3.4-2 summarizes the flight maneuver load conditions and table 3.4-3 summarizes the ground handling load conditions for BR-2, LL-2 and dynamic taxi conditions. The column number in the stacked matrices where these conditions are found is equivilent to the condition number. Matrix 2200 is the stacked balanced net load matrix. Table 3.4-4 summarizes pertinent parameters for each of the flight maneuver columns in the stacked matrices.

Although table 3.4-4 is basically self explanatory, the following definitions are emphasized:

Parameters For Stacked Matrices For Sizing Analysis

Parameter	Definition
Condition Number	The loads condition number and column number for that maneuver.
Column Number in Stacked Matrix	The column in the stacked matrix in which the above column is stacked.
Alpha FRL	Airplane angle of attack relative to fuselage reference axis. Angle required to balance A/P.
Hor. Stab. Defl.	Horizontal stabilizer angle required to balance A/P.
Elevator Defl.	Elevator angle - fixed function of stabilizer angle (geared to horiz. stab.).
Aileron Rigging	Denotes aileron angle for the ACS system on.

NOTE: Control surface angles are positive for trailing edge down.

Sign convention for external loads analysis is the left-hand rule, with positive Y acting outboard on the left wing.

External loads are generated for the left side of the A/P.

Cond #	Gross Weight	c.g. % Mac	Mach	Velocity (keas)	Altitude (1000 ft)	g's	Active Controls
111	350,300.	12.5	.80	360	20.0	1.0	On
112	350,300.	12.5	.80	360	20.0	2.5	On
113	350,300.	12.5	.80	360	20.0	-1.0	On
114	350,300.	12.5	.80	360	20.0	2.5	Off
115	350,300.	12.5	.80	360	20.0	-1.0	Off
122	504,000.	17.13	.473	316.	0.0	2.5	On
123	504,000.	17.13	.473	316.	0.0	-1.0	On
124	504,000.	17.13	.473	316.	0.0	2.5	Off
125	504,000.	17.13	.473	316.	0.0	-1.0	Off
132	504,000.	17.13	.82	356.	21.3	2.5	On
133	504,000.	17.13	.82	356.	21.3	-1.0	On
134	504,000.	17.13	.82	356.	21.3	2.5	Off
135	504,000.	17.13	.82	356.	21.3	-1.0	Off
142	504,000.	17.13	.88	418.	17.3	2.5	On
143	504,000.	17.13	.88	418.	17.3	0.0	On
144	504,000.	17.13	.88	418.	17.3	2.5	Off
145	504,000.	17.13	.88	418.	17.3	0.0	Off
152	504,000.	17.13	.33*	220.	0.0	2.0	On
153	504,000.	17.13	.33*	220.	0.0	0.0	On
154	504,000.	17.13	.33*	220.	0.0	2.0	Off
155	504,000.	17.13	.33*	220.	0.0	0.0	Off

*Flaps extended condition

TABLE 3.4-2 Maneuver load conditions for Baseline design

TABLE 3.4-3 Ground handling load conditions for Baseline design

Cond#	Gross Wt. lb	c.g. %MAC (ref.)	Mach	Velocity (keas)	Altitude (1000 ft)	g's	Active Controls
411	506000	26.9	NA	NA	NA	1.0	NA/BR2
431	368000	14.1	.27*	182.	0.	2.141	ON/LL2
432	368000	14.1	.27*	182.	0.	2.141	OFF/LL2
451	506000	26.9	NA	NA	NA	1.7	NA/TAXI

* FLAPS EXTENDED CONDITION
NA NOT APPLICABLE

TABLE 3.4-4 Load condition parameters

PARAMETER SHEET FOR STATIC FLIGHT LOADS						
CONDITION NUMBER		1101/ 1	1101/ 2	1101/ 3	1101/ 4	1101/ 5
CONDITION DESCRIPTION		BASIC	PSM	NSM	PSM	NSM
GROSS WEIGHT	LB	350300.000	350300.000	350300.000	350300.000	350300.000
FUEL WEIGHT	LB	12300.000	12300.000	12300.000	12300.000	12300.000
FS (C.G.)	0/0 MAC	12.500	12.500	12.500	12.500	12.500
WL (C.G.)	IN	194.000	194.000	194.000	194.000	194.000
AIRSPEED	KEAS	360.000	360.000	360.000	360.000	360.000
MACH NUMBER		0.804	0.804	0.804	0.804	0.804
ALTITUDE	1000 FT	20.000	20.000	20.000	20.000	20.000
POWER SETTING		0	0	0	0	0
NX (C.G.)	INERTIA	-0.072	0.020	-0.036	0.026	-0.035
NY (C.G.)	LOAD	0.0	0.0	0.0	0.0	0.0
NZ (C.G.)	FACTOR	-1.000	-2.500	1.000	-2.500	1.000
THETA DOT		0.0	0.081	-0.077	0.081	-0.077
THETA D-DOT	RAD/SEC	0.0	0.0	0.0	0.0	0.0
PHI DOT	AND	0.0	0.0	0.0	0.0	0.0
PHI D-DOT	RAD/SEC*2	0.0	0.0	0.0	0.0	0.0
PSI DOT		0.0	0.0	0.0	0.0	0.0
PSI D-DOT		0.0	0.0	0.0	0.0	0.0
ALPHA FRL	DEG	1.571	5.323	-3.611	5.452	-3.709
BETA	DEG	0.0	0.0	0.0	0.0	0.0
GEAR POSITION		UP	UP	UP	UP	UP
SLAT DEFL.		0	0	0	0	0
FLAP DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
AILERON RIGGING*	DEG	2.000	-15.000	11.000	0.0	0.0
AILERON 1 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
AILERON 2 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 1 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 2 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 3 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 4 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 5 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
SPOILER 6 DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
HOR. STAB. INCID.	DEG	-1.213	-4.401	4.191	-4.728	4.534
ELEVATOR DEFL.	DEG	-0.295	-2.982	0.425	-3.455	0.471
RUDDER DEFL.	DEG	0.0	0.0	0.0	0.0	0.0
GUST VELOCITY	FT/SEC	0.0	0.0	0.0	0.0	0.0
FUEL VENT PRESSURE	PSI	3.00/-2.00	3.00/-2.00	3.00/-2.00	3.00/-2.00	3.00/-2.00
CABIN PRESSURE	PSI	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50
PZ-TOT. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0	0.0
PZ-EXP. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0	0.0
H.M. STAB. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M. ELEV. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M.-INBAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
PZ-TOT. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0	0.0
PZ-EXP. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0	0.0
H.M. STAB. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M. ELEV. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M.-INBAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
PY (VT+DUCT) (NET)	1000 LB	0.0	0.0	0.0	0.0	0.0
H.M. RUDDER (NET)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
PY-INTERFACE (NET)	1000 LB	0.0	0.0	0.0	0.0	0.0
PY (VT+DUCT) (AIR)	1000 LB	0.0	0.0	0.0	0.0	0.0
HM RUDDER (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0	0.0
PY INTERFACE (AIR)	1000 LB	0.0	0.0	0.0	0.0	0.0
COLUMN NUMBER IN STACKED MATRIX		111	112	113	114	115

*For ACS "ON"

TABLE 3.4-4 (cont.) Load condition parameters

PARAMETER SHEET FOR STATIC FLIGHT LOADS

CONDITION NUMBER		1102/ 2	1102/ 3	1102/ 4	1102/ 5
CONDITION DESCRIPTION		PSM,VA	NSM,VA	PSM,VA	NSM,VA
GROSS WEIGHT	LB	504000.000	504000.000	504000.000	504000.000
FUEL WEIGHT	LB	166000.000	166000.000	166000.000	166000.000
FS (C.G.)	0/0 MAC	17.130	17.130	17.130	17.130
WL (C.G.)	IN	190.100	190.100	190.100	190.100
AIRSPEED	KEAS	316.000	316.000	316.000	316.000
MACH NUMBER		0.478	0.478	0.478	0.478
ALTITUDE	1000 FT	0.0	0.0	0.0	0.0
POWER SETTING		0	0	0	0
NX (C.G.)	INERTIA	0.263	0.035	0.265	0.036
NY (C.G.)	LOAD	0.0	0.0	0.0	0.0
NZ (C.G.)	FACTOR	-2.500	1.000	-2.500	1.000
THETA DOT		0.127	-0.121	0.127	-0.121
THETA D-DOT	RAD/SEC	0.0	0.0	0.0	0.0
PHI DOT	AND	0.0	0.0	0.0	0.0
PHI D-DOT	RAD/SEC*2	0.0	0.0	0.0	0.0
PSI DOT		0.0	0.0	0.0	0.0
PSI D-DOT		0.0	0.0	0.0	0.0
ALPHA FRL	DEG	11.853	-6.272	11.909	-6.321
BETA	DEG	0.0	0.0	0.0	0.0
GEAR POSITION		UP	UP	UP	UP
SLAT DEFL.		0	0	0	0
FLAP DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON RIGGING*	DEG	-15.000	11.000	0.0	0.0
AILERON 1 DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 1 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 3 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 4 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 5 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 6 DEFL.	DEG	0.0	0.0	0.0	0.0
HOR. STAB. INCID.	DEG	-6.998	6.381	-7.475	7.040
ELEVATOR DEFL.	DEG	-7.471	0.717	-8.509	0.805
RUDDER DEFL.	DEG	0.0	0.0	0.0	0.0
GUST VELOCITY	FT/SEC	0.0	0.0	0.0	0.0
FUEL VENT PRESSURE	PSI	3.00/-2.00	3.00/-2.00	3.00/-2.00	3.00/-2.00
CABIN PRESSURE	PSI	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50
PZ-TOT. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PZ-TOT. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. RUDDER (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PY-INTERFACE (NET)	1000 LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (AIR)	1000 LB	0.0	0.0	0.0	0.0
HM RUDDER (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY INTERFACE (AIR)	1000 LB	0.0	0.0	0.0	0.0
COLUMN NUMBER IN STACKED MATRIX		122	123	124	125

*For ACS "ON"

TABLE 3.4-4 (cont.) Load condition parameters

PARAMETER SHEET FOR STATIC FLIGHT LOADS

CONDITION NUMBER		1103/ 2	1103/ 3	1103/ 4	1103/ 5
CONDITION DESCRIPTION		PSM,VC	NSM,VC	PSM,VC	NSM,VC
GROSS WEIGHT	LB	504000.000	504000.000	504000.000	504000.000
FUEL WEIGHT	LB	166000.000	166000.000	166000.000	166000.000
FS (C.G.)	0/0 MAC	17.130	17.130	17.130	17.130
WL (C.G.)	IN	190.100	190.100	190.100	190.100
AIRSPEED	KEAS	356.000	356.000	356.000	356.000
MACH NUMBER		0.820	0.820	0.820	0.820
ALTITUDE	1000 FT	21.300	21.300	21.300	21.300
POWER SETTING		0	0	0	0
NX (C.G.)	INERTIA	0.088	-0.005	0.093	-0.004
NY (C.G.)	LOAD	0.0	0.0	0.0	0.0
NZ (C.G.)	FACTOR	-2.500	1.000	-2.500	1.000
THETA DOT		0.080	-0.076	0.080	-0.076
THETA D-DOT	RAD/SEC	0.0	0.0	0.0	0.0
PHI DOT	AND	0.0	0.0	0.0	0.0
PHI D-DOT	RAD/SEC*2	0.0	0.0	0.0	0.0
PSI DOT		0.0	0.0	0.0	0.0
PSI D-DOT		0.0	0.0	0.0	0.0
ALPHA FRL	DEG	8.117	-4.746	8.239	-4.840
BETA	DEG	0.0	0.0	0.0	0.0
GEAR POSITION		UP	UP	UP	UP
SLAT DEFL.		0	0	0	0
FLAP DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON RIGGING*	DEG	-15.000	11.000	0.0	0.0
AILERON 1 DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 1 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 3 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 4 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 5 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 6 DEFL.	DEG	0.0	0.0	0.0	0.0
HOR. STAB. INCID.	DEG	-5.517	4.830	-5.843	5.175
ELEVATOR DEFL.	DEG	-4.599	0.511	-5.072	0.577
RUDDER DEFL.	DEG	0.0	0.0	0.0	0.0
GUST VELOCITY	FT/SEC	0.0	0.0	0.0	0.0
FUEL VENT PRESSURE	PSI	3.00/-2.00	3.00/-2.00	3.00/-2.00	3.00/-2.00
CABIN PRESSURE	PSI	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50
PZ-TOT. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PZ-TOT. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. RUDDER (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PY-INTERFACE (NET)	1000 LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (AIR)	1000 LB	0.0	0.0	0.0	0.0
HM RUDDER (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY INTERFACE (AIR)	1000 LB	0.0	0.0	0.0	0.0
COLUMN NUMBER IN STACKED MATRIX		132	133	134	135

*For ACS "ON"

TABLE 3.4-4 (cont.) Load condition parameters

PARAMETER SHEET FOR STATIC FLIGHT LOADS

CONDITION NUMBER		1104/ 2	1104/ 3	1104/ 4	1104/ 5
CONDITION DESCRIPTION		PSM,VD	NSM,VD	PSM,VD	NSM,VD
GROSS WEIGHT	LB	504000.000	504000.000	504000.000	504000.000
FUEL WEIGHT	LB	166000.000	166000.000	166000.000	166000.000
FS (C.G.)	O/O MAC	17.130	17.130	17.130	17.130
WL (C.G.)	IN	190.100	190.100	190.100	190.100
AIRSPEED	KEAS	418.000	418.000	418.000	418.000
MACH NUMBER		0.880	0.880	0.880	0.880
ALTITUDE	1000 FT	17.300	17.300	17.300	17.300
POWER SETTING		0	0	0	0
NX (C.G.)	INERTIA	-0.079	-0.086	-0.072	-0.086
NY (C.G.)	LOAD	0.0	0.0	0.0	0.0
NZ (C.G.)	FACTOR	-2.500	0.0	-2.500	0.0
THETA DOT		0.073	-0.035	0.073	-0.035
THETA D-DOT	RAD/SEC	0.0	0.0	0.0	0.0
PHI DOT	AND	0.0	0.0	0.0	0.0
PHI D-DOT	RAD/SEC*2	0.0	0.0	0.0	0.0
PSI DOT		0.0	0.0	0.0	0.0
PSI D-DOT		0.0	0.0	0.0	0.0
ALPHA FRL	DEG	5.289	-0.921	5.439	-1.033
BETA	DEG	0.0	0.0	0.0	0.0
GEAR POSITION		UP	UP	UP	UP
SLAT DEFL.		0	0	0	0
FLAP DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON RIGGING*	DEG	-15.000	11.000	0.0	0.0
AILERON 1 DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 1 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 3 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 4 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 5 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 6 DEFL.	DEG	0.0	0.0	0.0	0.0
HOR. STAB. INCID.	DEG	-4.509	1.536	-4.622	1.637
ELEVATOR DEFL.	DEG	-3.138	0.071	-3.302	0.085
RUDDER DEFL.	DEG	0.0	0.0	0.0	0.0
GUST VELOCITY	FT/SEC	0.0	0.0	0.0	0.0
FUEL VENT PRESSURE	PSI	3.00/-2.00	3.00/-2.00	3.00/-2.00	3.00/-2.00
CABIN PRESSURE	PSI	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50
PZ-TOT. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PZ-TOT. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. RUDDER (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PY-INTERFACE (NET)	1000 LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (AIR)	1000 LB	0.0	0.0	0.0	0.0
HM RUDDER (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY INTERFACE (AIR)	1000 LB	0.0	0.0	0.0	0.0
COLUMN NUMBER IN STACKED MATRIX		142	143	144	145

*For ACS "ON"

TABLE 3.4-4 (cont.) Load condition parameters

PARAMETER SHEET FOR STATIC FLIGHT LOADS

CONDITION NUMBER		1105/ 2	1105/ 3	1105/ 4	1105/ 5
CONDITION DESCRIPTION		FLAP EXT	FLAP EXT	FLAP EXT	FLAP EXT
GROSS WEIGHT	LB	504000.000	504000.000	504000.000	504000.000
FUEL WEIGHT	LB	166000.000	166000.000	166000.000	166000.000
FS (C.G.)	0/0 MAC	17.130	17.130	17.130	17.130
WL (C.G.)	IN	190.100	190.100	190.100	190.100
AIRSPEED	KEAS	220.000	220.000	220.000	220.000
MACH NUMBER		0.330	0.330	0.330	0.330
ALTITUDE	1000 FT	0.0	0.0	0.0	0.0
POWER SETTING		0	0	0	0
NX (C.G.)	INERTIA	0.231	-0.079	0.226	-0.079
NY (C.G.)	LOAD	0.0	0.0	0.0	0.0
NZ (C.G.)	FACTOR	-2.000	0.0	-2.000	0.0
THETA DOT		0.130	-0.087	0.130	-0.087
THETA D-DOT	RAD/SEC	0.0	0.0	0.0	0.0
PHI DOT	AND	0.0	0.0	0.0	0.0
PHI D-DOT	RAD/SEC*2	0.0	0.0	0.0	0.0
PSI DOT		0.0	0.0	0.0	0.0
PSI D-DOT		0.0	0.0	0.0	0.0
ALPHA FRL	DEG	10.675	-10.177	10.539	-10.156
BETA	DEG	0.0	0.0	0.0	0.0
GEAR POSITION		UP	UP	UP	UP
SLAT DEFL.		FULL	FULL	FULL	FULL
FLAP DEFL.	DEG	28.000	28.000	28.000	28.000
AILERON RIGGING*	DEG	-19.330	3.330	0.0	0.0
AILERON 1 DEFL.	DEG	0.0	0.0	0.0	0.0
AILERON 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 1 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 2 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 3 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 4 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 5 DEFL.	DEG	0.0	0.0	0.0	0.0
SPOILER 6 DEFL.	DEG	0.0	0.0	0.0	0.0
HOR. STAB. INCID.	DEG	-6.983	8.480	-7.813	8.779
ELEVATOR DEFL.	DEG	-7.437	0.997	-9.244	1.037
RUDDER DEFL.	DEG	0.0	0.0	0.0	0.0
GUST VELOCITY	FT/SEC	0.0	0.0	0.0	0.0
FUEL VENT PRESSURE	PSI	0.0	0.0	0.0	0.0
CABIN PRESSURE	PSI	8.835/-0.50	8.835/-0.50	8.835/-0.50	8.835/-0.50
PZ-TOT. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PZ-TOT. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
PZ-EXP. STAB (AIR)	1000 LB	0.0	0.0	0.0	0.0
H.M. STAB. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M. ELEV. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-INBAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
H.M.-OUTAIL. (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (NET)	1000 LB	0.0	0.0	0.0	0.0
H.M. RUDDER (NET)	1000 FT-LB	0.0	0.0	0.0	0.0
PY-INTERFACE (NET)	1000 LB	0.0	0.0	0.0	0.0
PY (VT+DUCT) (AIR)	1000 LB	0.0	0.0	0.0	0.0
HM RUDDER (AIR)	1000 FT-LB	0.0	0.0	0.0	0.0
PY INTERFACE (AIR)	1000 LB	0.0	0.0	0.0	0.0
COLUMN NUMBER IN STACKED MATRIX		152	153	154	155

*For ACS "ON"

3.5 Finite Element Structural Analysis

This section describes the general procedure used for finite element structural analysis of the Preliminary Aeroelastic Design System (PADS) baseline airplane. The finite element structural model is used to determine internal loads, internal loads per unit external load, stresses and the deflections of the airframe. In addition, this model is also used to calculate a set of structural influence coefficients (SICs) at selected locations for aeroelastic analyses, and a set of stiffness derviatives for sizing the structure for flutter, deflection and other aeroelastic constraints. Specific elements of the model are sized for the basic strength level loads but outside of the NASTRAN system. The finite element analysis is performed using the Lockheed-California Company's version of the NASTRAN finite element analysis system, reference 2. The analysis is performed using Calac developed rigid format 145 (RF145), which performs a standard static solution. The model is not analyzed using substructure-coupling or Super element analysis methodology.

The structural analysis consists of four steps.

* Model definition
* Structural influence coefficient (SIC) calculation
* External load transformation
* Static solution

3.5.1 Model definition - This section describes the general procedure used to define the finite element structural model of the Preliminary Aeroelastic Design System (PADS) baseline airframe model.

3.5.1.1 General arrangement - The finite element model of the airframe is divided into several regions which are then coupled using multipoint constraints. The center fuselage, center wing and outer wing are 3-D models using axial, bending, and membrane elements. The forward fuselage, aft fuselage, horizontal stabilizer, and vertical stabilizer are represented by a series of 2-D bar (bending) elements. The PADS finite element models represent the left hand half of the aircraft. A schematic representation of the Baseline finite element model is shown in figure 3.5.1.

A list of the PANVALET names containing model grid point and element decks for the baseline airplane is given in Appendix E. These bulk data represents a complete model definition in the form of NASTRAN input card images, sorted in alphanumeric sequence. A brief description of a representative list of the bulk data cards is given below. A more detailed description of all the bulk data cards is given in reference 2.

* GRID cards:

    a) Define the location of each model grid point.

    b) Specify the permanent constraints required for each point stability.

* CBAR, CROD, and CMEMQ cards - define the grid point connectivity for beam, axial, and quadrilateral membrane panel elements, respectively. CBAR cards also define the bending plane and pin end options for the bending element.

* PBAR, PROD, and PMEMQ cards - define the section properties for the bar (Ix, Iy, A, and J), axial (A), and membrane panel (t, Ast) elements respectively.

* COORDxx cards - define the coordinate systems used for various major parts of the airframe structure.

* MATx cards - define the material properties (E, G, and nu etc. for isotropic material and E11,E12,E22,E33 etc. for anisotropic material) for the elements.

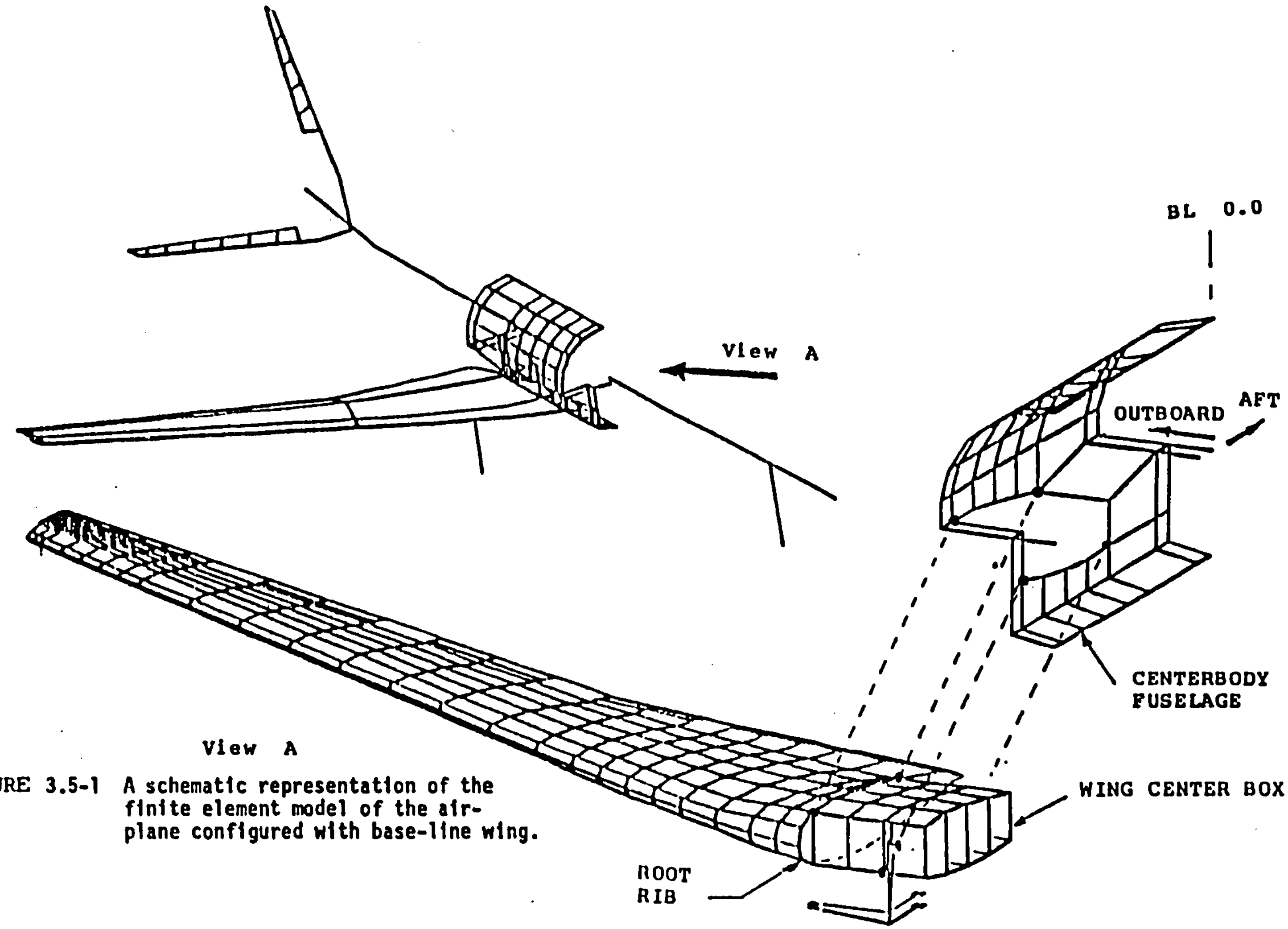


FIGURE 3.5-1 A schematic representation of the finite element model of the airplane configured with base-line wing.

* MPC, MPCA and MPCADD cards - MPC and MPCA define the multipoint constraints between grid points, which permit specified displacements in the model to be dependent functions of other displacements. The Calac developed MPCA cards add displacement equations to MPC cards. MPCADD cards are used to add different MPC sets to be used in the analysis.

* SPC, SPC1 and SPCADD cards - are used, in addition to GRID cards, to specify grid point constraints. SPCADD cards are used to add different SPC sets needed in the analysis. These cards are also used to apply symmetric or antisymmetric boundary conditions to the structure.

* LDREF, LGROUP, LMAT cards - These Calac developed enhancements are used to define and calculate the load distribution vectors between SIC points and structural nodes. These cards are also used to calculate the reduced stiffness matrix for the structural model.

3.5.1.2 Coordinate system definition and usage - A total of five different coordinate systems, including BASIC (0), are used to define all the parts of the airframe model. The general positive direction for the majority of the coordinate systems is given below:

x axis is positive aft

y axis is positive left

z axis is positive up

A list of various airframe regions and the coordinate system used to define these regions is given in Table 3.5-1.

Coordinate System	Defined Region
0	Forward and Aft Fuselage
11	Center Fuselage and Center Wing Box
12	Wing (Outboard of the Root Rib)
61	Horizontal Stabilizer
71	Vertical Stabilizer

Table 3.5-1 Coordinate Systems

** The coordinate systems 13 and 14 are defined in the bulk data, however these are not used.

The advantage of using different coordinate systems for various parts of the structural model is that the model defined in a specific coordinate system can be translated and rotated and reanalyzed by redefining that coordinate system.

3.5.1.3 Grid point generation, definition and identification

The grid point locations are calculated, generated, and formed into card image format by executing a lifting surface generator module. A flow chart of the lifting surface generator module is given in figure 3-5.2. This module consists of two programs, WBONES and SLICE. WBONES defines the basic wing geometry data which includes orientation, station identification, station location, bone line specification, and size of the mesh to be generated. SLICE is an Aerodynamics coded program modified to generate surface coordinates for the wing and empennage of the finite element model. An example of the two input datasets to WBONES is given in figures 3.5-3 and 3.5-4 respectively. Input to SLICE consists of planform geometry data, airfoil shape parameters, and slicing information generated by WBONES. No confinement is imposed on the capability of this generator and, therefore, all lifting surfaces and airfoil shapes can be rapidly generated as long as the airfoil parameters are defined. Only minimum effort is required to incorporate additional structural components, such as main landing gear, pylon attached points, etc., to the finite element model. This methodology is based on vector analysis. An example of the input datasets to SLICE is given in Figures 3.5-5 through 3.5-7. Figure 3.5-6 represents an example of a FLO-22 data set as provided from the airfoil selection process as described in section 3.2.

The mesh generator (QUILT) is a batch process module, used to generate connection cards for the finite element model. This program can generate grid point coordinates as well as connectivities, however, QUILT is only used to generate element connection cards. Its use in conjunction with the aforementioned lifting surface generator to generate surface coordinates for wing and empennage results in a high quality finite element model with a precise representation of the airfoil section.

The model node numbering system is an integral part of the data management and documentation of the analysis. The numbering system chosen for the NASTRAN model uses as its basic element a coded grid point number. The coded grid point number consists of the model station of the grid point and a local grid point number. For the majority of the grid points on the wing, an odd number identifies a point on the upper surface and an even number identifies a point on the lower surface. On the center fuselage structure, an odd grid point represents the left hand side of the airplane.

Coded grid point number

GID = MS + LN (4 digit number)

Where:

GID = Grid point identification number.

MS = Model station.

LN = Local grid point number (2 digit number).

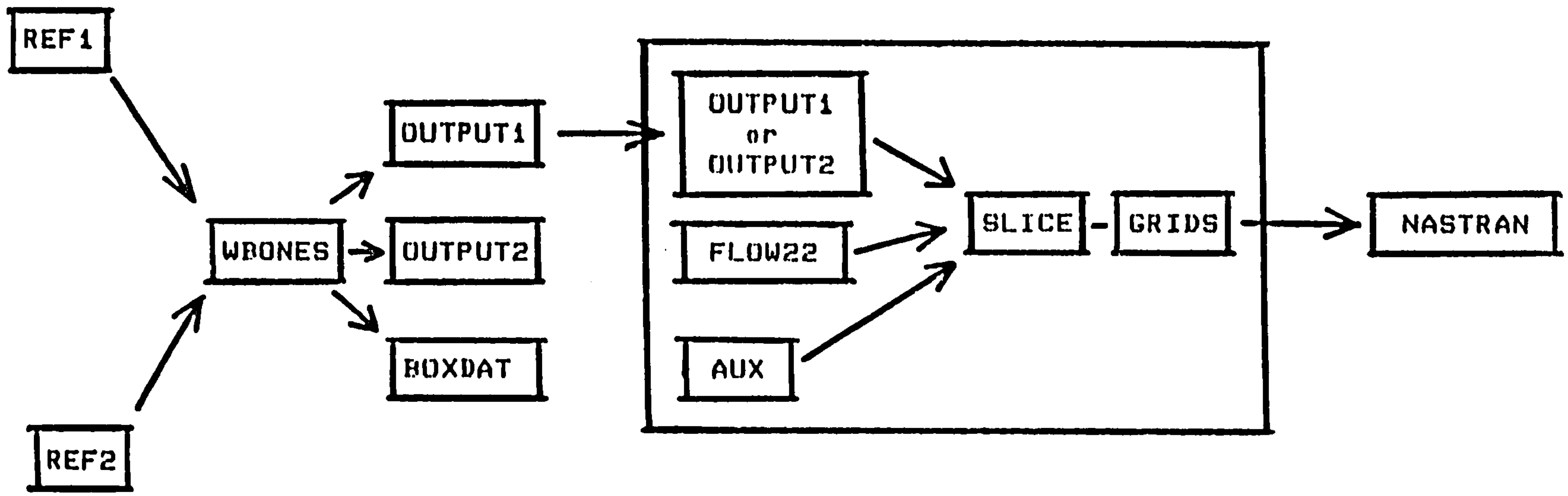


FIGURE 3.5-2 Flow Chart of the Lifting Surface Generator

```
POINT,11,,  0.0   ,  0.0   ,0.0
POINT,12,, 60.06  ,115.73  ,0.0
POINT,13,,242.34  ,470.44  ,0.0
POINT,14,,576.73  ,1121.17,0.0
POINT,21,, 93.83  ,  0.0   ,0.0
POINT,22,, 93.83  ,115.73  ,0.0
POINT,23,,610.49  ,1121.17,0.0
POINT,31,,313.09  ,  0.0   ,0.0
POINT,32,,313.09  ,115.73  ,0.0
POINT,33,,316.16  ,127.66  ,0.0
POINT,34,,340.52  ,223.02  ,0.0
POINT,35,,378.63  ,372.17  ,0.0
POINT,36,,403.75  ,470.47  ,0.0
POINT,37,,584.12  ,944.36  ,0.0
POINT,38,,634.93  ,1077.83,0.0
POINT,39,,651.42  ,1121.17,0.0
POINT,41,,480.91  ,  0.0   ,0.0
POINT,42,,480.91  ,115.73  ,0.0
POINT,43,,483.22  ,470.44  ,0.0
POINT,44,,688.20  ,1121.17,0.0
POINT,51,,437.24  ,  0.0   ,0.0
POINT,52,,434.05  ,115.73  ,0.0
POINT,53,,424.34  ,470.44  ,0.0
POINT,54,,595.85  ,944.37  ,0.0
POINT,55,,660.33  ,1121.17,0.0
END
```

FIGURE 3.5-3 An example of the Input Dataset 'REF1' to WBONES

```
WING,DOF=12,N=3,N1=11,N2=41
SPARS,FWD
LE,11,14
FS,21,22,23
RS,31,32,36,39
HL,51,52,53,54,55
TE,41,43,44
END
STATIONS
MS,2500,32,,,0.0,PT,,1.0
MS,2300,33,,,0.0,PT,RS2,1.0
MS,2100,34,,,0.0,PT,RS2,1.0
MS,1700,35,,,0.0,PT,RS2,1.0
MS,1500,36,,,0.0,PT,RS3,1.0
MS, 800,37,,,0.0,PT,RS3,1.0
MS, 300,38,,,0.0,PT,RS3,1.0
MS, 100,39,,,0.0,PT,,1.0
MS,2400,,,,,,,
MSB,2400,1,-100,2500,2300,1.0
MS,2200,,,,RS2,A,,
MSB,2200,1,-100,2300,2100,1.0
MS,2000,,,,RS2,A,,
MSB,2000,3,-100,2100,1700,1.0
MS,1600,,,,RS2,A,,
MSB,1600,1,-100,1700,1500,1.0
MS,1400,,,,RS3,A,,
MSB,1400,6,-100,1500, 800,1.0
MS, 700,,,,RS3,A,,
MSB, 700,4,-100, 800, 300,1.0
MS, 200,,,,,,,
MSB, 200,1,-100, 300, 100,1.0
END
NTASK
SICBOX,M=13,N=8,IDO=151,I1=1,IM=9;
KYN,14,44,13,43
DSTM,1,6,1121.17,1076.97,1032.77,988.57,944.37,825.89
DSTM,7,13,707.41,588.92,470.44,421.31,372.17,243.95,115.73
DSTN,1,8,0.0,3.0,10.0,30.0,55.0,75.0,88.0,100.0
NTASK
SICBOX,I1=9,IM=13;
KYN,13,43,12,42
NTASK
NOMORE
```

FIGURE 3.5-4 An Example of the Input Dataset 'REF2' to WBONES

```
 25    10 0 T F 'FWD'     7
        0.000000       115.730000        2500
 0.00000 0.02032 0.08129 0.21120 0.34111 0.47102 0.60093 0.88760 0.96066 1.00000
       14.336076        85.448032        2300
 0.00000 0.01959 0.07834 0.20192 0.32550 0.44908 0.57266 0.87900 0.95765 1.00000
       14.336076       186.906334        2100
 0.00000 0.02218 0.08872 0.21290 0.33707 0.46125 0.58542 0.85309 0.94858 1.00000
       14.336076       345.596867        1700
 0.00000 0.02800 0.11200 0.23752 0.36303 0.48854 0.61405 0.79498 0.92824 1.00000
       20.837869       440.278430        1500
 0.00000 0.03321 0.13284 0.25722 0.38159 0.50596 0.63034 0.72730 0.90455 1.00000
       20.837869       927.915361         800
 0.00000 0.05640 0.22562 0.33311 0.44061 0.54811 0.65560 0.73775 0.90821 1.00000
       20.837869      1064.364996         300
 0.00000 0.06876 0.27505 0.37015 0.46525 0.56035 0.65545 0.73754 0.90814 1.00000
        0.000000      1121.170000         100
 0.00000 0.07572 0.30286 0.39466 0.48645 0.57825 0.67005 0.74998 0.91249 1.00000
       14.336076       136.177183        2200
 0.00000 0.02080 0.08321 0.20707 0.33092 0.45478 0.57864 0.86685 0.95340 1.00000
       14.336076       226.578967        2000
 0.00000 0.02339 0.09358 0.21803 0.34248 0.46694 0.59139 0.84097 0.94434 1.00000
       14.336076       266.251600        1900
 0.00000 0.02475 0.09900 0.22376 0.34853 0.47330 0.59806 0.82744 0.93960 1.00000
       14.336076       305.924233        1800
 0.00000 0.02627 0.10510 0.23021 0.35533 0.48045 0.60556 0.81222 0.93428 1.00000
       14.336076       397.891169        1600
 0.00000 0.03066 0.12264 0.24877 0.37489 0.50101 0.62713 0.76843 0.91895 1.00000
       20.837869       512.665492        1400
 0.00000 0.03657 0.14627 0.27366 0.40105 0.52845 0.65584 0.74312 0.91009 1.00000
       20.837869       581.873804        1300
 0.00000 0.03883 0.15533 0.28045 0.40557 0.53069 0.65581 0.74251 0.90988 1.00000
       20.837869       651.082115        1200
 0.00000 0.04140 0.16560 0.28815 0.41069 0.53324 0.65578 0.74181 0.90963 1.00000
       20.837869       720.290427        1100
 0.00000 0.04434 0.17736 0.29696 0.41655 0.53615 0.65575 0.74101 0.90936 1.00000
       20.837869       789.498738        1000
 0.00000 0.04774 0.19095 0.30714 0.42333 0.53952 0.65571 0.74009 0.90903 1.00000
       20.837869       858.707050         900
 0.00000 0.05171 0.20682 0.31903 0.43124 0.54345 0.65566 0.73902 0.90866 1.00000
       20.837869       955.205288         700
 0.00000 0.05850 0.23402 0.33941 0.44480 0.55019 0.65558 0.73765 0.90818 1.00000
       20.837869       982.495215         600
 0.00000 0.06077 0.24307 0.34619 0.44931 0.55243 0.65555 0.73762 0.90817 1.00000
       20.837869      1009.785142         500
 0.00000 0.06322 0.25286 0.35353 0.45419 0.55486 0.65552 0.73760 0.90816 1.00000
       20.837869      1037.075069         400
 0.00000 0.06587 0.26349 0.36149 0.45949 0.55749 0.65549 0.73757 0.90815 1.00000
        7.168038       100.589016        2400
 0.00000 0.01995 0.07982 0.20656 0.33331 0.46005 0.58679 0.88330 0.95915 1.00000
       10.418935      1092.767498         200
 0.00000 0.07224 0.28896 0.38240 0.47585 0.56930 0.66275 0.74376 0.91032 1.00000
END
```

FIGURE 3.5-5   An example of the input dataset 'OUTPUT1' to SLICE.

```
WING W55,  WITH FINITE FUSELAGE, 30% TRANS.

FNX       FNY       FNZ       FPLOT     AFPLOT    FCONT     FNMAX
64.       8.        12.       0.2       0.1       0.        4.0
FIT       COV       P10       P20       P30       FHALF     FPMAX
 100.     1.E-06    1.6       1.0       1.0       1.0       0.
 50.      1.E-06    1.6       1.0       1.0       0.0       1.
 30.      1.E-06    1.6       1.0       1.0       0.0       1.
 20.      1.E-06    1.6       1.0       1.0       0.0       1.
FMACH     YA        AL        FDSGN
0.82      0.        -.09      0.
ZSYM      FNC
1.        9.
SWEEP1    SWEEP2    SWEEP3    DIHED1    DIHED2    DIHED3
27.3053   27.3053   27.3053   5.5       5.5       5.5
SREF/2    CREF      XREF      BO2
1.633     .61935    .15466    2.78
BLAYER    PINF      TINF      SCALE     XFIX      YFIX
1.        1350.     490.      1.0       .30       .30
XBOD      YBOD      ZBOD      RBOD      TL        ALF       ALA
2.361     .15365     .2973     .33566    18.       3.        15.
MODEL     ZL        ZMU       ISTORE
0.         0.        0.        0.
ZS(1)     XL        YL        CHORD     THICK     AINC      NEWSEC
115.73    .1652     0.        1.05814   1.        4.57      1.
METHOD
3.
XSING      YSING     NPTS      FWDEXP    FWDEXL    ZTEU      ZTEL
.015      0.        50.        .5        .5        .014     .006
XCRES      YCRES    XM         YM        THETUS    AFTEXU
.26        .078     .5         -.16      14.036    3.0
XCRESL     YCRESL   XML        YML       THETLS    AFTEXL
.30        -.060    .35        .300      5.5       3.0
FLAG      S1        S2        DEL1      DEL2
0.
ZS(2)     XL        YL        CHORD     THICK     AINC      NEWSEC
194.093   .27706    .020766   .94628    1.        3.71      1.
METHOD
3.
XSING      YSING     NPTS      FWDEXP    FWDEXL    ZTEU      ZTEL
.0135     0.        50.        .5        .5        .0112    .0032
XCRES      YCRES    XM         YM        THETUS    AFTEXU
.310       .068     .75        -.16      14.036    3.0
XCRESL     YCRESL   XML        YML       THETLS    AFTEXL
.30        -.060    .35        .300      5.5       3.0
FLAG      S1        S2        DEL1      DEL2
0.
ZS(3)     XL        YL        CHORD     THICK     AINC      NEWSEC
224.234   .32008    .028754   .90326    1.        3.45      1.
```

FIGURE 3.5-6 An example of the Input Dataset 'FLO-22' to SLICE

```
1
5
1.0
2
60.060,   115.73
576.73,  1121.177
7
480.9100,  115.73,  0.0
482.1100,  300.00,  0.0
483.2100,  469.44,  0.0
483.2200,  470.44,  0.0
483.5350,  471.44,  0.0
587.0310,  800.00,  0.0
688.2000, 1121.17,  0.0
```

FIGURE 3.5-7 An example of the input dataset 'AUX' to SLICE.

Table 3.5-2 defines the range of grid cards used for representation of various regions of the finite element model.

TABLE 3.5-2 Grid Point Identification	
Grid Point Identification	Model Region
99 - 3000	Wing (Inner and Outer)
5000 - 5799	Center Fuselage
6001 - 6019	Horizontal Stabilizer
7001 - 7015	Vertical Stabilizer
8000 - 8241	Forward and Aft Fuselage

The grid points ending with 51 through 67 between grid points 99 and 2600 are used as Structural Influence Coefficient (SIC) nodes. These nodes are used to transfer external loads to the structural model and are connected to the model through multipoint constraints.

This numbering scheme enables the user to identify the location of the airframe structure from the grid point identification.

3.5.1.4 Element identification and definition - The following load-carrying NASTRAN elements are used in the finite element model:

ROD elements

BAR elements

MEMQ elements

ELAS4 elements

Each element is defined on a CROD, CBAR, CMEMQ or CELAS4 element connection card. The connection cards contain grid point identification numbers which are related through a geometrical transformation matrix to the degrees of freedom associated with the element. Each connection card, with the exception of the CELAS4 card, references a PROD, PBAR or PMEMQ element property card that provides the cross-section (flexibility) data for the element. If the property card identification field (field 3) is blank on the element connection card then the element property identification is the same as the element connection identification designated in field 2 of the element connection card. Each property card in turn references a material card that defines the material properties for that element. A detailed explanation of the entries on these cards can be obtained from Reference 2.

ROD Elements:

The NASTRAN ROD element, as used in the finite element model, transmits a constant axial force between two grid points. These elements are used extensively in the wing region to represent wing surface stringers, rib posts and spar posts. These elements are also used to represent center wing fore and aft stringer members. The ROD element internal load output is constant axial force (lb) and stress (psi) in the element.

BAR Elements:

The NASTRAN BAR element load-carrying capability, as used in the finite element model, includes transmission of a constant axial load, bending in two perpendicular planes and a torsional moment. The NASTRAN BAR element is used exclusively to represent the forward and aft fuselage and the horizontal and vertical stabilizers. The BAR element is also used to represent frames and floor beams in the center fuselage.

The NASTRAN BAR element internal force output includes constant axial force (lb) in the element, bending moments (in-lb) at both ends of the element and the associated constant transverse shear force in one or both bending planes.

The output also includes an internal torsional moment (in-lb) whenever applicable. The internal stress output includes axial stress (psi) in the element.

ELAS4 Elements:

This is standard NASTRAN bulk data defined element for scalar spring property and connection. It defines a scalar element of the structual model which is connected only to the scalar points without reference to a property value. ELAS4 is used primarily to represent the control surfaces actuator springs.

MEMQ Elements:

The Calac developed MEMQ element is a warped semi-monocoque quadrilateral membrane element. This element dispenses with the usual stringer lumping and delumping process. In the element formulation, the basic skin is assumed to be of homogeneous anisotropic material, uniform thickness, and in a state-of-plane stress. Attached to the skin are two sets of nonorthogonal reinforcing members in the plane of the skin. Effects of any offsets are neglected in this element. In the case where the element is rectangular and the stiffeners are parallel to the edges, then the shear flow is constant and the normal stresses are constant in the normal direction but vary linearly in the transverse direction. The element has a capability to transmit inplane forces (shear and axial). The MEMQ element also has a capability to separate internal stringer forces and forces in the skin panels. This element is used to represent wing surface panels, spar webs, rib webs and the wing carry-through surfaces. This element is also used to represent the center fuselage floors and surfaces.

The NASTRAN internal force and stress output for the MEMQ element consists of the following:

At the center of the element

Nx, Ny, Nxy	Total force per unit length in the local element coordinate system or other specified output coordinate system.
Fstgr1, Fstgr2	Stringer force per unit length in the local element coordinate system or other specified output coordinate system.

Sigx, Sigy, Tauxy	Normal stress and shear stress in the local element coordinate system or other specified output coordinate system.
SIG1, SIG2	Maximum and Minimum principal skin stresses.
Taumax	Maximum skin shear stress at 45 deg. to SIG1.
Sigs1, Sigs2	Normal stress in the set 1 and 2 stiffeners.
Alpha	Angle between the element X axis and maximum principal stress.
Sigx1, Sigy1, Tauxy1	Skin stresses at an arbitrary location and direction.

Element Identification Numbers

The element numbers are composed of an element-type identification code followed by the grid point number of either the first or second point specified on the element connection card. The general rule for numbering of elements for the model is as follows:

Coded element identification number

EID = (I)(10000) + GID (5 digit number)

where:

I = Element type identification code

I = 1 is used for wing surface panels (EID's ending with an odd number identify elements on the upper surface and EID's ending with an even number identify elements on the lower surface (MEMQ elements)). This element identification code is also used to include surface panels for the center Fuselage.

I = 2 is mainly used for wing rib panels (MEMQ elements). This code is also used for the longitudinal stringers in the center fuselage (ROD elements).

I = 3 is used for wing spar webs (MEMQ elements).

I = 4 is used for chordwise axial members (rib caps) represented by ROD elements.

I = 6 is used for spanwise axial members ROD elements.

I = 8 is used for flexible BAR elements in the 2-D and 3-D fuselage and 2-D vertical and horizontal stabilizers.

I = 9 is used for rigid BAR and rigid ROD elements.

An example of a typical rib and a typical fuselage frame is given in figures 3.5-8 and 3.5-9 respectively.

The grid point number (GID) used in the definition of the element identification number (EID) in the preceding formula is obtained from the general rules given below.

* For element type I = 1, wing surface panels, the GID is the outboard, forward located grid point (See figure 3.5-10). For the fuselage surface panels, the GID is the upper, forward located grid point.

* For element type I = 2, wing rib panels (MEMQ elements), the GID is the upper, forward grid point (See figure 3.5-8). For the longitudinally oriented ROD elements in the center fuselage, the GID is the forward located grid point.

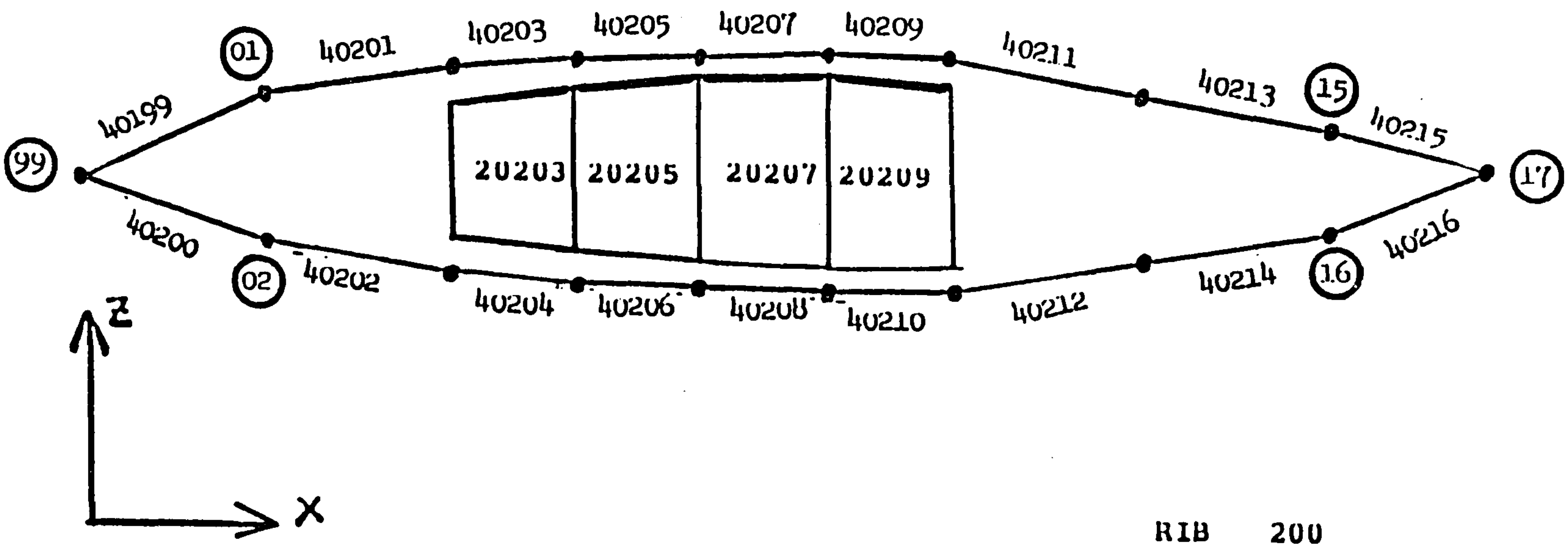


FIGURE 3.5-8 An example of node numbering system and element numbering system for a typical wing rib.

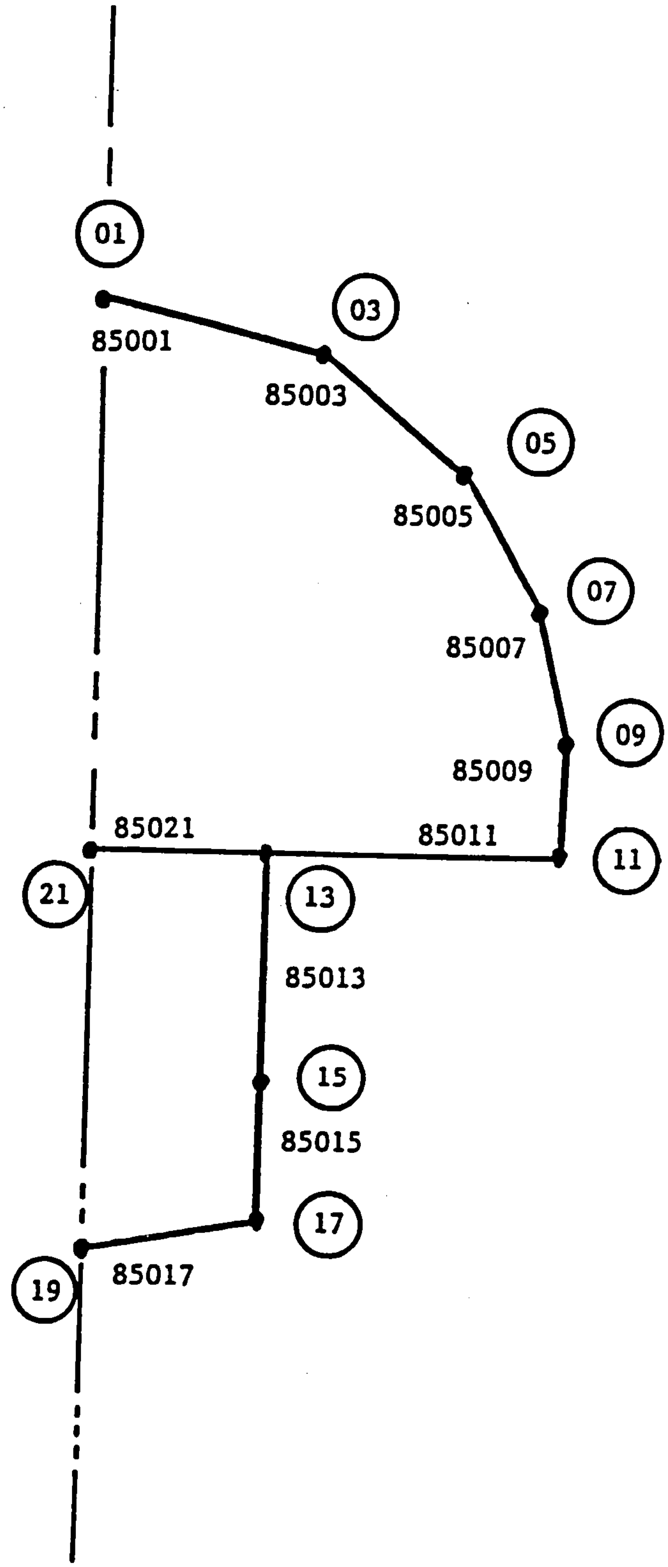


FIGURE 3.5-9 An Example of Node Numbering System and Element Numbering System for a Typical Fuselage Frame

* For element type I = 3, wing spar web elements, the GID is the upper, outboard grid point (See figure 3.5-11).

* For element type I = 4, chordwise rib caps, the GID is the forward grid point (See figure 3.5-8).

* For element type I = 6, spanwise stiffeners, the GID is the outboard grid point (See figure 3.5-12).

* For element type I = 8, longitudinal BAR elements located on the fuselage, the GID is the forward grid point. For the frame BAR elements, the GID is the upper grid point; and for the bending BAR elements in the horizontal stabilizer, the GID is the outboard grid point. For the bending BAR elements representing the vertical stabilizer, the GID is the upper grid point. See figures 3.5-9, 3.5-13 and 3.5-14 for the examples of these numbering schemes.

3.5.1.5 Sign convention -

3.5.1.5.1 Displacement sign convention - The displacement output for each grid point consists of three translations and three rotations in the displacement coordinate system. The displacement coordinate system is designated in field 7 of the grid card. The majority of the grid points in the structural model have the basic coordinate system as the displacement coordinate system. All model drawings and displacement plots for the finite element model are normally plotted in the left-hand coordinate system.

3.5.1.5.2 Element sign convention -

ROD elements: Positive element internal force output represents tension in the element. Negative element internal force output represents compression in the element.

BAR Elements: The BAR element bending plane definition, the positive force convention, and the bending sign convention are illustrated in figure 3.5-15. In the figure, the grid points

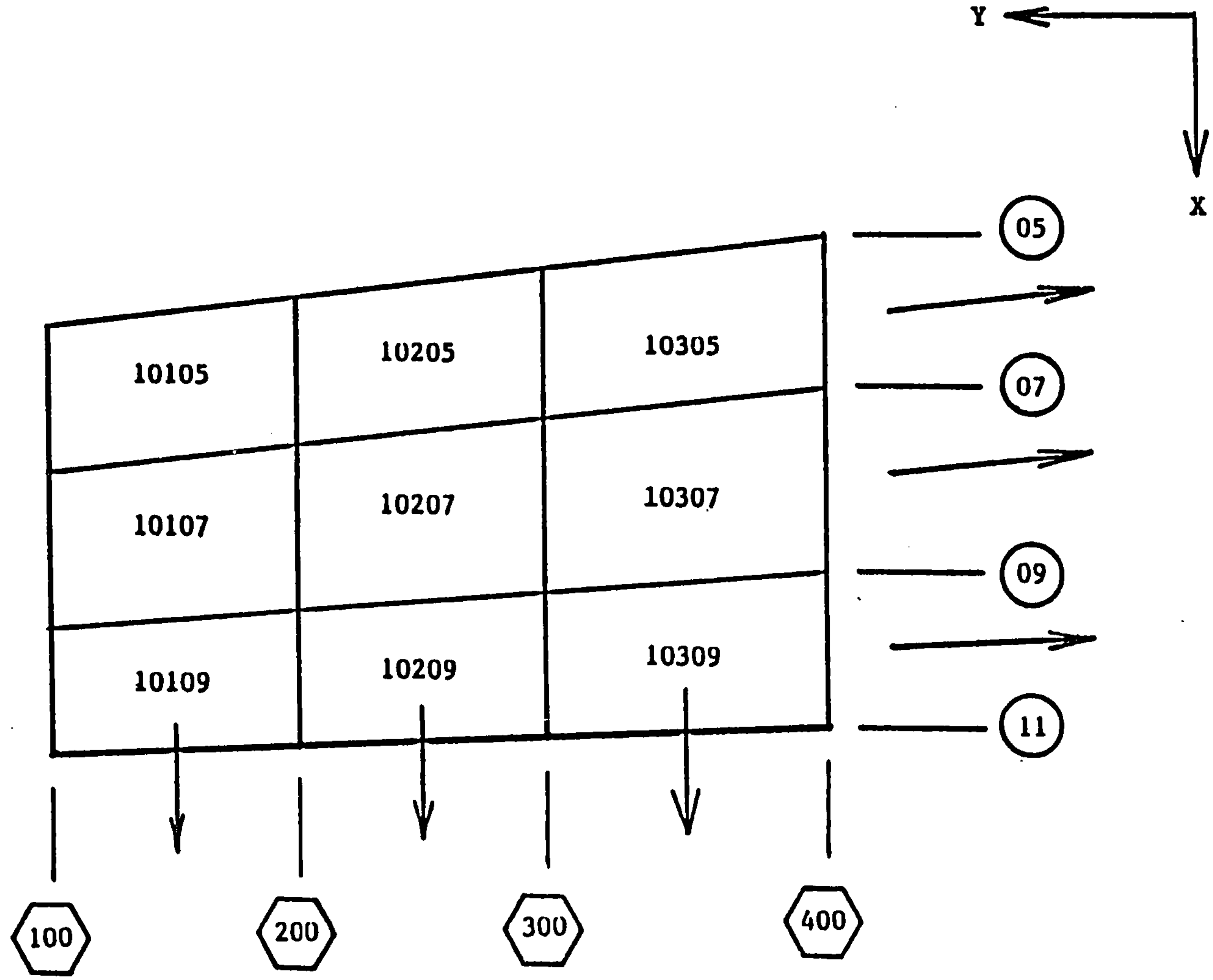


FIGURE 3.5-10 An example of element numbering system for wing upper surface panel (MEMQ) elements.

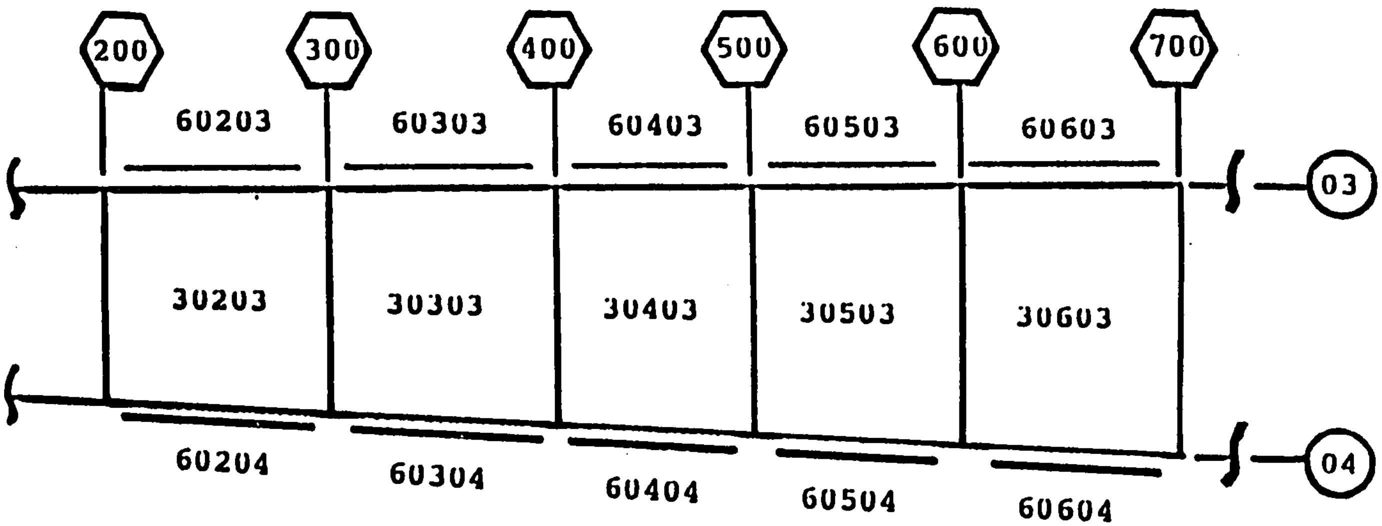


FIGURE 3.5-11 An example of element numbering system for wing spar axial (ROD) and panel (MEMQ) elements

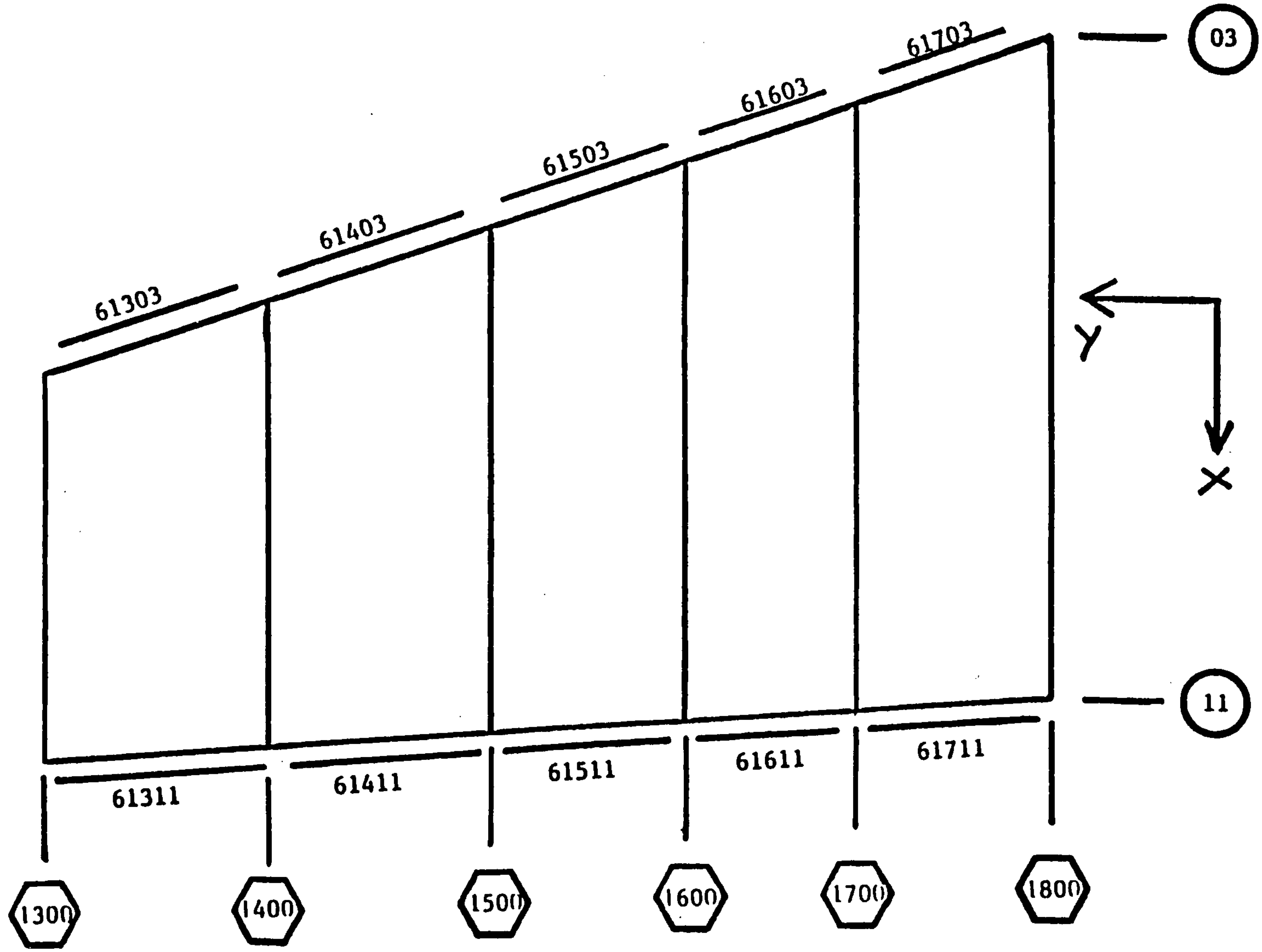


FIGURE 3.5-12 An example of element numbering system for wing spanwise stiffeners (ROD elements).

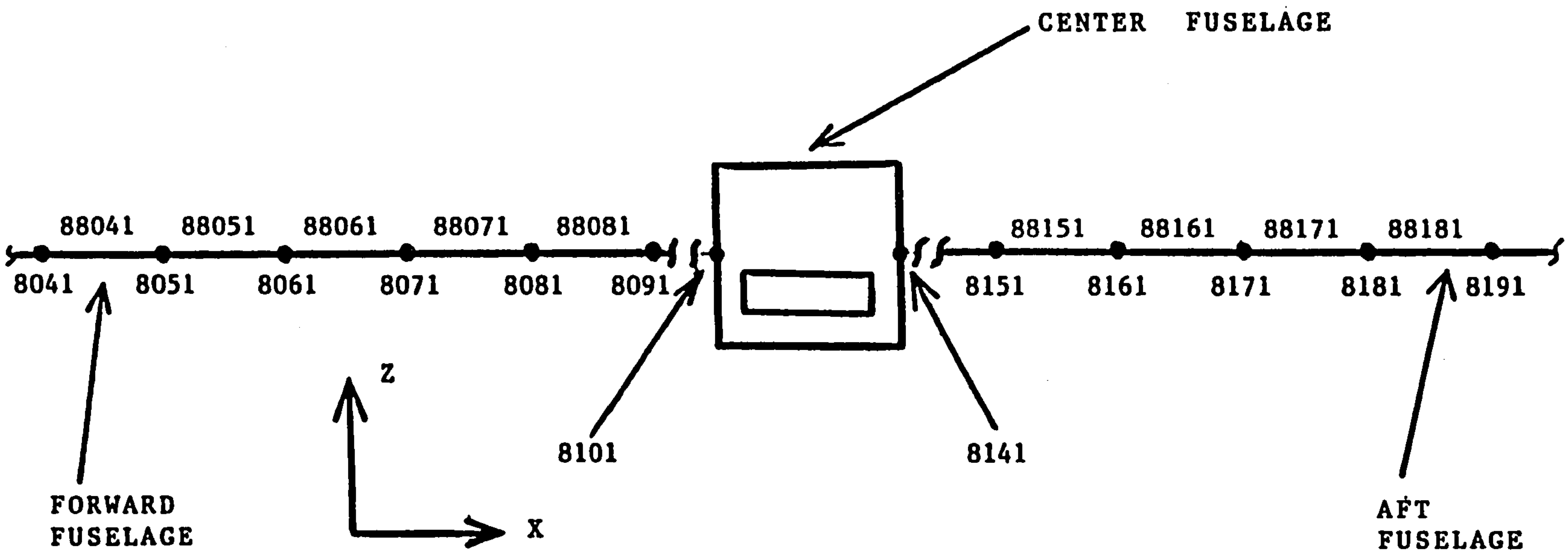


FIGURE 3.5-13 An example of element numbering system for bending elements (BAR) in forward and aft fuselage

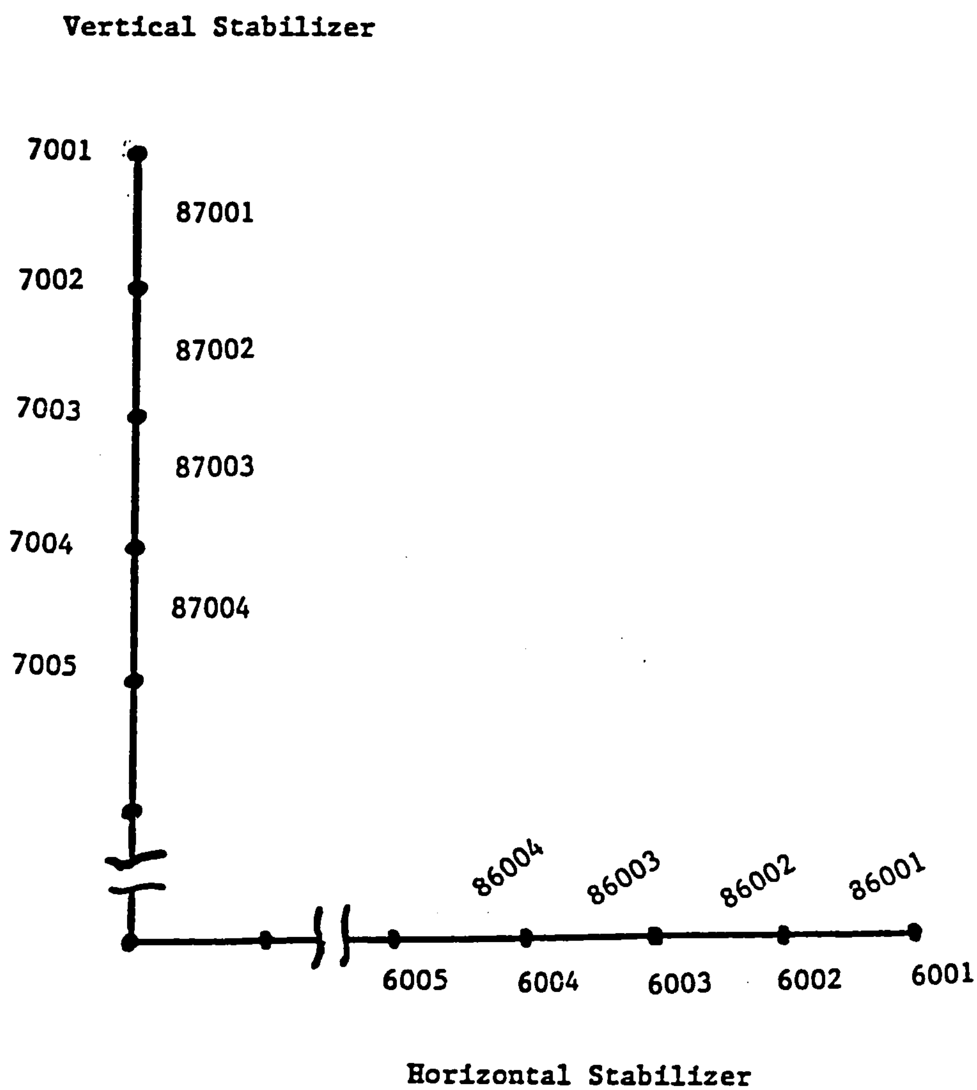


FIGURE 3.5-14 An example of element numbering system for bending elements (BAR) on the vertical and horizontal stabilizers

A, B, and C refer to the sequence in which the grid points are specified on the CBAR card. The vector AB defines the element x-axis, where AB is defined as the vector from grid point A to grid point B. The first bending plane (Plane 1) is defined by the element x-axis and the vector AC which is generally defined by specifying a third Reference grid point C, but can be specified directly with three vector components originating from grid point A. The second bending plane (Plane 2) is defined by the vector cross products (AB x AC) and the element x-axis. The specification of vector AC also establishes positive sign convention for the BAR element bending forces.

The definition of positive bending, as given in figure 3.5-15, is generally applied to the fuselage models in such a way that positive bending moments in the output of BAR elements located on the fuselage surface produce tension in the outside surface and compression on the inside. In a similar fashion, positive bending moments in BAR elements representing floor bending are generally specified to produce tension in the upper surface and compression in the lower surface of the floor. BAR elements representing floor post bending are specified to produce positive bending moments that put the outboard side of the element in tension.

MEMQ Elements: The positive sign convention for the output of the MEMQ element is given in figure 3.5-16. The normal forces and stresses are positive in the tension direction and negative in the compression direction.

3.5.2 Special structural modeling using multipoint constraints -

All the special structural systems, requiring rigid load transformations, are described in this section. These systems are designed to transfer load or displacement components from a reference point to the main part of the airframe structure. A brief description of each of these systems is given below. The elements that are labeled 'RIGID' in the figures are either partially or completely rigidized using MPC BAR elements.

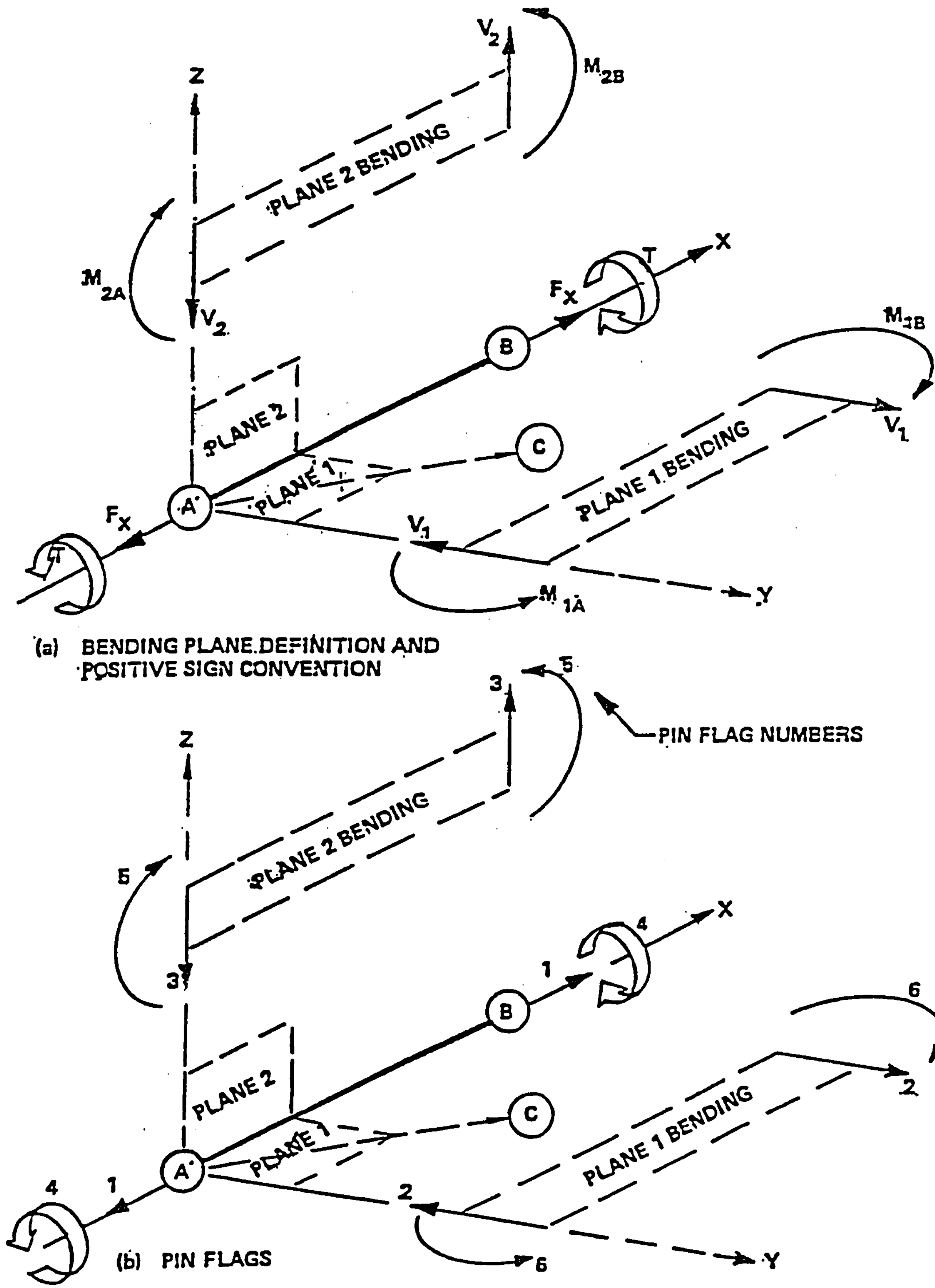


FIGURE 3.5-15 BAR Element Bending Plane Definition and Positive Force Sign Convention

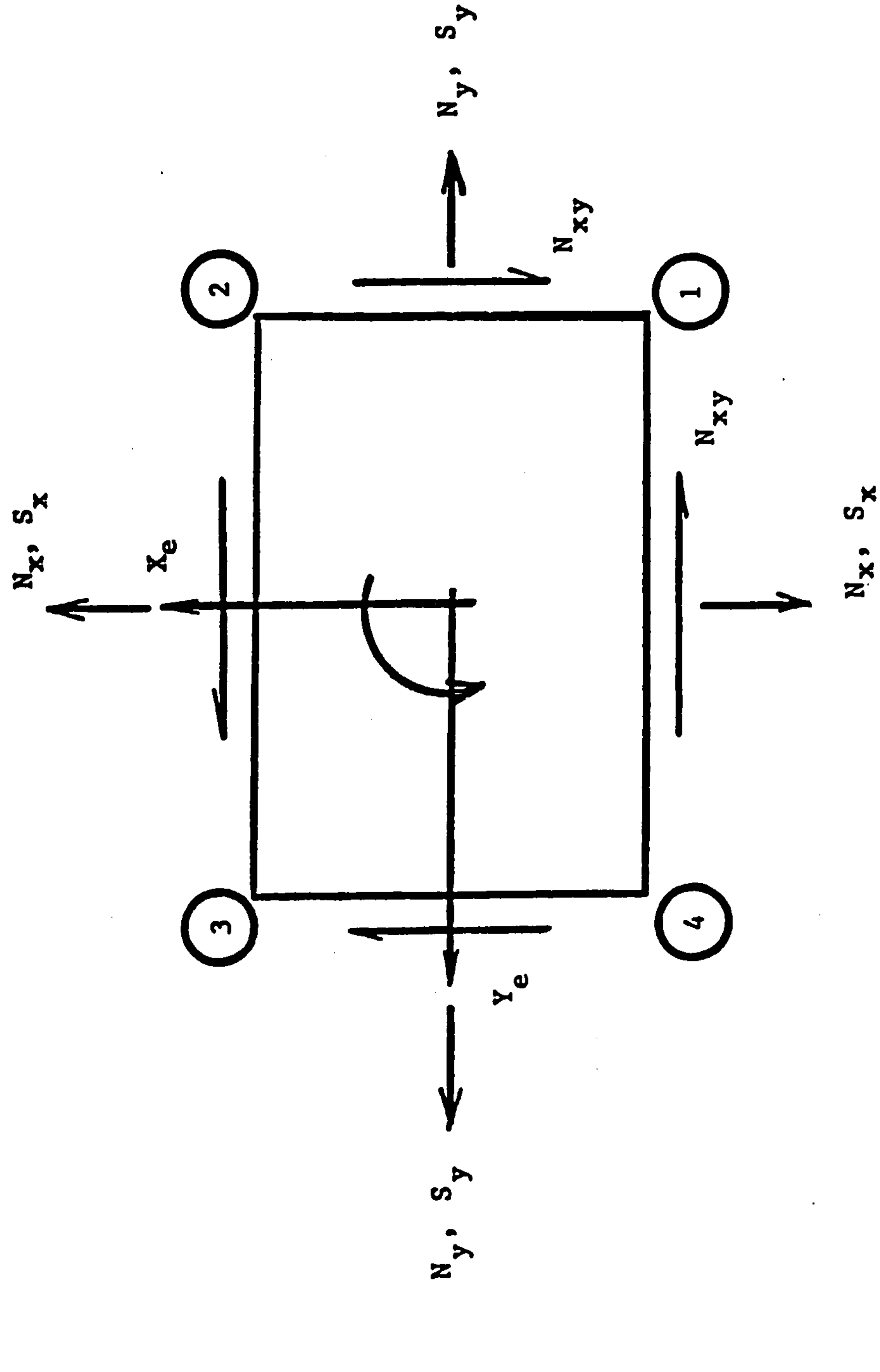


FIGURE 3.5-16 Positive sign convention for the MEMQ forces.

* Engine-Pylon Attachment:- Figure 3.5-17 shows a schematic diagram of the Engine-Pylon attachment structure. The required input data (MPC cards) to define this system are given in figure 3.5-18. A Pylon stiffness matrix is input at grid point 1894 for the baseline airplane.

* Main Landing Gear (Up and Down):- Figures 3.5-19 and 3.5-20 show a schematic diagram of the main landing gear up position and main landing gear down position respectively. Figures 3.5-21 and 3.5-22 show the input data required to rigidize these two systems.

* Nose Landing Gear (Up and Down):- Figure 3.5-23 shows a schematic diagram of the nose landing gear in both up and down positions. The required input data are shown in figure 3.5-24.

* Inboard and Outboard Actuators:- Figures 3.5-25 and 3.5-26 show a schematic diagram of inboard and outboard actuators respectively. The required input data are shown in figure 3.5-27.

* Horizontal Stabilizer Actuator:- Figure 3.5-28 shows a schematic diagram of the horizontal stabilizer actuator. The actuator degree of freedom shown in the figure as 6030. The required input data are shown in figure 3.5-29.

* Beam Fuselage to 3-D Fuselage Connections:- Figure 3.5-30 shows a schematic diagram of the connection between forward fuselage beam model and the center fuselage 3-D model. The required input data are shown in figure 3.5-31. A similar MPC connection is modeled between aft fuselage and the center fuselage. The input data required for the connection between aft fuselage and the center fuselage are shown in figure 3.5-32.

* Wing Box - Center Fuselage Connection: - Figure 3.5-33 shows a diagram of the connection between wing box and the center fuselage. Wing box grid points are connected to the fuselage grid points by rigid MPC cards. The input data required for this connection are given in figure 3.5-34.

* Horizontal Stabilizer - Control Surface Connections: - In the model, the horizontal stabilizer and the control surface have been modeled as a flexible two dimensional member. The connections between these two beams have been modeled by rigid MPC members. Figure 3.5-35 shows a schematic diagram of the horizontal stabilizer - control surface connections. The input data required for this connection are given in figure 3.5-36.

* Vertical Stabilizer - Control Surface Connections: - In the model, the vertical stabilizer and the control surface have been modeled as a flexible two dimensional member also. The connections between these two beams have been modeled by rigid MPC members. Figure 3.5-37 shows a schematic diagram of the vertical stabilizer - control surface connections. The input data required for this connection are given in figure 3.5-38.

* Main Support Structure: - The airplane model is supported at grid point 3000. This point is connected to the front spar and rear spar grid points (2503, 2511, 2803 and 2811) by a series of rigid BAR elements. Figure 3.5-39 shows a schematic diagram of the main support structure. Figure 3.5-40 shows the required MPC data for these connections.

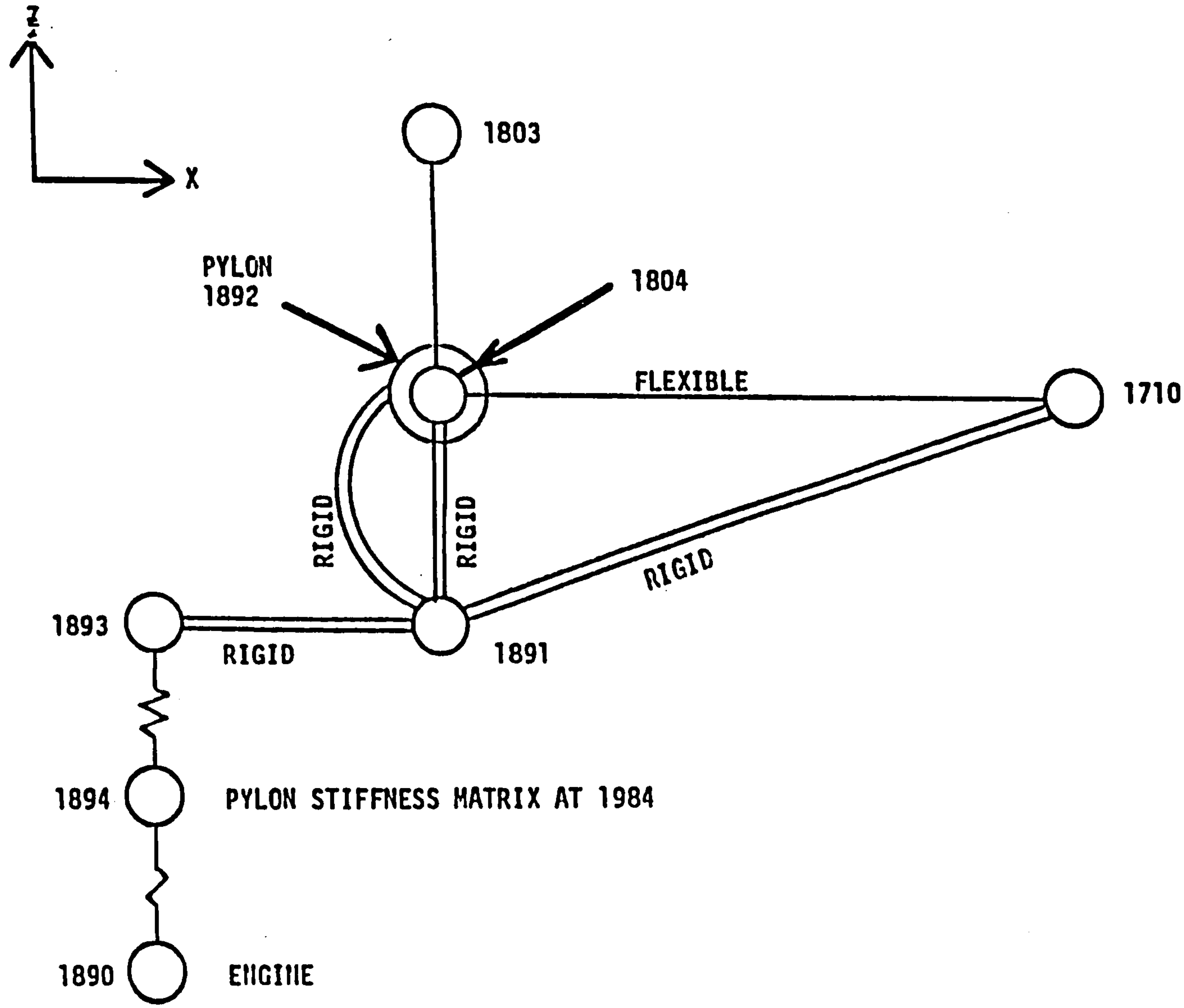


FIGURE 3.5-17 A schematic diagram of the Engine-Pylon structure modeled using RIGID BAR elements.

$ ENGINE PYLON ATTACHMENT MPCS $

CBAR	91710	999	1891	1710	0.0	0.0	10.0	1	
CROD	91805		1804	1710					
CBAR	91804	999	1891	1804	-5.0	0.0	10.0	1	
CBAR	91803	999	1804	1803	-5.0	0.0	10.0	1	
CBAR	91893	999	1893	1891	0.0	0.0	10.0	1	
CBAR	91891	999	1892	1891	10.0	0.0	0.0	1	
MPC	111	1804	3	BAR	91804	1	1		-1.000
MPC	111	1710	1	BAR	91710	1	1		-1.000
MPC	111	1804	2	BAR	91804	0	5		-1.000
MPC	111	1710	2	BAR	91710	5	0		1.000
MPC	111	1804	5	BAR	91804	6	6		-1.000
MPC	111	1804	4	BAR	91804	5	5		1.000
MPC	111	1804	6	BAR	91804	4	4		-1.000
MPC	111	1803	2	BAR	91803	5	0		1.000
MPC	111	1803	1	BAR	91803	6	0		-1.000
MPC	111	1893	1	BAR	91893	1	1		-1.000
MPC	111	1893	2	BAR	91893	0	5		-1.000
MPC	111	1893	3	BAR	91893	0	6		1.000
MPC	111	1893	4	BAR	91893	4	4		-1.000
MPC	111	1893	5	BAR	91893	6	6		-1.000
MPC	111	1893	6	BAR	91893	5	5		1.000
MPC	111	1891	1	BAR	91891	0	6		1.000
MPC	111	1891	2	BAR	91891	0	5		-1.000
MPC	111	1891	3	BAR	91891	1	1		-1.000
MPC	111	1891	4	BAR	91891	5	5		1.000
MPC	111	1891	5	BAR	91891	6	6		-1.000
MPC	111	1891	6	BAR	91891	4	4		-1.000

FIGURE 3.5-18 Input Dataset for the Engine-Pylon MPC Structure

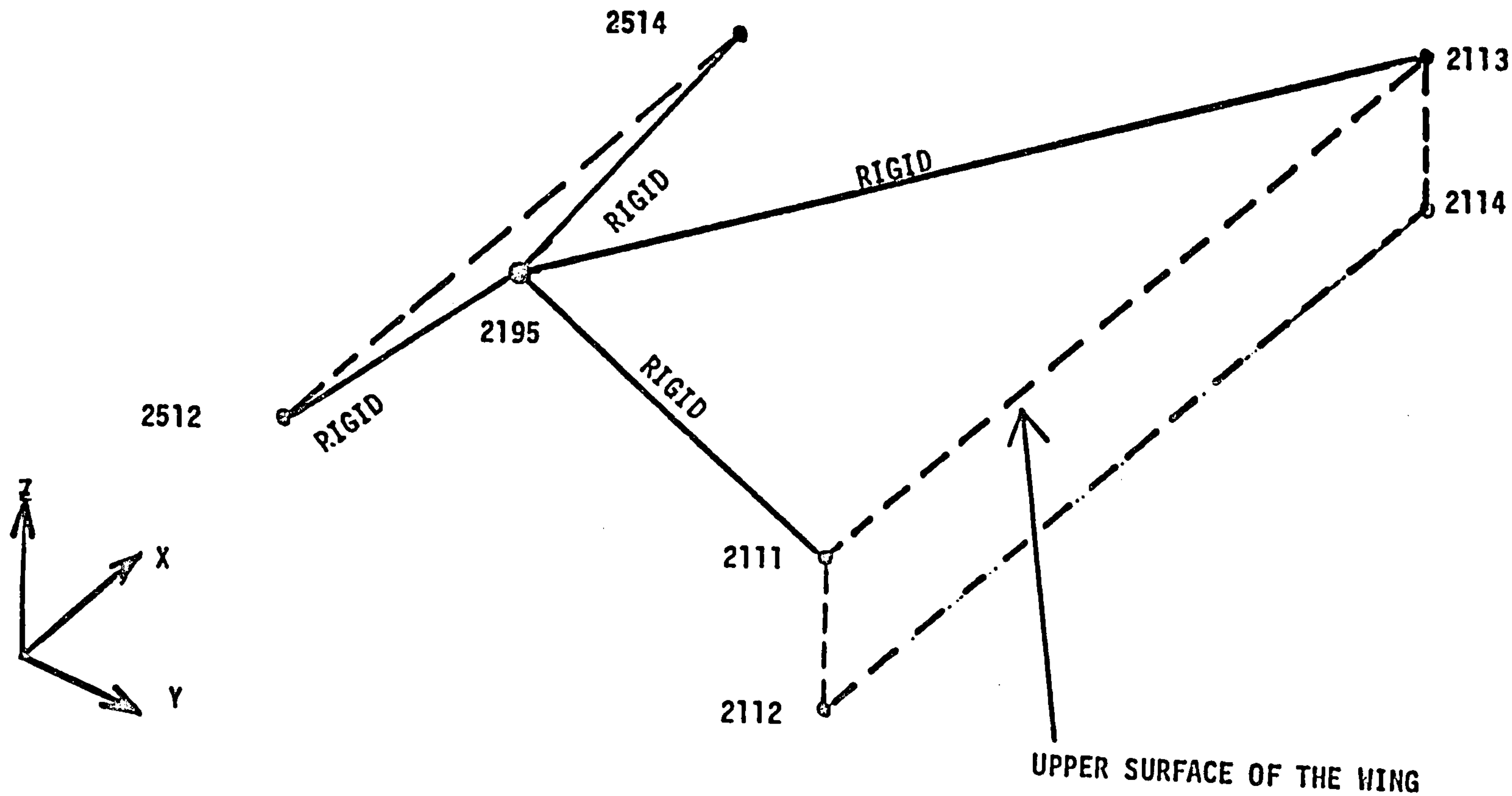


FIGURE 3.5-19 A schematic diagram of the main landing gear in the 'UP' position modeled using RIGID BAR element

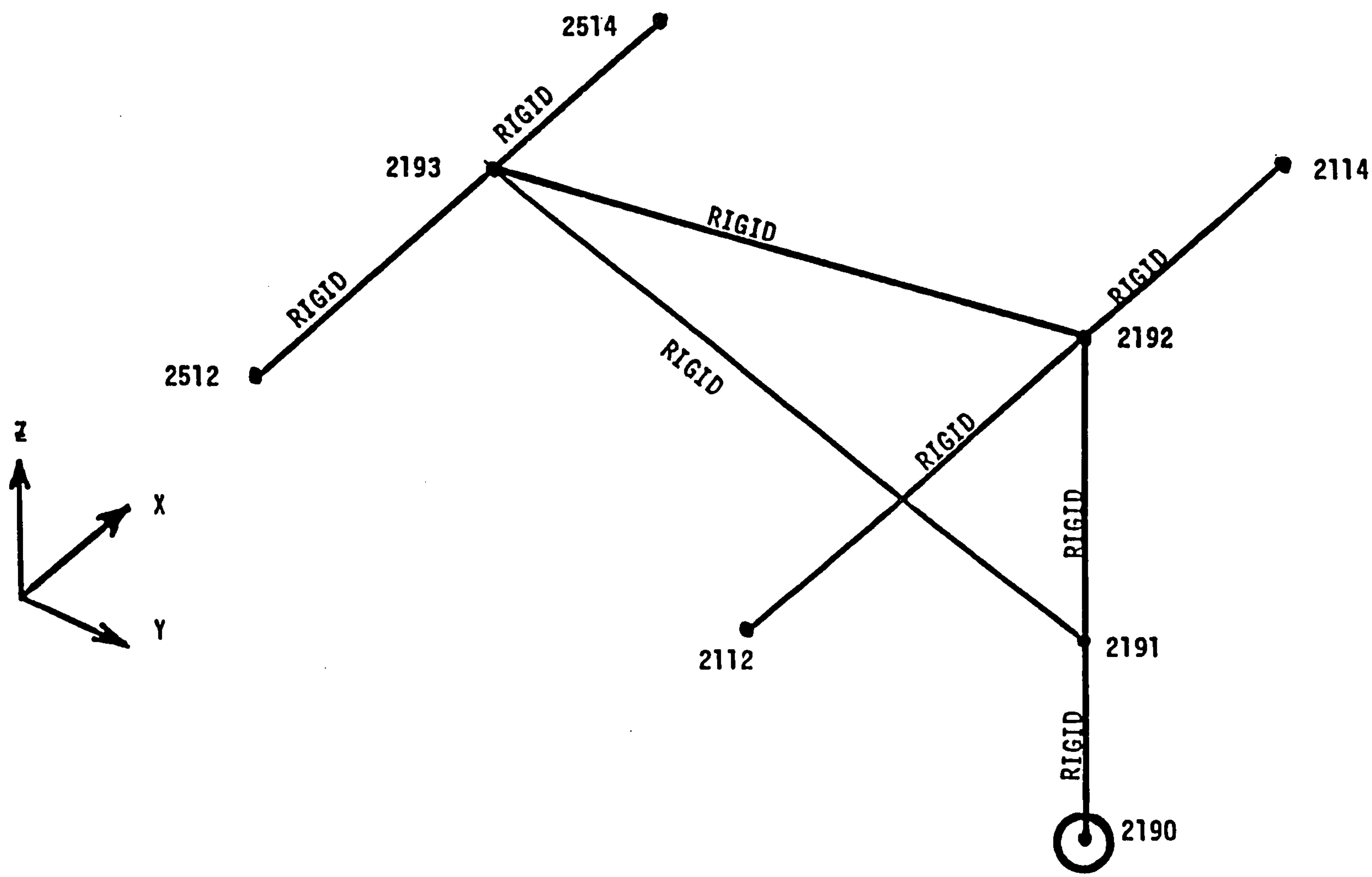


FIGURE 3.5-20 A schematic diagram of the main landing gear in the 'DOWN' position modeled, using RIGID BAR elements

$ MAIN LANDING GEAR(UP)

CBAR	92181	999	2111	2195	0.0	0.0	10.0	1	
CBAR	92182	999	2113	2195	0.0	0.0	10.0	1	
CBAR	92183	999	2512	2195	0.0	0.0	10.0	1	
CBAR	92184	999	2514	2195	0.0	0.0	10.0	1	
MPC	111	2113	1	BAR	92181	1	1		-1.000
MPCA	111	2113	1	BAR	92182	1	1		1.000
MPC	111	2111	2	BAR	92181	1	1		-1.000
MPCA	111	2111	2	BAR	92182	1	1		-1.000
MPC	111	2514	3	BAR	92183	0	6		1.000
MPCA	111	2514	3	BAR	92184	0	6		1.000
MPC	111	2113	3	BAR	92181	0	6		1.000
MPCA	111	2113	3	BAR	92182	0	6		1.000
MPC	111	2111	3	BAR	92181	0	6		1.000
MPCA	111	2111	3	BAR	92182	0	6		-1.000
MPC	111	2113	2	BAR	92181	0	5		-1.000
MPCA	111	2113	2	BAR	92182	0	5		-1.000

FIGURE 3.5-21 Input Dataset for the Main Landing Gear 'UP' Position

$ MAIN LANDING GEAR(DOWN)

```
CBAR     92190   999     2190    2191    10.0    0.0     0.0     1
CBAR     92191   999     2191    2192    10.0    0.0     0.0     1
CBAR     92112   999     2192    2112    0.0     0.0     10.0    1
CBAR     92114   999     2114    2192    0.0     0.0     10.0    1
CBAR     92193   999     2193    2191    0.0     10.0    0.0     1
CBAR     92192   999     2192    2193    0.0     0.0     10.0    1
CBAR     92512   999     2193    2512    0.0     0.0     10.0    1
CBAR     92514   999     2514    2193    0.0     0.0     10.0    1
MPC      111     2191    1       BAR     92190   0       6            1.000
MPC      111     2191    2       BAR     92190   0       5           -1.000
MPC      111     2191    3       BAR     92190   1       1           -1.000
MPC      111     2191    4       BAR     92190   5       5            1.000
MPC      111     2191    5       BAR     92190   6       6           -1.000
MPC      111     2191    6       BAR     92190   4       4           -1.000
MPC      111     2193    3       BAR     92193   1       1           -1.000
MPC      111     2193    2       BAR     92192   1       1           -1.000
MPC      111     2192    3       BAR     92191   1       1           -1.000
MPC      111     2192    1       BAR     92191   0       6            1.000
MPC      111     2192    2       BAR     92191   5       0            1.000
MPC      111     2192    5       BAR     92191   6       6           -1.000
MPC      111     2192    6       BAR     92191   4       4           -1.000
MPC      111     2192    4       BAR     92191   0       5           -1.000
MPC      111     2112    1       BAR     92112   1       1           -1.000
MPCA     111     2112    1       BAR     92114   1       1            1.000
MPC      111     2112    2       BAR     92112   5       0            1.000
MPCA     111     2112    2       BAR     92114   0       5           -1.000
MPC      111     2112    3       BAR     92112   6       0           -1.000
MPCA     111     2112    3       BAR     92114   0       6            1.000
MPC      111     2114    3       BAR     92112   6       0           -1.000
MPCA     111     2114    3       BAR     92114   0       6           -1.000
MPC      111     2114    2       BAR     92112   5       0           -1.000
MPCA     111     2114    2       BAR     92114   0       5           -1.000
MPC      111     2512    1       BAR     92512   1       1           -1.000
MPCA     111     2512    1       BAR     92514   1       1            1.000
MPC      111     2512    3       BAR     92512   6       0           -1.000
MPCA     111     2512    3       BAR     92514   0       6            1.000
MPC      111     1894    1       1.0     1890    1       -1.0                MPC18941
+PC18941         1893    1       1.0
MPC      111     1894    2       1.0     1890    2       -1.0                MPC18942
+PC18942         1893    2       1.0
MPC      111     1894    3       1.0     1890    3       -1.0                MPC18943
+PC18943         1893    3       1.0
MPC      111     1894    4       1.0     1890    4       -1.0                MPC18944
+PC18944         1893    4       1.0
MPC      111     1894    5       1.0     1890    5       -1.0                MPC18945
+PC18945         1893    5       1.0
MPC      111     1894    6       1.0     1890    6       -1.0                MPC18946
+PC18946         1893    6       1.0
```

FIGURE 3.5-22 Input Dataset for the Main Landing Gear 'DOWN' Position

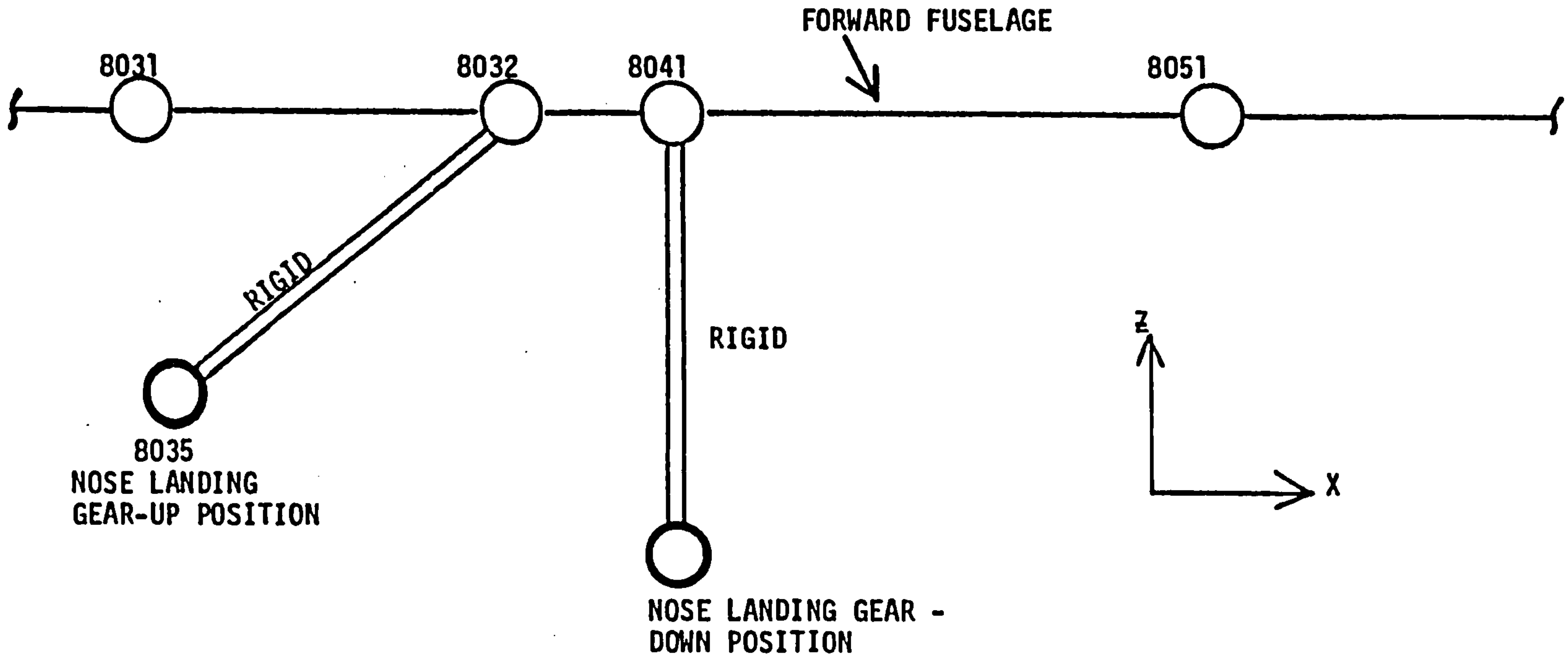


FIGURE 3.5-23 A schematic diagram of the nose landing gear in the 'UP' and 'DOWN' position modeled using RIGID BAR elements

$ NOSE LANDING GEAR(UP)

MPC	100	8032	3	BAR	98035	1	1	-1.000
MPC	105	8032	2	BAR	98035	0	5	-1.000
MPC	100	8032	1	BAR	98035	0	6	1.000
MPC	105	8032	6	BAR	98035	4	4	-1.000
MPC	105	8032	4	BAR	98035	5	5	1.000
MPC	100	8032	5	BAR	98035	6	6	-1.000

$ NOSE LANDING GEAR(DOWN)

MPC	100	8041	3	BAR	98000	1	1	-1.000
MPC	105	8041	2	BAR	98000	0	5	-1.000
MPC	100	8041	1	BAR	98000	0	6	1.000
MPC	105	8041	6	BAR	98000	4	4	-1.000
MPC	105	8041	4	BAR	98000	5	5	1.000
MPC	100	8041	5	BAR	98000	6	6	-1.000

FIGURE 3.5-24 Input dataset for the Nose-Landing gear for 'UP' and "DOWN' position.

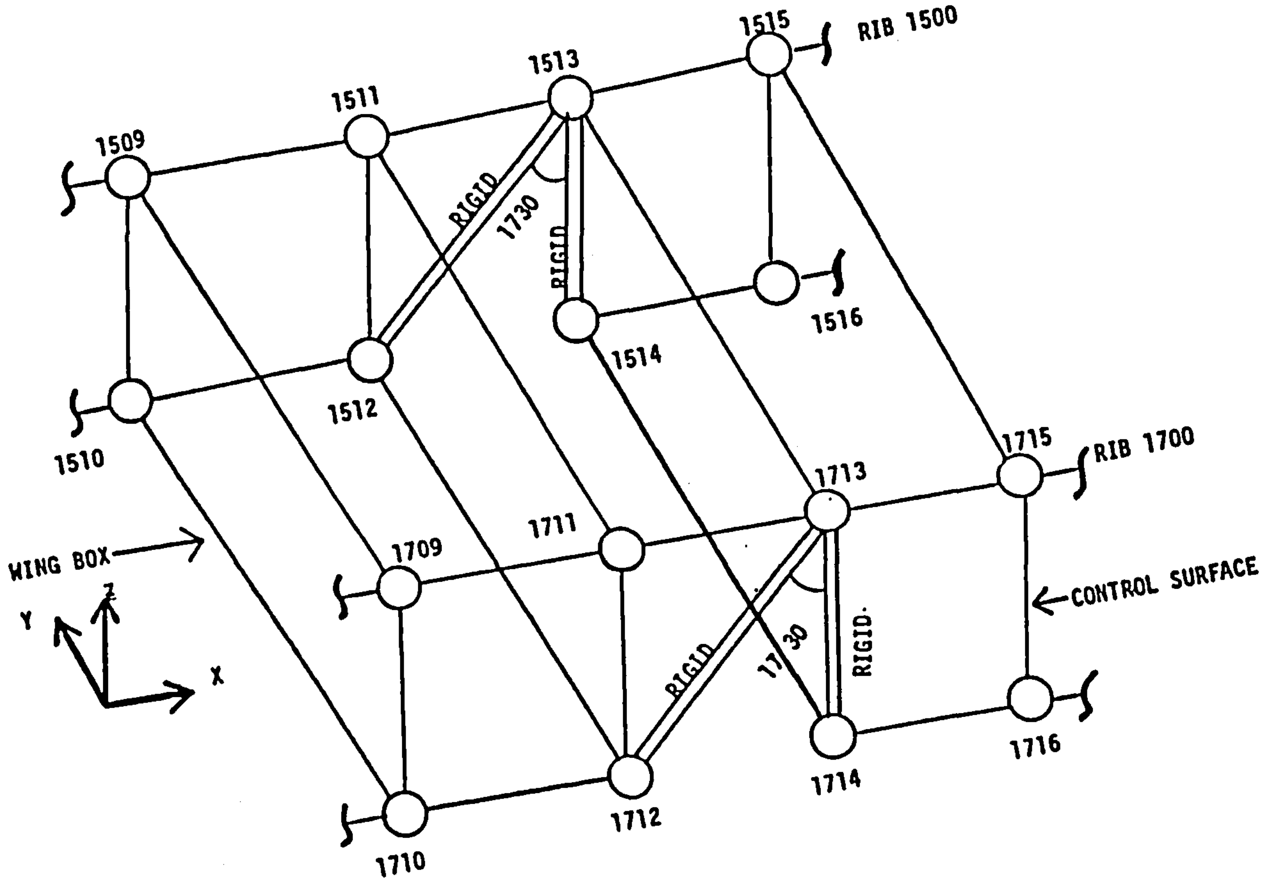


FIGURE 3.5-25 A Schematic Diagram of the Inboard Actuator Modeled Using RIGID BAR Elements and CELAS4 Elements

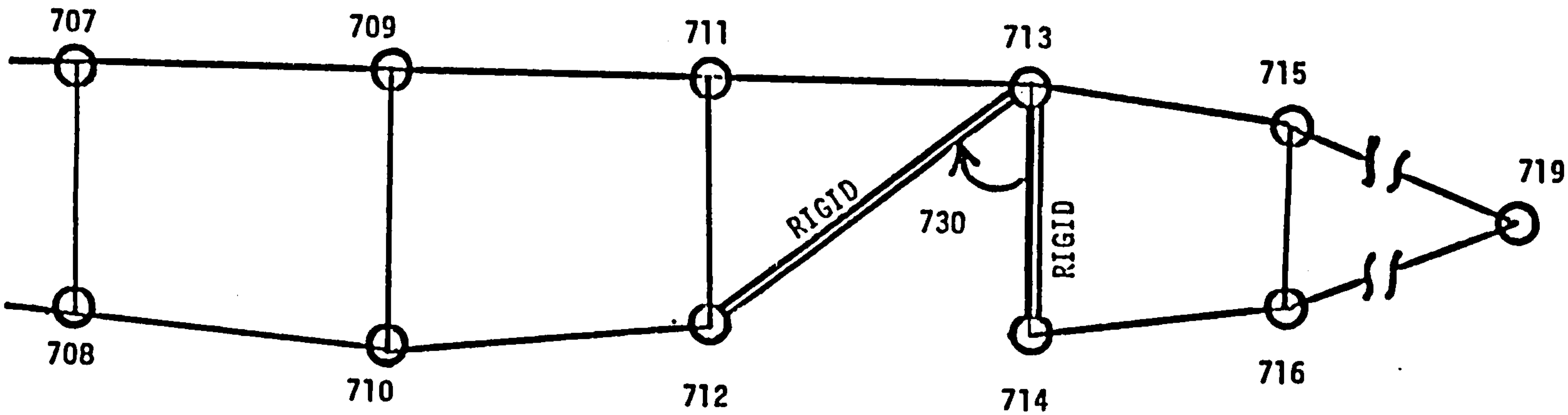


FIGURE 3.5-26 A schematic diagram of the outboard actuator modeled using RIGID BAR elements and CELAS4 elements

```
$ OUTBOARD ACTUATOR MPCS
CBAR      90712     999     712     713     714                         2
CBAR      90714     999     714     713     712                         2
SPOINT      730
MPCA        111     714       1     BAR   90712       0       6      -1.0
MPCA        111     714       1     BAR   90714       0       6      -1.0
MPC         111     714       1     0.0     730       0     1.0
CELAS4    80713  2.26+6     730


$ INBOARD ACTUATOR MPCS
CBAR      91712     999    1712    1713    1714                         2
CBAR      91512     999    1512    1513    1514                         2
CBAR      91714     999    1714    1713    1712                         2
CBAR      91514     999    1514    1513    1512                         2
SPOINT     1730
MPCA        111    1714       1     BAR   91712       0       6      -.50
MPCA        111    1714       1     BAR   91512       0       6      -.50
MPCA        111    1714       1     BAR   91514       0       6      -.50
MPCA        111    1714       1     BAR   91714       0       6      -.50
MPC         111    1714       1     0.0    1730       0     1.0
CELAS4    81713   1.7+7    1730
```

FIGURE 3.5-27 Input Data Required for the Inboard and Outboard Actuators

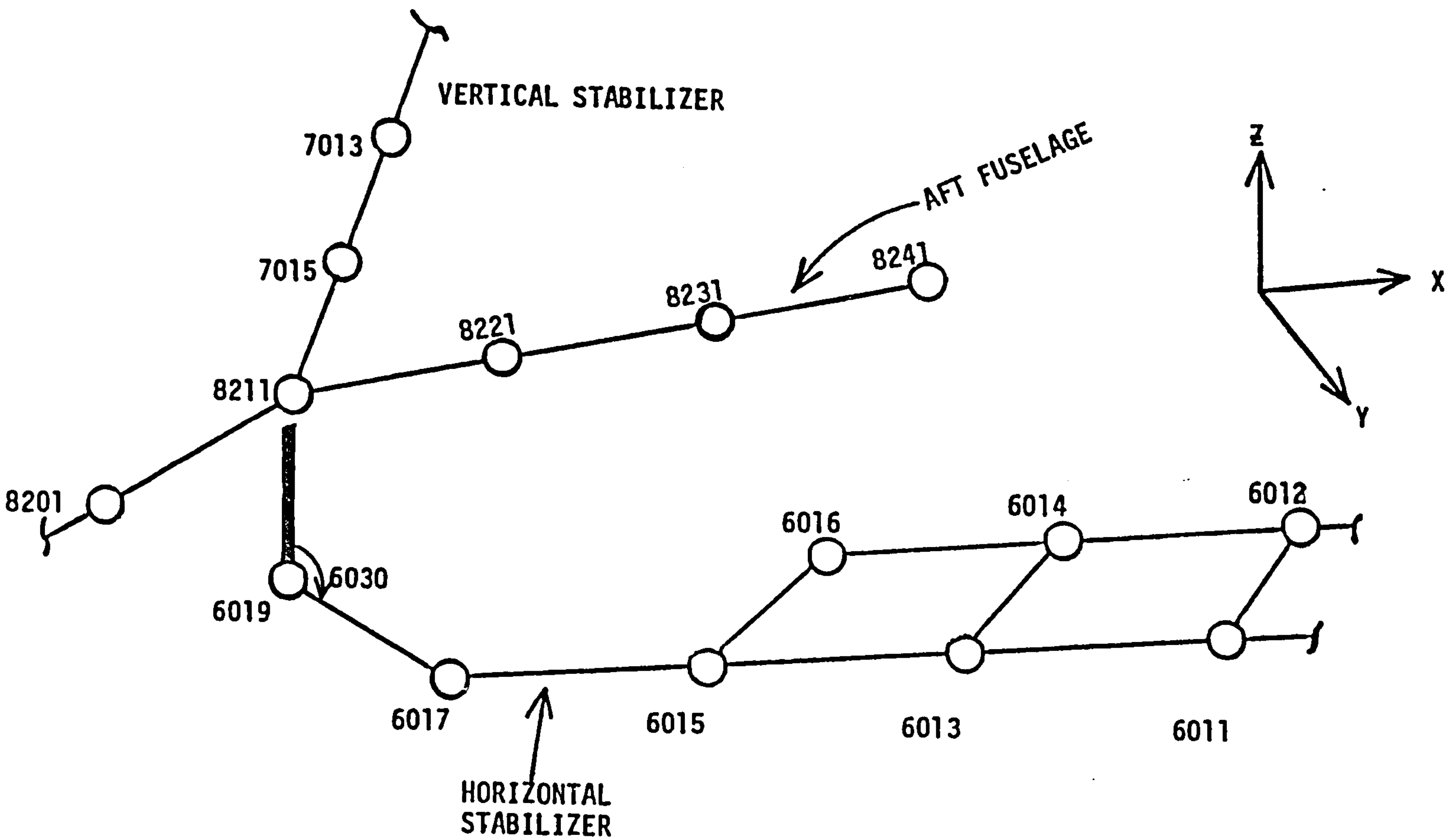


FIGURE 3.5-28 A schematic diagram of the actuator located between horizontal stabilizer and the pivot point

```
$ HORIZ STAB. TO PIVOT ACTUATOR MPCS.
SPOINT      6030
CELAS4     86030  6.06+8    6030
MPC          111    6019       5     0.0    6030         0      1.0
MPCA         111    6019       5     BAR   96019         5        5     -1.0
```

FIGURE 3.5-29 Input data required for the Horizontal Stabilizer Actuator

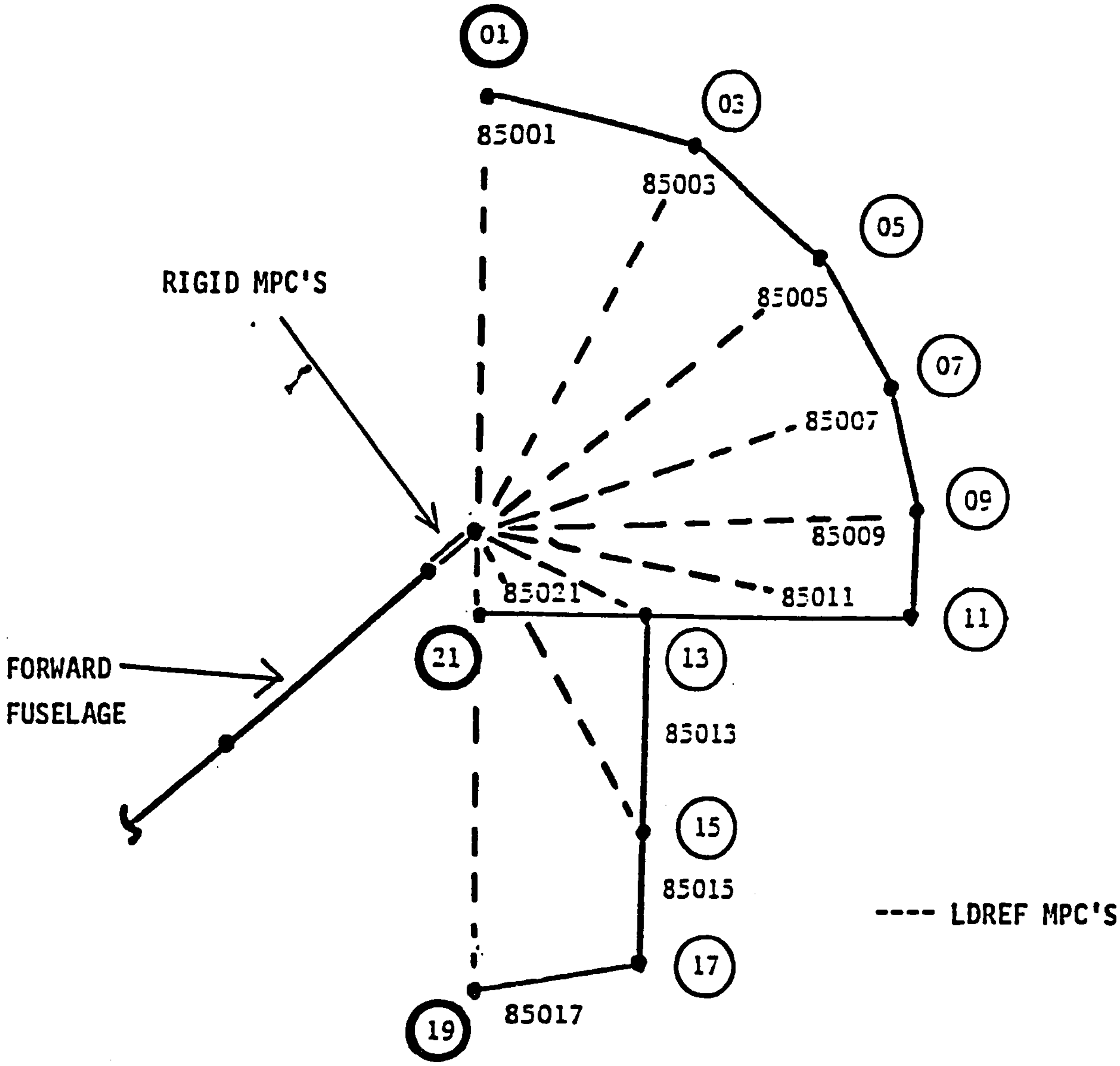


FIGURE 3.5-30 A Schematic Diagram of the Connection Between Forward Fuselage and the Center Wing Using a RIGID BAR MPC and 'LDREF' MPCS

```
$ MPC FOR FORWARD FUSELAGE BEAM MODEL TO CENTERBODY 3-D MODEL.
LDREF   50001   FORCE   1.0     1.0     .0      .0      0
LDREF   50001   LOC     5000                            PERT    900
LDREF   50002   FORCE   1.0     .0      1.0     .0      0
LDREF   50002   LOC     5000                            PERT    901
LDREF   50003   FORCE   1.0     .0      .0      1.0     0
LDREF   50003   LOC     5000                            PERT    900
LDREF   50004   MOMENT  1.0     1.0     .0      .0      0
LDREF   50004   LOC     5000                            PERT    901
LDREF   50005   MOMENT  1.0     .0      1.0     .0      0
LDREF   50005   LOC     5000                            PERT    900
LDREF   50006   MOMENT  1.0     .0      .0      1.0     0
LDREF   50006   LOC     5000                            PERT    901
LGROUP  900     FVECQ   5003    5001                            1.0
LGROUP  900     FVECQ   5005    5003                            1.0
LGROUP  900     FVECQ   5007    5005                            1.0
LGROUP  900     FVECQ   5009    5007                            1.0
LGROUP  900     FVECQ   5011    5009                            1.0
LGROUP  900     FVECQ   5013    5011                            1.0
LGROUP  900     FVECQ   5015    5013                            1.0
LGROUP  900     FVECQ   5017    5015                            1.0
LGROUP  900     FVECQ   5019    5017                            1.0
LGROUP  900     FGRIDQ  5003    5001    1               0       1.0
LGROUP  900     FGRIDQ  5005    5003    1               0       1.0
LGROUP  900     FGRIDQ  5007    5005    1               0       1.0
LGROUP  900     FGRIDQ  5009    5007    1               0       1.0
LGROUP  900     FGRIDQ  5011    5009    1               0       1.0
LGROUP  900     FGRIDQ  5013    5011    1               0       1.0
LGROUP  900     FGRIDQ  5015    5013    1               0       1.0
LGROUP  900     FGRIDQ  5017    5015    1               0       1.0
LGROUP  900     FGRIDQ  5019    5017    1               0       1.0
LGROUP  900     FGRID   5000    .0      1.0E+8  .0
LGROUP  900     MGRID   5000    1.0E+8  .0      1.0E+8
LGROUP  901     FVECQ   5003    5001                            1.0
LGROUP  901     FVECQ   5005    5003                            1.0
LGROUP  901     FVECQ   5007    5005                            1.0
LGROUP  901     FVECQ   5009    5007                            1.0
LGROUP  901     FVECQ   5011    5009                            1.0
LGROUP  901     FVECQ   5013    5011                            1.0
LGROUP  901     FVECQ   5015    5013                            1.0
LGROUP  901     FVECQ   5017    5015                            1.0
LGROUP  901     FVECQ   5019    5017                            1.0
LGROUP  901     FGRIDQ  5003    5001    1               0       1.0
LGROUP  901     FGRIDQ  5005    5003    1               0       1.0
LGROUP  901     FGRIDQ  5007    5005    1               0       1.0
LGROUP  901     FGRIDQ  5009    5007    1               0       1.0
LGROUP  901     FGRIDQ  5011    5009    1               0       1.0
LGROUP  901     FGRIDQ  5013    5011    1               0       1.0
LGROUP  901     FGRIDQ  5015    5013    1               0       1.0
LGROUP  901     FGRIDQ  5017    5015    1               0       1.0
LGROUP  901     FGRIDQ  5019    5017    1               0       1.0
LGROUP  901     FGRID   5000    1.0E+8  .0      1.0E+8
LGROUP  901     MGRID   5000    .0      1.0E+8  .0
MPC     100     5003    1       LOAD    MPLTB   50001           1.0
MPC     105     5003    2       LOAD    MPLTB   50002           1.0
MPC     100     5011    3       LOAD    MPLTB   50003           1.0
MPC     105     5017    2       LOAD    MPLTB   50004           1.0
MPC     100     5017    1       LOAD    MPLTB   50005           1.0
MPC     105     5011    1       LOAD    MPLTB   50006           1.0
MPC     100     5000    1       1.0     8101    1       -1.0
MPC     105     5000    2       1.0     8101    2       -1.0
MPC     100     5000    3       1.0     8101    3       -1.0
MPC     105     5000    4       1.0     8101    4       -1.0
MPC     100     5000    5       1.0     8101    5       -1.0
MPC     105     5000    6       1.0     8101    6       -1.0
```

FIGURE 3.5-31 Input Data Required for the Connection Between Forward Fuselage and the Center Wing

```
$ MPC FOR AFT FUSELAGE BEAM MODEL TO CENTERBODY 3-D MODEL.
LDREF   57001   FORCE   1.0     1.0     .0      .0      0
LDREF   57001   LOC     5700                            PERT    970
LDREF   57002   FORCE   1.0     .0      1.0     .0      0
LDREF   57002   LOC     5700                            PERT    971
LDREF   57003   FORCE   1.0     .0      .0      1.0     0
LDREF   57003   LOC     5700                            PERT    970
LDREF   57004   MOMENT  1.0     1.0     .0      .0      0
LDREF   57004   LOC     5700                            PERT    971
LDREF   57005   MOMENT  1.0     .0      1.0     .0      0
LDREF   57005   LOC     5700                            PERT    970
LDREF   57006   MOMENT  1.0     .0      .0      1.0     0
LDREF   57006   LOC     5700                            PERT    971
LGROUP  970     FVECQ   5703    5701                            1.0
LGROUP  970     FVECQ   5705    5703                            1.0
LGROUP  970     FVECQ   5707    5705                            1.0
LGROUP  970     FVECQ   5709    5707                            1.0
LGROUP  970     FVECQ   5711    5709                            1.0
LGROUP  970     FVECQ   5713    5711                            1.0
LGROUP  970     FVECQ   5715    5713                            1.0
LGROUP  970     FVECQ   5717    5715                            1.0
LGROUP  970     FVECQ   5719    5717                            1.0
LGROUP  970     FGRIDQ  5703    5701    1               0       1.0
LGROUP  970     FGRIDQ  5705    5703    1               0       1.0
LGROUP  970     FGRIDQ  5707    5705    1               0       1.0
LGROUP  970     FGRIDQ  5709    5707    1               0       1.0
LGROUP  970     FGRIDQ  5711    5709    1               0       1.0
LGROUP  970     FGRIDQ  5713    5711    1               0       1.0
LGROUP  970     FGRIDQ  5715    5713    1               0       1.0
LGROUP  970     FGRIDQ  5717    5715    1               0       1.0
LGROUP  970     FGRIDQ  5719    5717    1               0       1.0
LGROUP  970     FGRID   5700    .0      1.0E+8  .0
LGROUP  970     MGRID   5700    1.0E+8  .0      1.0E+8
LGROUP  971     FVECQ   5703    5701                            1.0
LGROUP  971     FVECQ   5705    5703                            1.0
LGROUP  971     FVECQ   5707    5705                            1.0
LGROUP  971     FVECQ   5709    5707                            1.0
LGROUP  971     FVECQ   5711    5709                            1.0
LGROUP  971     FVECQ   5713    5711                            1.0
LGROUP  971     FVECQ   5715    5713                            1.0
LGROUP  971     FVECQ   5717    5715                            1.0
LGROUP  971     FVECQ   5719    5717                            1.0
LGROUP  971     FGRIDQ  5703    5701    1               0       1.0
LGROUP  971     FGRIDQ  5705    5703    1               0       1.0
LGROUP  971     FGRIDQ  5707    5705    1               0       1.0
LGROUP  971     FGRIDQ  5709    5707    1               0       1.0
LGROUP  971     FGRIDQ  5711    5709    1               0       1.0
LGROUP  971     FGRIDQ  5713    5711    1               0       1.0
LGROUP  971     FGRIDQ  5715    5713    1               0       1.0
LGROUP  971     FGRIDQ  5717    5715    1               0       1.0
LGROUP  971     FGRIDQ  5719    5717    1               0       1.0
LGROUP  971     FGRID   5700    1.0E+8  .0      1.0E+8
LGROUP  971     MGRID   5700    .0      1.0E+8  .0
MPC     100     5703    1       LOAD    MPLTB   57001           1.0
MPC     105     5703    2       LOAD    MPLTB   57002           1.0
MPC     100     5711    3       LOAD    MPLTB   57003           1.0
MPC     105     5717    2       LOAD    MPLTB   57004           1.0
MPC     100     5717    1       LOAD    MPLTB   57005           1.0
MPC     105     5711    1       LOAD    MPLTB   57006           1.0
MPC     100     5700    1       1.0     8141    1       -1.0
MPC     105     5700    2       1.0     8141    2       -1.0
MPC     100     5700    3       1.0     8141    3       -1.0
MPC     105     5700    4       1.0     8141    4       -1.0
MPC     100     5700    5       1.0     8141    5       -1.0
MPC     105     5700    6       1.0     8141    6       -1.0
```

FIGURE 3.5-32 Input Required for the Connection Between Aft Fuselage and the Center Wing

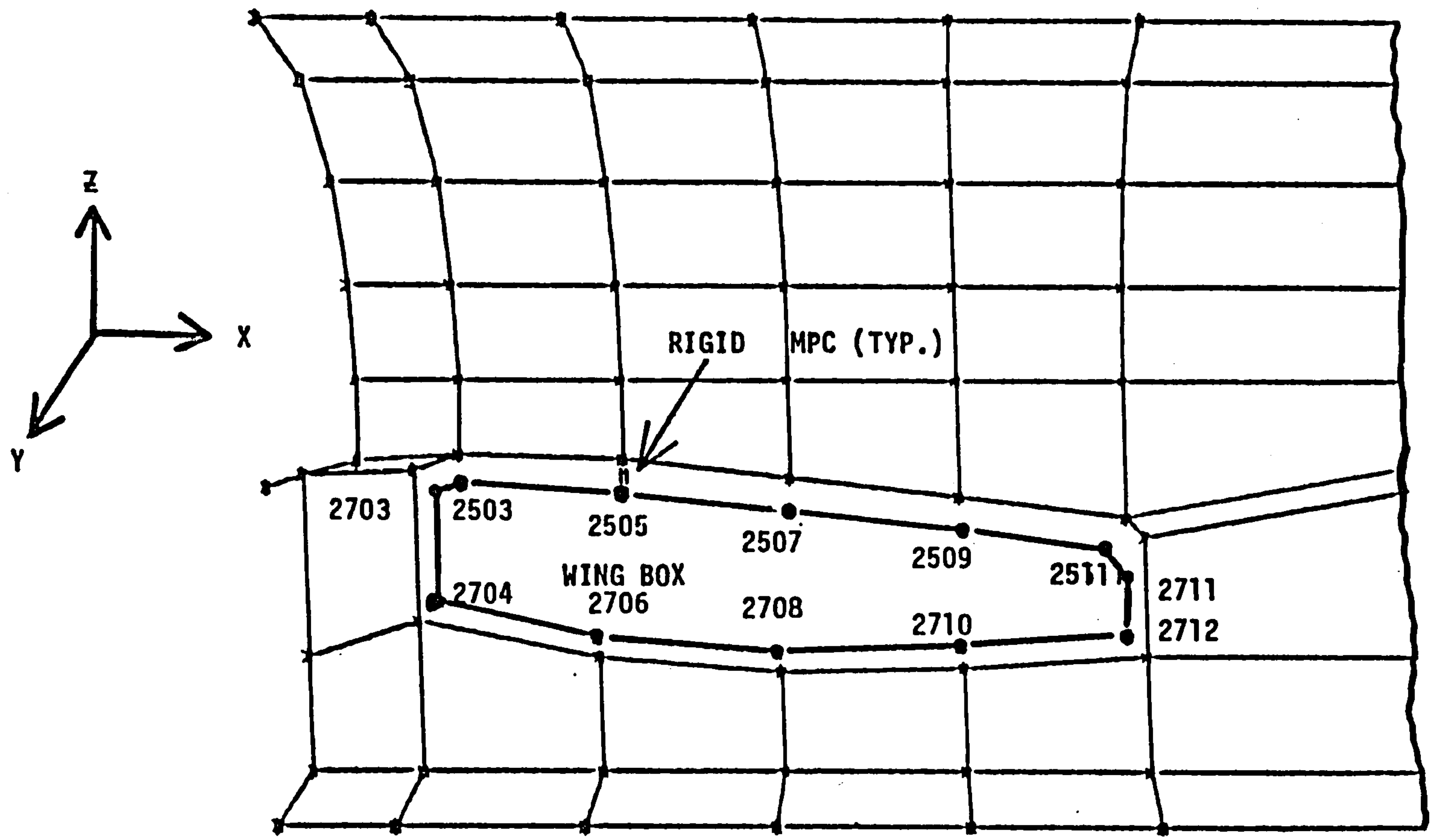


FIGURE 3.5-33 A Schematic Diagram of the Connection Between Center Wing Box and the Center fuselage Modeled Using 'RIGID' MPC's

$ WING - CENTERBODY INTERFACE

MPC	112	5111	1	1.0	2503	1	-1.0	
MPC	112	5111	2	1.0	2503	2	-1.0	
MPC	112	5111	3	1.0	2503	3	-1.0	
MPC	112	5113	1	1.0	2703	1	-1.0	
MPC	112	5113	2	1.0	2703	2	-1.0	
MPC	112	5113	3	1.0	2703	3	-1.0	
MPC	112	5115	1	1.0	2704	1	-1.0	
MPC	112	5115	2	1.0	2704	2	-1.0	
MPC	112	5115	3	1.0	2704	3	-1.0	
MPC	112	5211	1	1.0	2505	1	-1.0	
MPC	112	5211	2	1.0	2505	2	-1.0	
MPC	112	5211	3	1.0	2505	3	-1.0	
MPC	112	5215	1	1.0	2706	1	-1.0	
MPC	112	5215	2	1.0	2706	2	-1.0	
MPC	112	5215	3	1.0	2706	3	-1.0	
MPC	112	5311	1	1.0	2507	1	-1.0	
MPC	112	5311	2	1.0	2507	2	-1.0	
MPC	112	5311	3	1.0	2507	3	-1.0	
MPC	112	5315	1	1.0	2708	1	-1.0	
MPC	112	5315	2	1.0	2708	2	-1.0	
MPC	112	5315	3	1.0	2708	3	-1.0	
MPC	112	5411	1	1.0	2509	1	-1.0	
MPC	112	5411	2	1.0	2509	2	-1.0	
MPC	112	5411	3	1.0	2509	3	-1.0	
MPC	112	5415	1	1.0	2710	1	-1.0	
MPC	112	5415	2	1.0	2710	2	-1.0	
MPC	112	5415	3	1.0	2710	3	-1.0	
MPC	112	5511	1	1.0	2511	1	-1.0	
MPC	112	5511	2	1.0	2511	2	-1.0	
MPC	112	5511	3	1.0	2511	3	-1.0	
MPC	112	5513	1	1.0	2711	1	-1.0	
MPC	112	5513	2	1.0	2711	2	-1.0	
MPC	112	5513	3	1.0	2711	3	-1.0	
MPC	112	5515	1	1.0	2712	1	-1.0	
MPC	112	5515	2	1.0	2712	2	-1.0	
MPC	112	5515	3	1.0	2712	3	-1.0	
MPC	112	5611	3	ROD	95611	1		1.000
CROD	95611	999	5611	2513				

FIGURE 3.5-34 Input Data Required for the Connection Between Center Wing Box and the Center Fuselage

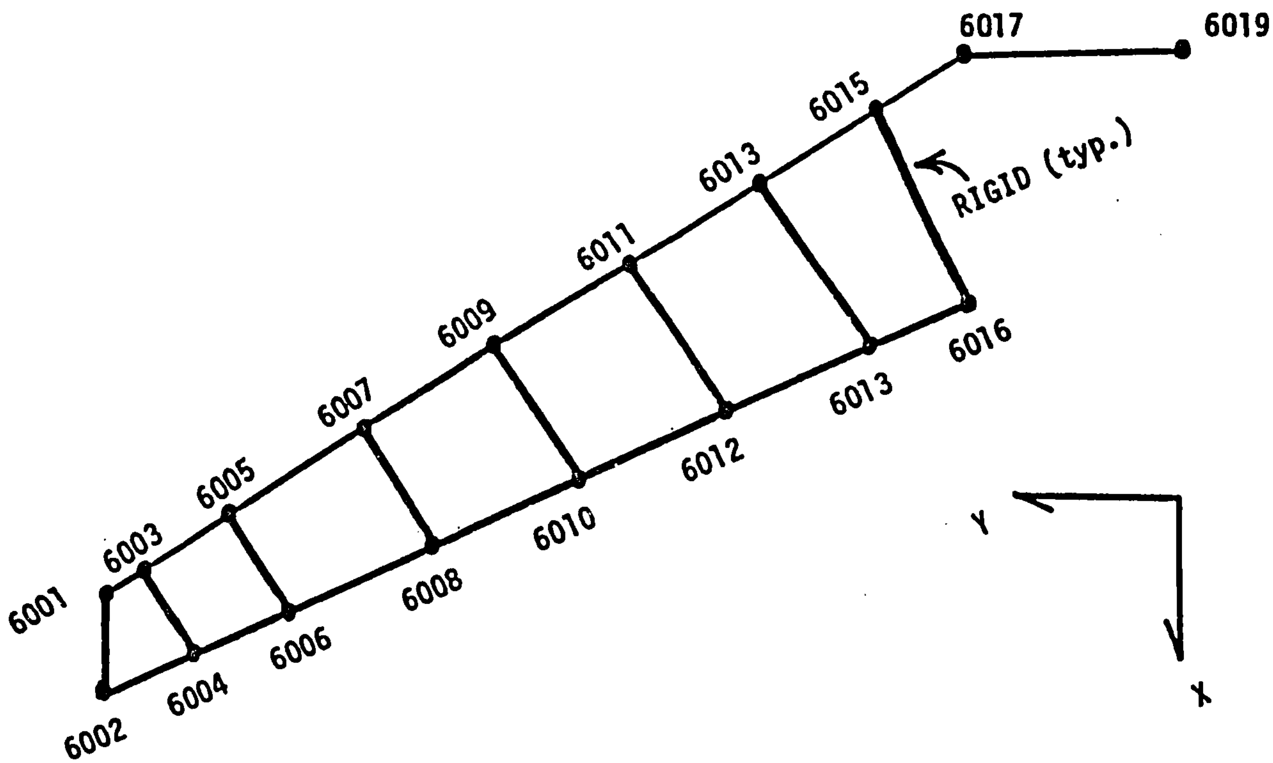


FIGURE 3.5-35 A Schematic Diagram of the Connection Between Horizontal Stabilizer and the Control Surface Modeled Using 'RIGID' BAR Elements

$ HORIZONTAL STABILIZER

MPC	112	6001	5	BAR	96001	5	0	1.000
MPC	112	6003	5	BAR	96003	5	0	1.000
MPC	112	6005	5	BAR	96005	5	0	1.000
MPC	112	6007	5	BAR	96007	5	0	1.000
MPC	112	6009	5	BAR	96009	5	0	1.000
MPC	112	6011	5	BAR	96011	5	0	1.000
MPC	112	6013	5	BAR	96013	5	0	1.000
MPC	112	6015	5	BAR	96015	5	0	1.000
MPC	112	6016	5	BAR	96015	5	5	1.000
MPC	112	6019	3	BAR	96019	1	1	-1.000
$MPC	112	6019	5	BAR	96019	5	5	
MPC	112	6019	4	BAR	96019	6	6	-1.000

FIGURE 3.5-36 Input Data Required for the Connection Between Horizontal Stabilizer and the Control Surface (Elevator)

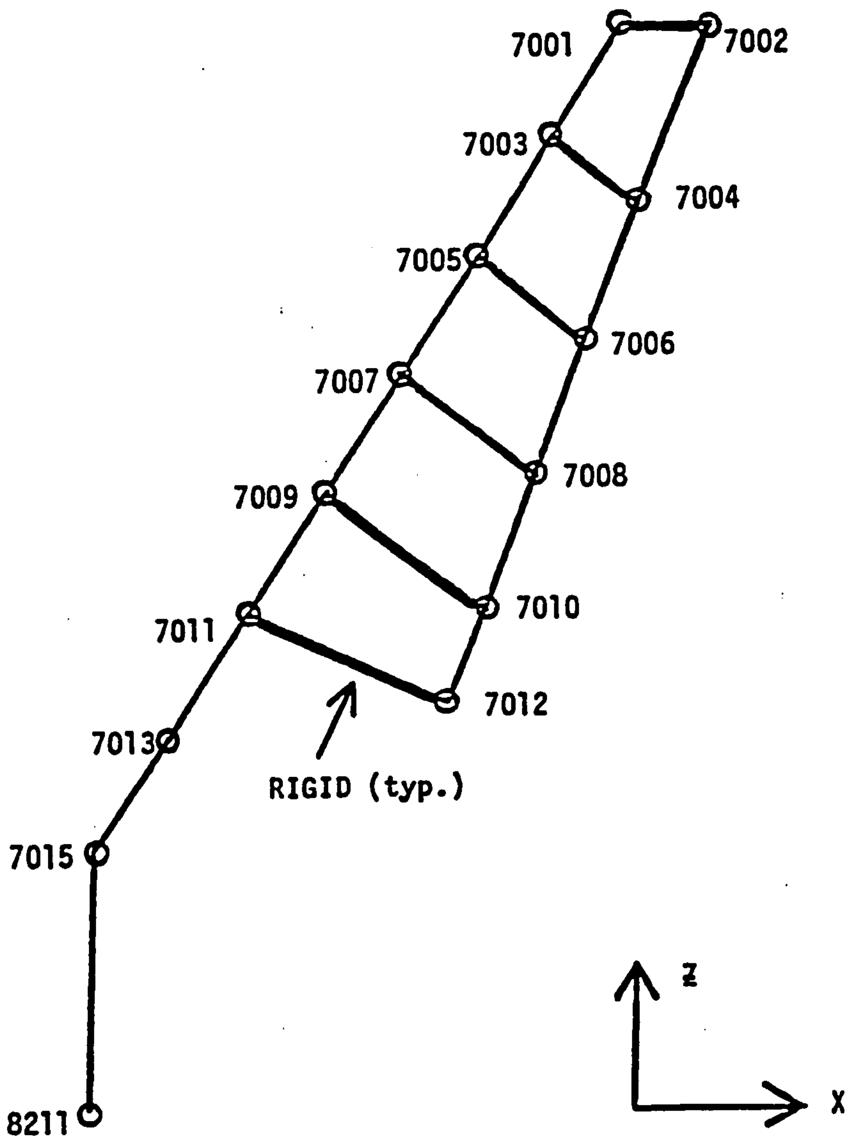


FIGURE 3.5-37 A Schematic Diagram of the Connection Between Vertical Stabilizer and the Control Surface (Rudder)

$ VERTICAL STABILIZER

MPC	112	7001	6	BAR	97001	5	0	1.000
MPC	112	7003	6	BAR	97003	5	0	1.000
MPC	112	7005	6	BAR	97005	5	0	1.000
MPC	112	7007	6	BAR	97007	5	0	1.000
MPC	112	7009	6	BAR	97009	5	0	1.000
MPC	112	7011	6	BAR	97011	5	0	1.000
MPC	112	7012	6	BAR	97011	5	5	1.000
MPC	112	7015	2	BAR	97015	0	5	-1.000
MPC	112	7015	4	BAR	97015	5	5	1.000
MPC	112	7015	6	BAR	97015	4	4	-1.000

FIGURE 3.5-38 Input Data Required for the Connection Between Vertical Stabilizer and the Control Surface (Rudder)

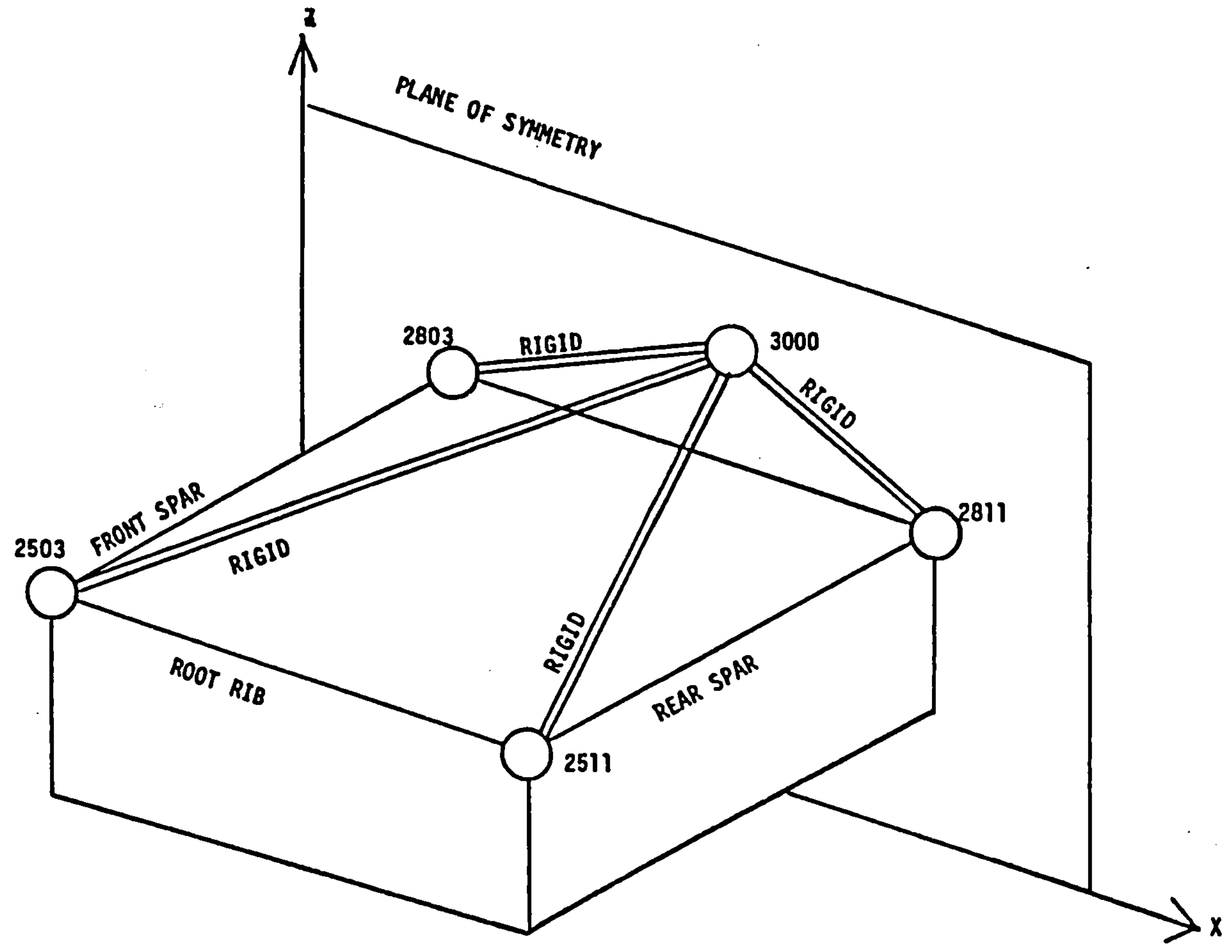


FIGURE 3.5-39 A Schematic Diagram of the Main Support Structure Modeled Using 'RIGID' BAR Elements

$ SUPPORT STRUCTURE MECHANISM MPCS.

CBAR	92803	999	3000	2803	-10.0	0.0	0.0	1	
CBAR	92503	999	3000	2503	0.0	0.0	-10.0	1	
CBAR	92811	999	3000	2811	0.0	0.0	-10.0	1	
CBAR	92511	999	3000	2511	0.0	0.0	-10.0	1	
MPC	120	2503	3	BAR	92503	6	0		-1.000
MPCA	120	2503	3	BAR	92511	6	0		-1.000
MPC	119	2511	3	BAR	92503	6	0		-1.000
MPCA	119	2511	3	BAR	92511	6	0		1.000
MPC	120	2811	2	BAR	92803	5	0		1.000
MPCA	120	2811	2	BAR	92811	5	0		-1.000
MPC	119	2803	1	BAR	92803	1	1		-1.000
MPCA	119	2803	1	BAR	92811	1	1		1.000
MPC	119	2811	1	BAR	92803	6	0		-1.000
MPCA	119	2811	1	BAR	92811	6	0		1.000
MPC	120	2803	2	BAR	92803	5	0		1.000
MPCA	120	2803	2	BAR	92811	5	0		1.000

FIGURE 3.5-40 Input Data Required for the Main Support Structure

3.5.3 SIC definition and SIC matrix calculation - A structural influence coefficient (SIC) is defined as the deflection at one point due to a unit load at the same point or at another point. The generalized mathematical definition of a SIC is given in reference 3. The concept of the SIC does not imply that the applied load acts at only one element node, only that it is applied at a single point. In fact, a unit applied SIC load may be distributed over several grid points within a region of the airframe. The deflections calculated by the use of such unit loads can be physically interpreted as the weighted averages of the deflections of the grid points loaded by the unit SIC load.

The SIC degrees of freedom and location definitions for both the symmetric and antisymmetric Baseline airplane are given in the MASTER table shown in Appendix D. An example of a MASTER table, showing symmetric and antisymmetric SIC points, is given in figure 3.5-41.

In general, the SIC unit loads and the information contained in the MASTER table are self explanatory; however, fuselage unit SIC loads have different significance for symmetric and antisymmetric SIC's. Since the stick (beam) fuselage is on the plane of symmetry, the symmetric and antisymmetric boundary conditions effect the fuselage nodes. Therefore, unit loads for generation of a SIC matrix may be applied to degrees of freedom which are constrained by the boundary conditions.

A brief description of the various steps required to calculate the SIC matrix is given below.

1. A unit force or moment is applied at each SIC degree of freedom. [P] (SIC by SIC)

2. The unit load is distributed to the structural nodes using an external load transformation matrix (See section 3.5.4)

   [TRNSIC] (G by SIC).

3. A set of displacement vectors is then calculated for SIC load conditions. The [K] is a constrained stiffness matrix.

$$[\ U\ ]\ (G \text{ by } SIC) = [\ K\ ]^{-1} * [\ TRNSIC\ ] * [\ P\ ]$$

$$$	SYMMETRIC SIC TABLE								
SICSYA	TABLE	PADS	VERSN-B						
SICSYC	SIC-COL	KAA-COL	PART#	DOF	GRIDID	X	Y	Z	CP
SICSYM	1	1	10.0120	3	153	567.109	1099.072.375		12
SICSYM	2	2	10.0120	3	155	572.902	1099.072.317		12
SICSYM	3	3	10.0120	3	157	588.544	1099.072.153		12
SICSYM	103	103	10.0810	5	2190	1315.307207.589		17.1	
SICSYM	104	104	10.0810	6	2190	1315.307207.589		17.1	
SICSYM	105	105	50.0100	1	3000	203.849	.0	71.86	11
SICSYM	106		50.0100	2	3000	203.849	.0	71.86	11
SICSYM	107	106	50.0100	3	3000	203.849	.0	71.86	11
SICSYM	108		50.0100	4	3000	203.849	.0	71.86	11
SICSYM	109	107	50.0100	5	3000	203.849	.0	71.86	11

$$$	ANTI-SYMMETRIC SIC TABLE								
SICASC	SIC-COL	KAA-COL	PART#	DOF	GRIDID	X	Y	Z	CP
SICASY	1	1	10.0120	3	153	567.109	1099.072.375		12
SICASY	2	2	10.0120	3	155	572.902	1099.072.317		12
SICASY	3	3	10.0120	3	157	588.544	1099.072.153		12
SICASY	103	103	10.0810	5	2190	1315.307207.589		17.1	
SICASY	104	104	10.0810	6	2190	1315.307207.589		17.1	
SICASY	105		50.0100	1	3000	203.849	.0	71.86	11
SICASY	106	105	50.0100	2	3000	203.849	.0	71.86	11
SICASY	107		50.0100	3	3000	203.849	.0	71.86	11
SICASY	108	106	50.0100	4	3000	203.849	.0	71.86	11
SICASY	109		50.0100	5	3000	203.849	.0	71.86	11

FIGURE 3.5-41 An Example of the 'MASTER' Table Showing Symmetric and Antisymmetric SIC Column Locations

4. These displacement vectors are then pre-multiplied by the the transpose of the TRNSIC matrix to obtain the SIC matrix.

$$[\ SIC\ ]\ (\ SIC\ \text{by}\ SIC\ ) = [\ TRNSIC\ ]^{T} * [\ U\ ]$$

3.5.4 External load transformation - The external loads are applied at the SIC points. In the static solution, these loads are then transformed and applied to the structural grid points using a transformation matrix TRNSIC. This section describes the general procedure utilized to calculate this transformation matrix.

The Static Loads analysis involves the SIC degrees of freedom. The external loads formed by the Static Loads analysis are generated at the SIC nodes. A set of SIC points are designated on the airframe structure, which for most of the structure are different from the structural nodes. However, SIC points on segments with beam representation, (forward and aft fuselage etc.), have SIC points which are also structural nodes. Therefore the transformation matrix between these SIC points and the structural nodes is an identity matrix. The transformation between the SIC points located on the 3-D model and the structural grid points is calculated using a Calac developed 'load distribution' technique. A PLI coded program called PANLWT is executed to generate a set of LDREF and LGROUP cards which are used in NASTRAN to generate a transformation matrix for each unit load condition. Weighting factors computed by PANLWT based on areas are used by NASTRAN. Figure 3.5-42 shows two examples of LDREF and LGROUP cards for SIC points 153 and 155. Figure 3.5-43 shows a general load allocation selection procedure used by PANLWT. The structural analyst may optionally modify the output from PANLWT to minimize the cross-coupling between SIC points and the structural nodes by use of a program called FGRIDCRD. The load distribution module in NASTRAN calculates percent load applied to each grid point based on the weighting factors designated in field 9 of the LGROUP cards. The load distribution obtained from this procedure satisfies equilibrium.

```
$       $$      $$      $$      $$      $$      $$      $$      $$
LDREF   101     LOC             740.759 954.452 -7.475  12      153
LDREF   101     FORCE   1.000   0.000   0.000   1.000
$
LGROUP  153     FGRIDP  199     99      101     201             0.33
LGROUP  153     FGRIDP  299     199     201     301             0.35
LGROUP  153     FGRIDP  399     299     301     401             0.36
LGROUP  153     FGRIDP  499     399     401     501             0.38
$       $$      $$      $$      $$      $$      $$      $$      $$
$
LDREF   102     LOC             746.571 954.452 -6.676  12      155
LDREF   102     FORCE   1.000   0.000   0.000   1.000
LGROUP  155     FGRIDP  499     399     401     501             0.62
LGROUP  155     FGRIDP  199      99     101     201             0.67
LGROUP  155     FGRIDP  399     299     301     401             0.64
LGROUP  155     FGRIDP  299     199     201     301             0.65
```

FIGURE 3.5-42   An Example of Load Distribution Cards

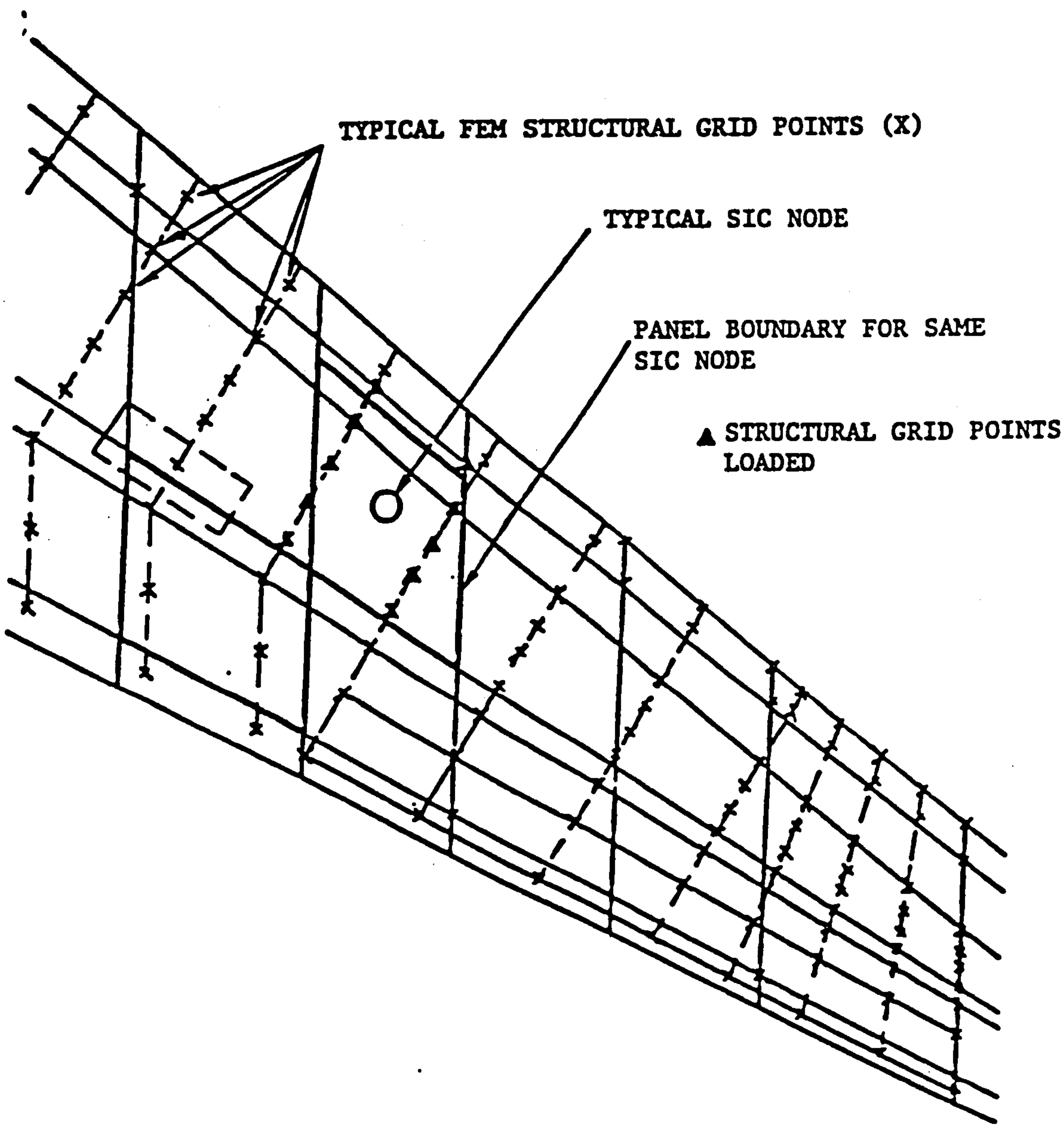


FIGURE 3.5-43 An example of typical FEM structural grid points overlayed on the SIC node points

3.5.5 Static solution - The static solution of the finite element model is performed using Calac developed Rigid Format 145 (RF145). RF145 DMAP sequence is a modification and enhancement of the standard NASTRAN (COSMIC) RIGID FORMAT 1 (RF1).

RF145 contains the following enhancements that RF1 does not have.

* multipoint constraint matrix (RM) check.
* Stiffness matrix (KAA AND KLL) check.
* External load (PG) summation check.
* Reaction load (QG) summation check.
* Balance (between external loads and reactions) check.
* Rigid body check.

After the above mentioned checks have verified the model, the static solution is obtained. The displacement vectors, internal loads and internal stresses are printed and scanned for high stresses by the stress analyst.

3.6 Stress Analysis

The flow diagram shown in figure 3.6-1 depicts a typical PADS preliminary structural design operation. The functions enclosed by the broken line are the stress analysis functions. The basic components required for stress analysis are a static analysis program and a sizing program. These programs must be capable of analyzing the types of cross sections under consideration. The stress analysis program's function is to determine the element internal loads which occur for external loading conditions. The sizing analysis program's function is to determine the optimum detail dimensions of cross sections for specified element internal load conditions. In PADS, NASTRAN performs the static analysis, and PSASA (Panel Sizing And Stress Allowable) performs the sizing analysis.

PSASA is constructed around a computer program called SPOT. SPOT optimizes stringer stiffened and integrally stiffened panels for fuselage and lifting surfaces. PSASA generates stress allowables and sizing for design elements of a finite element model.

NASTRAN (Calac-NASTRAN) also contains a module (FSD algorithm) for element sizing. The module currently assumes that allowable stresses remain constant during the resizing process. In PADS, this module has been used for validation comparison purposes only.

3.6.1 SPOT/OPCOM Comparison - Calac's version of SPOT was developed at Lockheed over a number of years. The analysis methodology applies to most metals, and the skin and stringer materials need not be from the same family. Over 110 failure modes are checked to produce the least weight cross-section. Smooth envelopes are produced which have no abrupt change in slope. The method of analysis has been correlated with the data from over eight hundred test panels and by use in the analysis of several aircraft which have successfully passed static tests.

SPOT was used to produce a family of least weight allowable load envelopes for Z and J stiffened panels of various dimensions. The family of Z stiffened panels consisted of panels capable of sustaining light, moderate, and heavy loads. The family of J stiffened panels consisted of panels capable of sustaining light and heavy loads. Table 3.6-1 describes these panels and figure 3.6-2 defines the cross section configuration.

A comparison of allowable load envelopes produced by the Kentron Technical Center (OPCOM) computer program and SPOT was conducted for the Z and J stiffened panels of table 3.6-1.

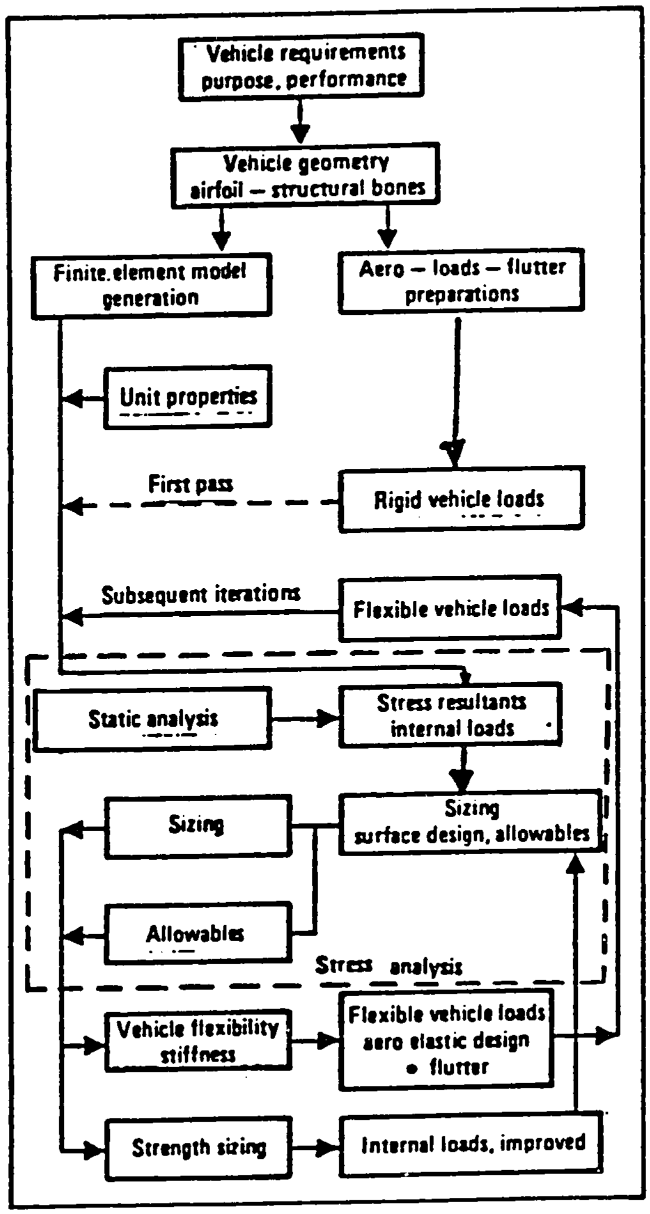


FIGURE 3.6-1 Flow Diagram Depicting PADS Structural Design Operation

TABLE 3.6-1 STRINGER STIFFENED PANELS FOR COMPARISON

LOAD	STRG	LENGTH	TS	BS	TW	H	TAF	WAF	TF	WF
LGHT	Z	26.0	0.08	5.20	0.08	1.50	0.08	1.04	.096	0.64
MOD	Z	26.0	0.08	5.20	0.08	1.70	.082	1.04	.096	0.71
HEAV	Z	26.0	.383	5.20	.182	3.12	0.26	1.14	.219	1.37
LGHT	J	26.0	0.08	7.40	0.08	1.50	0.08	2.00	.096	0.64
HEAV	J	26.0	.555	7.40	.163	3.70	.808	2.08	.195	1.45

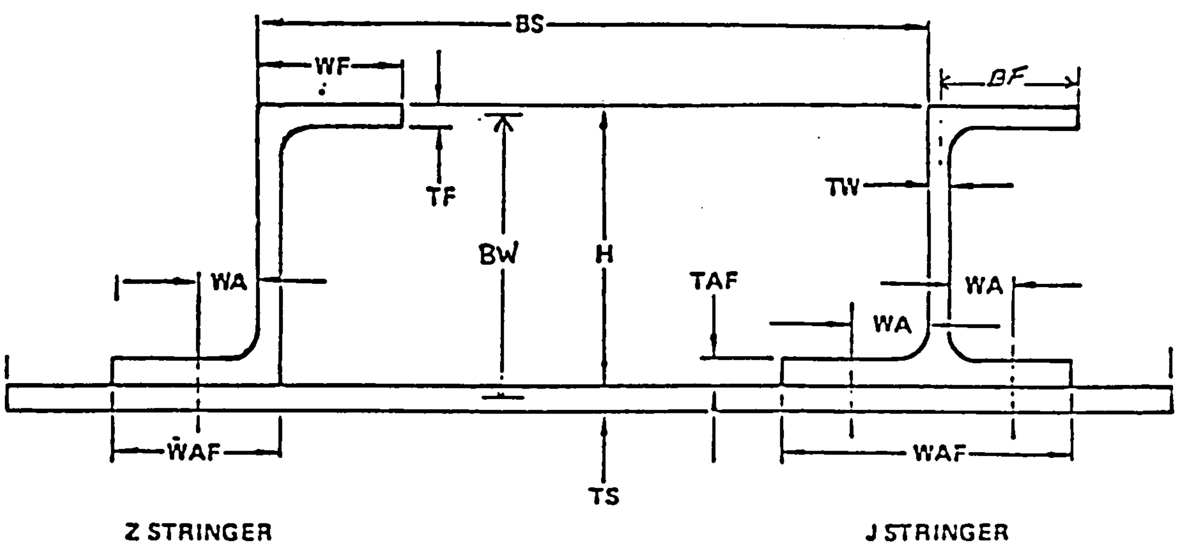


FIGURE 3.6-2 Cross-Section Configuration

The results of this comparison are shown in figures 3.6-3 through 3.6-7. Significant differences are apparent which are considered unacceptable for this type of comparison.

OPCOM
SPOT

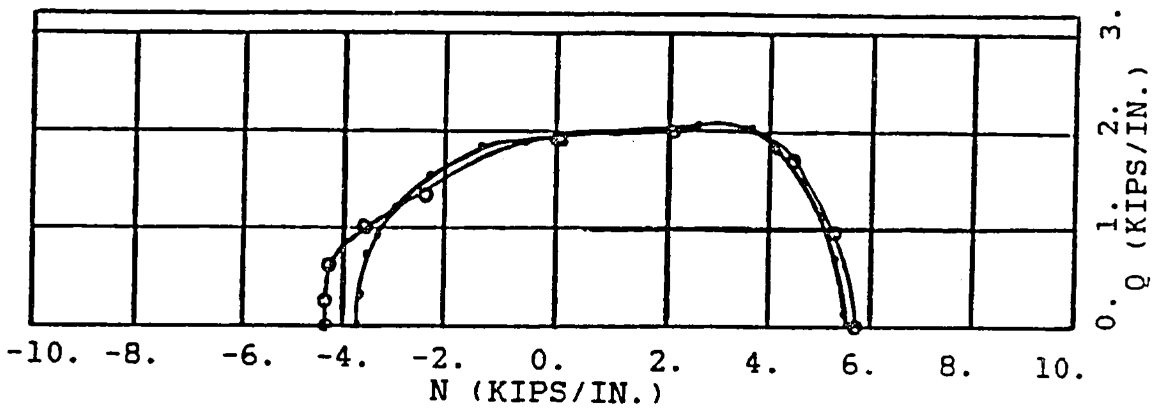


FIGURE 3.6-3 - LIGHTLY LOADED Z-STIFFENED PANEL

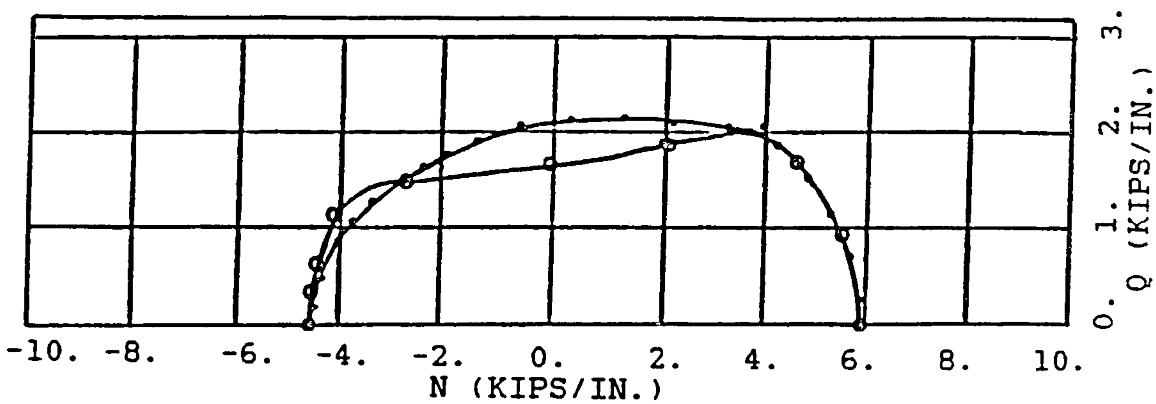


FIGURE 3.6-4 - MODERATELY LOADED Z-STIFFENED PANEL

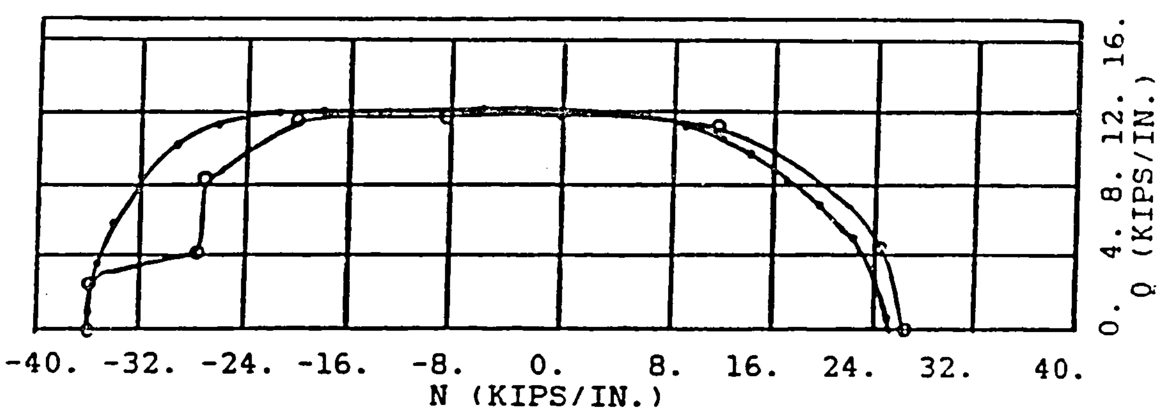


FIGURE 3.6-5 - HEAVILY LOADED Z-STIFFENED PANEL

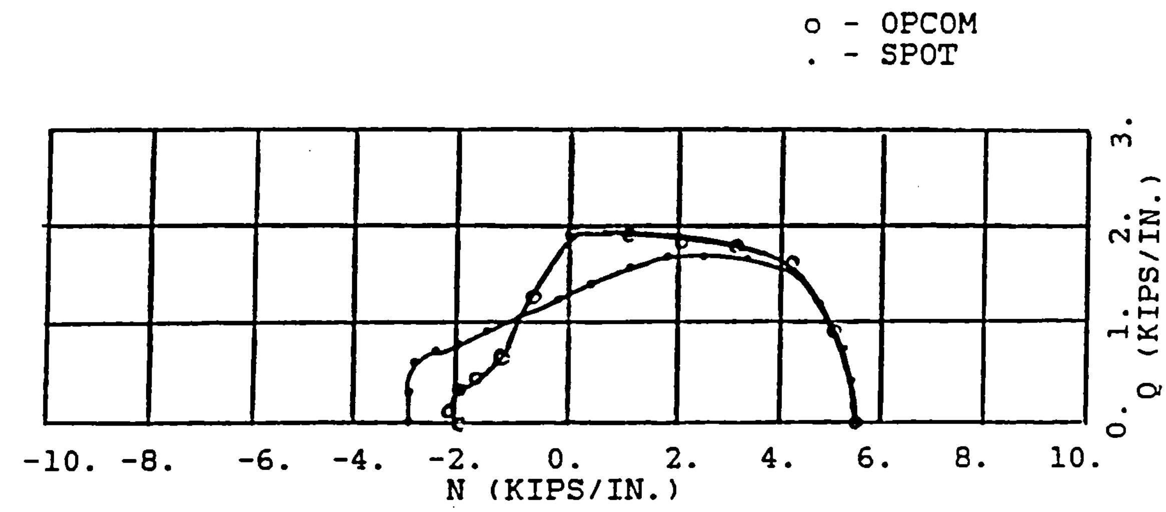


FIGURE 3.6-6 - LIGHTLY LOADED J-STIFFENED PANEL

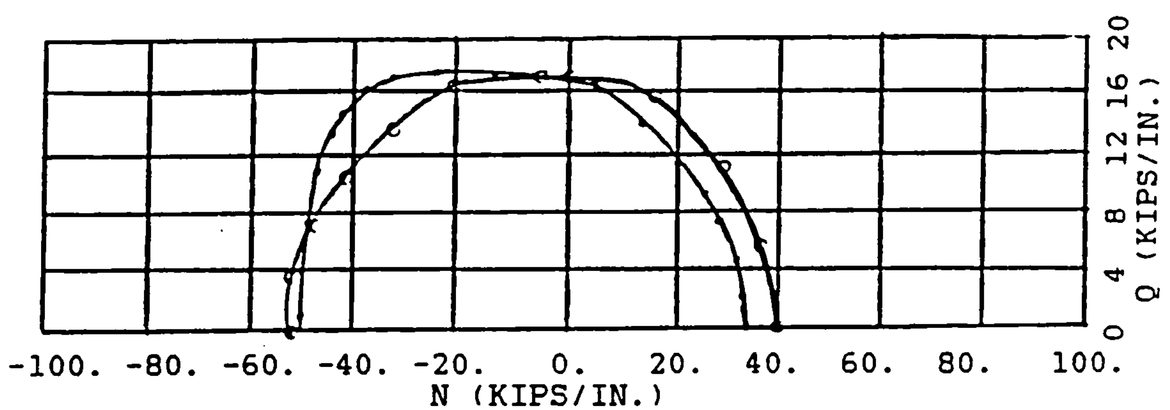


FIGURE 3.6-7 - HEAVILY LOADED J-STIFFENED PANEL

3.7 Automated Sizing For Preliminary Design

PSASA is a complex array of programs which take as input, the elements to be designed and the internal load conditions derived from applied external load conditions, and generates the stress allowables and new sizings for these elements.

The panel optimization algorithm of PSASA (SPOT) is capable of analyzing Z and J cross sections as shown in figure 3.6-2. This algorithm is based on a modified "try-them-all" method and operates on optimization variables within a specified ranges.

Figure 3.7-1 shows the location of the geometry constraints; the rib spacing (L=26 inches) and the stringer spacing (BU=5.2 inches and BL=7.4 inches). The minimum dimensions are shown in table 3.7-1 and two additional constraints in table 3.7-2. The material proportions for the aluminum alloy skin and stringers, which are the same for the upper and lower surfaces, are shown in table 3.7-3. The variables for optimization are the stiffener height (H), the stiffener web thickness (TW), the stiffener flange thickness (TAF), and the panel skin thickness (TS).

The key to the approach of PSASA is the panel database (or panel design library). The panel database is an array of detail dimensions and strength envelopes for a family of preselected cross sections. This array is stored in a direct-access file accessible in order of weight per unit area. SPOT uses the panel database to converge to optimal panel dimensions much more economically than a complete optimization of each panel from minimum cross sections.

The basic methodologies of PSASA were previously developed as separate entities. The major thrust of the PADS work was to integrate the separate entities and obtain a smoothly flowing computational procedure.

The major components of the procedure developed are indicated in the flow diagram of figure 3.7-2. The overall procedure consists of two distinct phases:

- o Database generation (panel design library)
- o Resizing cycle

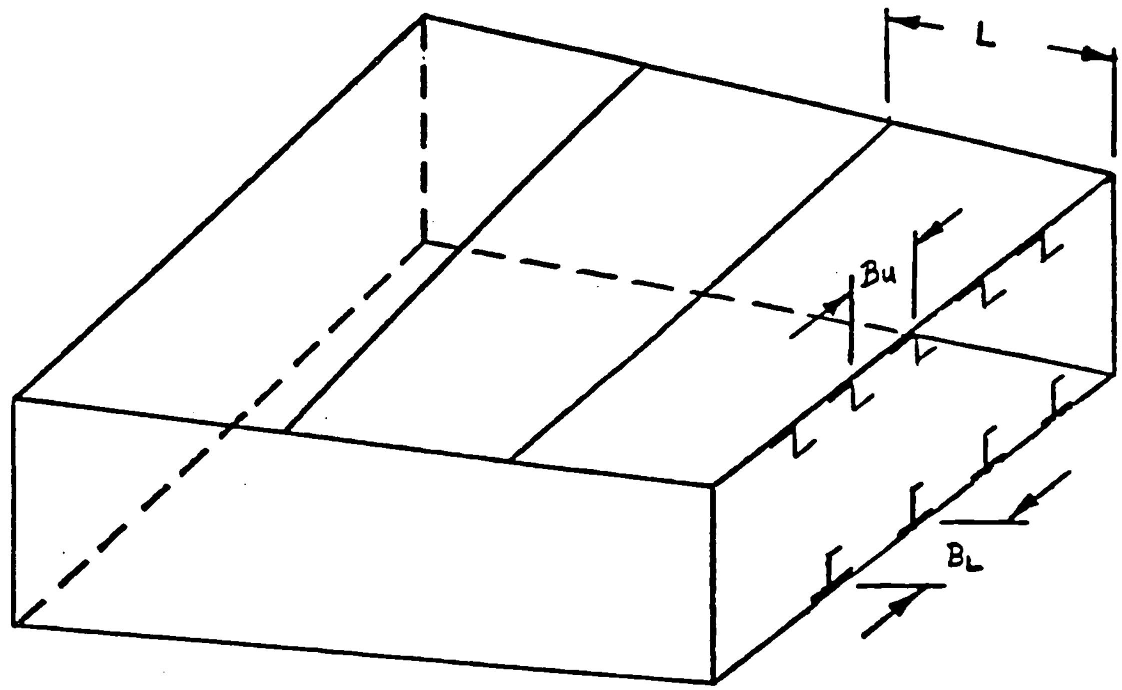


FIGURE 3.7-1 Geometry Constraints for Panel Sizing

PARAMETER	UPPER SURFACE (INCHES)	LOWER SURFACE (INCHES)	DESCRIPTION OF PARAMETER
BS*	5.200	7.400	STIFFENER SPACING
TW	0.080	0.080	WEB THICKNESS
H	1.500	1.500	STIFFENER HEIGHT
TAF	0.080	0.080	FLANGE THICKNESS (NEAR SKIN)
TS	0.080	0.080	SKIN THICKNESS

* Pitch is constant

TABLE 3.7-1 MINIMUM DIMENSIONS

	UPPER	LOWER
ASKIN/(ASKIN + ASTR) MAX	0.800	0.800
ASKIN/(ASKIN + ASTR) MIN	0.500	0.500
BF/BW MAX	0.400	0.400

TABLE 3.7-2 ADDITIONAL CONSTRAINTS

MECHANICAL PROPERTIES	SKIN 7075-T7651 CLAD SHEET	STRINGER 7075-T6511 EXTRUSION
FTU (KSI)	71	85
FTY (KSI)	61	76
FCX (KSI)	61	76
FSU (KSI)	40	45
FEW (KSI)	120	130
FTAT (KSI)	45	
E (PSI)	10.3 x 10E6	10.4 x 10E6
EC (PSI)	10.5 x 10E6	10.7 x 10E6
G (PSI)	3.9 x 10E6	4.0 x 10E6
NU	.33	.33
MASS DENSITY	.101	.101

TABLE 3.7-3 ROOM TEMPERATURE PROPERTIES

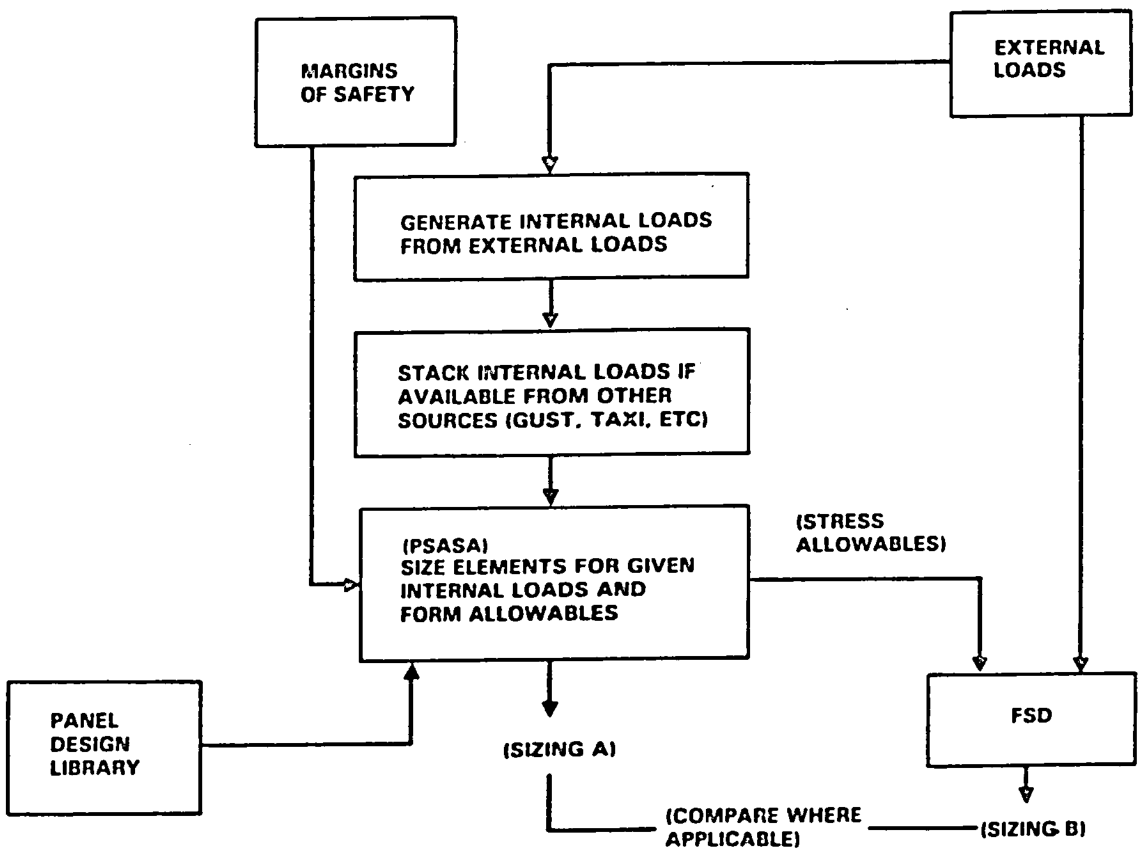


FIGURE 3.7-2 STRUCTURAL SIZING PROCEDURE

3.7.1 Database generation - In line with the initial objectives of the PADS project, the database was constructed for the purpose of resizing wing metal surfaces of the type used in the L-1011 airplane; these surfaces can be analyzed with the SPOT program.

The database, as constructed, consists of two design configurations; the first, for Z-stringer stiffened skin, applies to the upper surface, and the second, for J-stringer stiffened skin, applies to the lower surface.

It is noted that a configuration, as used in this context, entails a number of fixed dimensions, such as column length and stringer spacing. In addition, some more or less empirical constraints (among them minimum dimensions) need to be imposed.

The database for each configuration must cover a range of allowable loads sufficient to handle all internal loads that are expected in the structural elements to be resized. It must further contain data for a sufficient number of cross sections, so that, by means of a suitable interpolation scheme, near-optimum designs can be obtained for any combination of internal load conditions. For any design configuration, the data stored in the database files consist of:

- Panel weight per unit area
- Material data
- Fixed dimensions and constraints
- Detail dimensions for each cross section
- Strength envelope coordinates (shear stress resultant, q, versus axial stress resultant, N)

Each cross section contained in the database file was optimized (by means of the SPOT optimization algorithm as a standalone program) for a single load condition. Each load condition consists of a combination of an axial load per unit of width (N) and a shear flow (q). A total of 202 cross sections were optimized for Z-stringers, and 260 cross sections for J-stringers.

Figure 3.7-3 illustrates the generation of the database file.

A range of load conditions, which start from allowable loads of minimum cross-sections, are formed. These load conditions should cover all internal load combinations that are expected on the structure. The upper portion of figure 3.7-3 shows load conditions used for the optimization of a Z-stringer stiffened panel. The lower portion shows a typical strength envelope for a selected load combination. The database file consists of definition of many panels, each with an associated strength envelope.

3.7.2 Resizing cycle - A resizing cycle is illustrate in figure 3.7-2.

The process is starting with computation of external loads for a given structure. Then element internal loads are formed for the external loads and the given structure. These internal loads are then combined with any other internal loads. PSASA is then executed with or without specified stress margins of safety.

PSASA has capability to reduce the internal load conditions which are passed to SPOT. Critical conditions can be selected for reduction of cost. Currently, all load conditions are passed to SPOT.

For every element of the finite element model, the database is accessed in order of ascending unit weight, and margins of safety are computed for all load conditions until an adequate cross section is found. The final cross section must satisfy the margin of safety specified (or zero) for that particular element. Appendix B describes the specified margins of safety as applied in the PADS sizing process. An interpolation is then performed between the dimensions of this cross section and the most nearly adequate of the lighter cross sections. The resulting interpolated cross section, for which a new strength envelope is computed and appropriately stored, is then used for the resized structure. In this manner all the information required to perform either a new resizing cycle or a NASTRAN FSD run is available.

Margins of safety are computed from the strength envelopes as shown in figure 3.7-4.

Once the first resizing cycle has been completed, further resizing cycles can be repeated as shown in figure 2.3-1 for a typical PADS sizing process through rigid, 1st, and 2nd flex airplane loads.

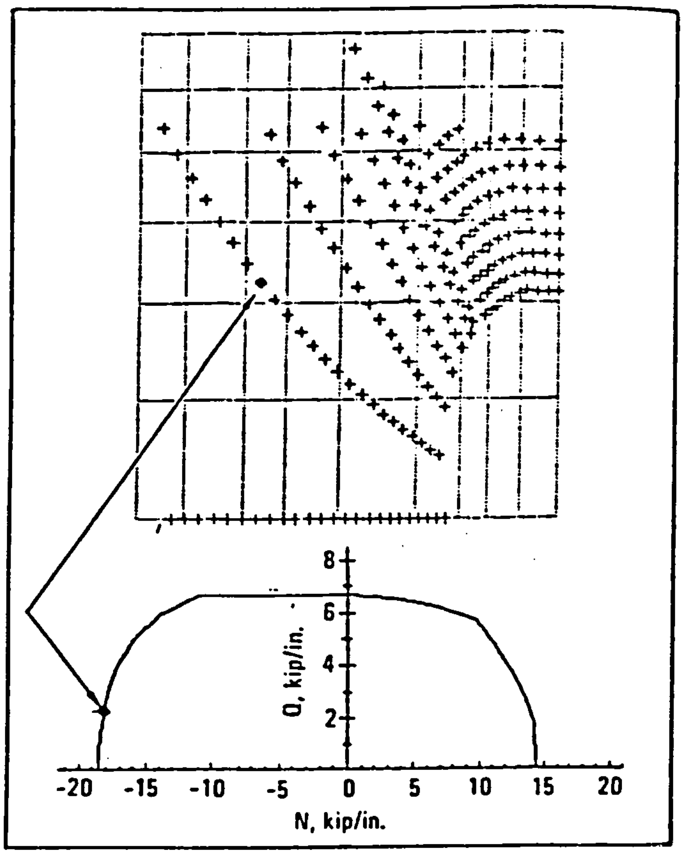


FIGURE 3.7-3
Z-Stringer Stiffened Skin
Applied Loads for Optimization
and Typical Strength Envelope

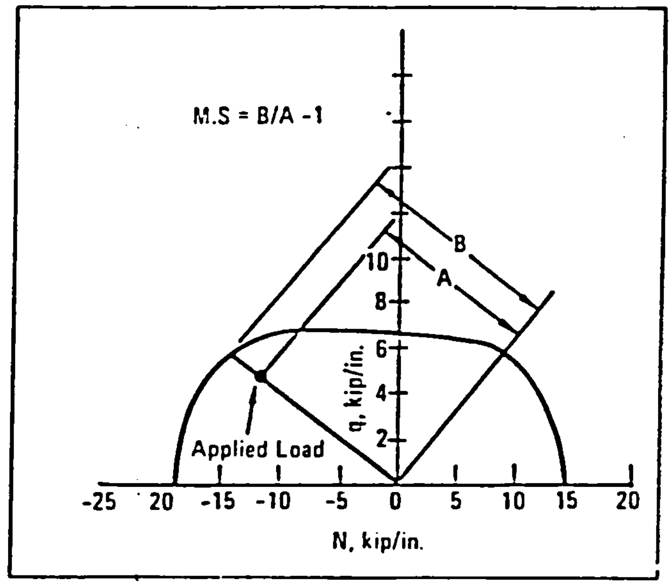


FIGURE 3.7-4
Margin of Safety Calculation

3.7.3 Baseline sizing results - Twenty five static load conditions were formed for the initial wing panel sizing and stiffness generation as presented in table 3.4-2. These 25 conditions include various weight configurations, velocities, and altitudes of maneuvers, as well as ground handling conditions as discussed in section 3.4.2. The initial loads were computed for a rigid airplane and applied to the structural model for computation of the first sizing. First and second flex loads included the effects of flexibility.

Sizing was accomplished with the PADS data base approach for selection of optimal panel dimensions. Stress margins of safety as described in Appendix B were supplied for selected sizing runs. Sizing runs for the Baseline aircraft were run for the standard ACS gain setting of -11.33 degrees per g and for no ACS. The finite element model design regions for the baseline aircraft wing are represented in figure 3.7-5. The design regions include the 1st through 4th row of surface panels for both upper and lower surfaces. These panels were represented by a Lockheed-California Company (Calac) developed quadrilateral finite element (CMEMQ) in the structural model. Internal loads for the wing were computed in a NASTRAN static solution run. Panel sizing and stress allowable (PSASA) module was used for sizing of the surface panels. Initial internal loads were formed from arbitrary starting sizing.

The following table summarizes upper and lower surface panel weights for half an airplane which were computed as direct NASTRAN output for the Baseline design:

NASTRAN Weight Calculation for PADS Baseline Sizing.

External Loads	Properties to be Updated	ACS Gain deg/g	Stress Margins	Upper Cover Weight lb	Low Cover Weight lb
rigid	starting	- 11.33	no	4147.75	5123.20
1st flex	rigid	- 11.33	no	3870.20	4845.49
2nd flex	1st flex	- 11.33	no	3829.56	4847.12
2.5 g no ACS	2nd flex	0.00	no	4103.17	5418.23
2nd flex w margins	2nd flex	- 11.33	yes	4106.54	5611.50

The NASTRAN calculated weights (as described in more detail in section 3.3.4) get factored to account for effects not represented in the finite element. The upper and lower cover

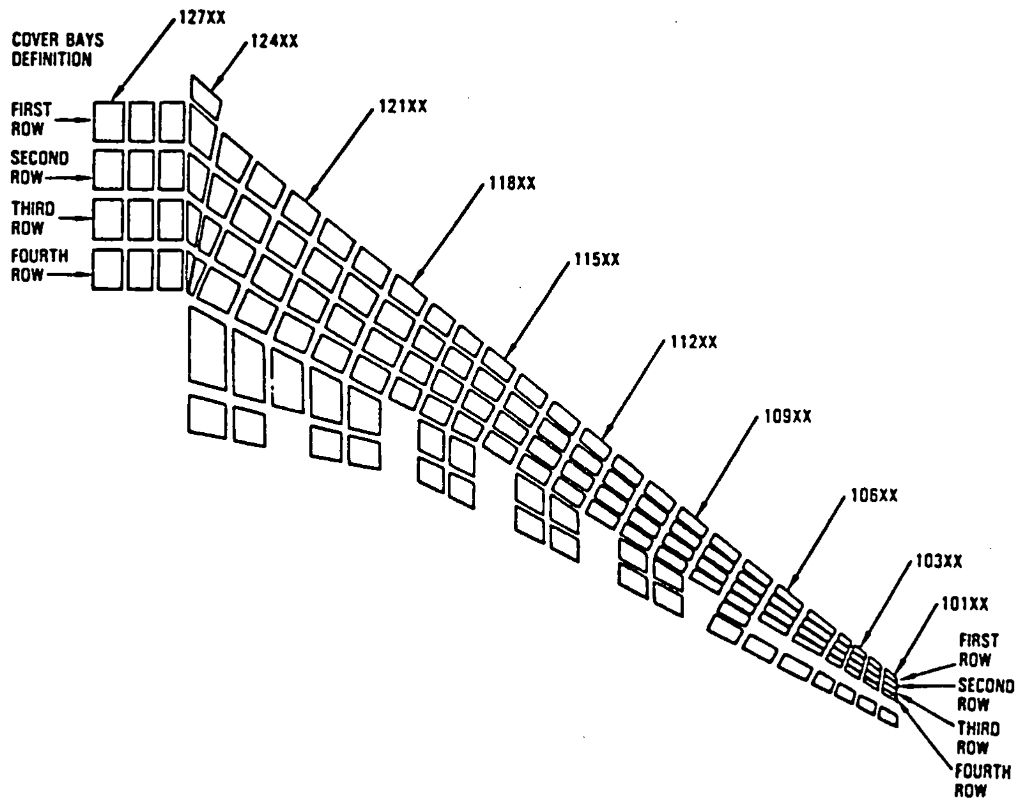


FIGURE 3.7-5 Definition of Cover Bays

weight for the Baseline production sizing (as described in section 3.3.5.2) is 4618.62 and 6342.52 pounds respectively.

Figure 3.7-6 shows the load conditions which have the minimal margin of safety for each panel for upper and lower surfaces, respectively, for the Second flex aircraft loads. The margin of safety is computed as a ratio of internal load which the panel can withstand over the applied internal load for an external load condition minus one. The computed margin must be equal to or greater then the specified margin of safety which is input for the particular panel. In cases where fail safe or fatigue conditions are imposed, the computed minimal margin may be somewhat greater than the specified margin. Braking conditions determined the sizing around the main gear. Wing midsection panels were designed by a 2.5g maneuver and are identified by symbol x.

LEGEND

Letter	Cond#	g's	Mach	Active Controls
A. O	142	+2.5	.88	ON
B. J	411	BRAKE		
C.E. K	132	+2.5	.82	ON
D. L	124	+2.5	.478	OFF
F. I	142	+2.5	.88	ON
G	143	0.0	.88	ON
H	451	TAXI		
M	122	+2.5	.478	ON
N	133	-1.0	.82	ON

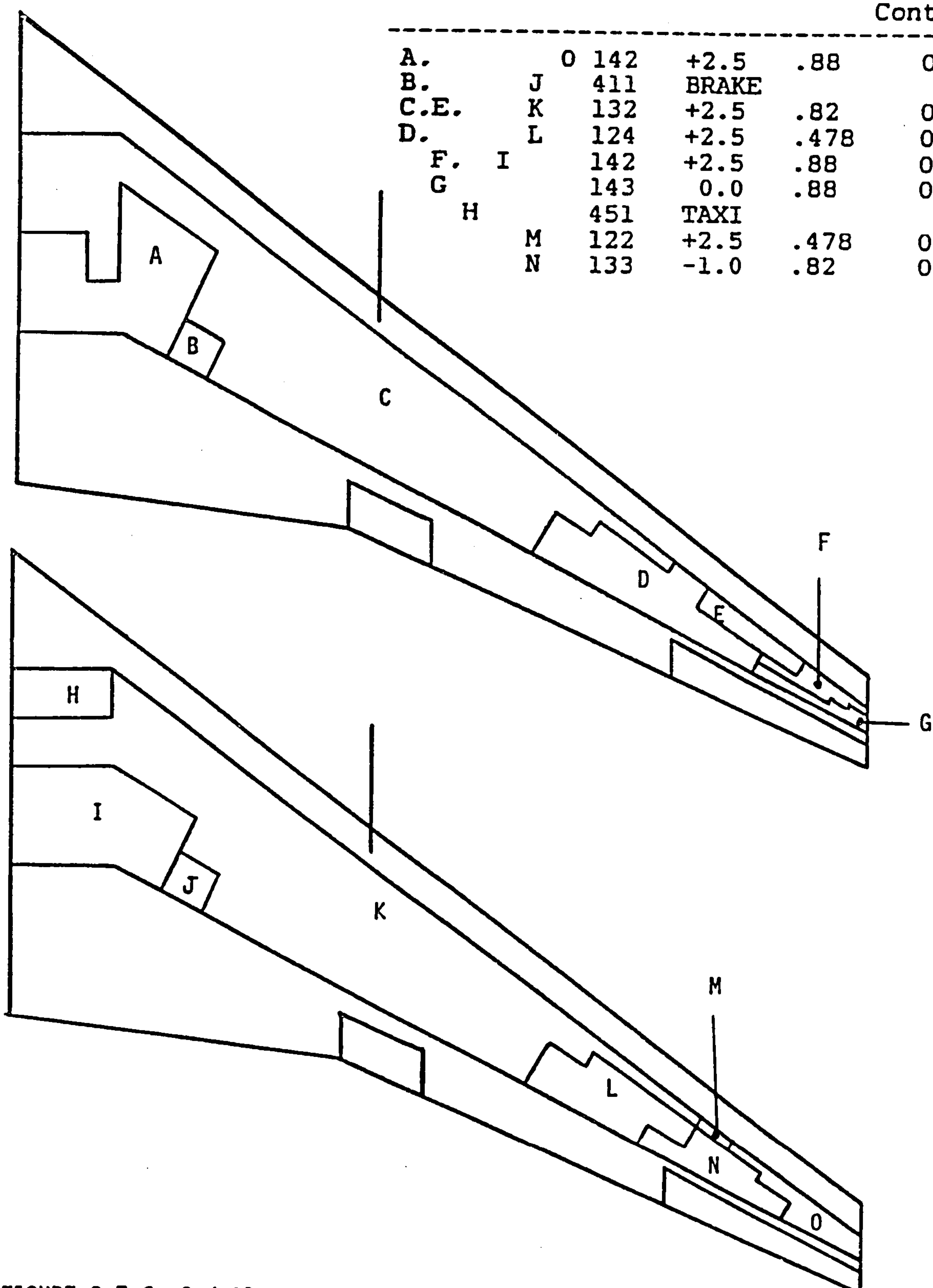


FIGURE 3.7-6 2nd Flex Loads Which Have Minimal Margin on Baseline

3.8 Flutter

3.8.1 Discussion of flutter model - A limited flutter analysis was conducted on the baseline aircraft for various flight conditions. Basic data such as mass, stiffness, and aerodynamic representations for the baseline aircraft were generated using PADS modules.

The analysis degrees of freedom used for the flutter survey were the stiffness degrees of freedom. The stiffness matrices used in all PADS analyses were constructed by NASTRAN. The element properties such as skin thickness and stiffener area which determine the stiffness of the model were the result of the PADS sizing process. The process in NASTRAN of how a stiffness matrix is produced and reduced down to the final analysis set is described in more detail in the section titled Finite Element Analysis.

The selection of the degrees of freedom which were to be retained as the analysis set (ASET in NASTRAN) required an examination of what type of information was required from the model. These degrees of freedom had to be selected early in the design stage with insight as to what was to follow. For the benchmarking exercise, as well as all contract requirements it was decided that the initial analysis would be with a symmetric representation. Table 3.8-1 contains a list of the degrees of freedom which were selected to represent the dynamic characteristics of the PADS model. Appendix D defines the location of each degree of freedom on the model.

TABLE 3.8-1 - Symmetric Analysis DOF for PADS Models

Quantity	Description
84	Wing structure z DOF
17	Fuselage z DOF
17	Horizontal stabilizer z DOF
4	Vertical tail z DOF
24	x, y, z, phi, theta, and psi DOF on wing engine, pylon, main landing gear up, and main landing gear down
9	x, z, and theta DOF on center engine, nose gear up, and nose gear down
2	Aileron actuator relative DOF
1	Stabilizer actuator relative DOF
3	x, z, and theta support DOF

161	

Construction of all mass matrices to be used for the PADS design studies consisted of execution of a series of PADS modules. The mass distribution module (MDM) described in section 3.3 produced the initial weight distribution matrices for the desired flight conditions. Weight matrices are produced for the left hand side of the aircraft in a left hand rule coordinate system. Locations and distributions for the lumped weights and inertias were computed from supplied input. Locations for the lumped weights are the centers of panels which for all PADS design work, were different than the locations of the analysis set to be used for the flutter survey. The output weight matrices, contained three degrees of freedom for each panel (x,y,z) and six degrees of freedom for each component represented as a single point concentrated mass. Weight matrices from the MDM are diagonal and are output with a corresponding PADS type geometry matrix. The number of degrees of freedom as output from MDM was 3765 degrees of freedom for the Baseline aircraft and 1101 for the Aspect Ratio 12 designs.

The difference in the basic weight matrix size was the result of how the input data was prepared for the MDM. In the Baseline case, many entries in the weight matrix were for zero mass and many entries were for identical degrees of freedom.

Reduction in size of MDM matrices was accomplished by a PADS module called POSTWTS. POSTWTS eliminates degrees of freedom not required for the analysis. In the PADS symmetric design studies, all x's and y's for points on the wing and vertical stabilizer, and all x's on the fuselage and vertical tail were eliminated because none of the grids requiring the mass included these degrees of freedom. This, in fact, was an oversimplification of the analysis grids data requirments for flutter and dynamic loads. Some x and y degrees of freedom on the wing and fuselage should have been retained in the analysis degree of freedom definitions. It was found that flutter results were not substantially changed when inertia relief in the x direction was included for the wing.

The number of degrees of freedom was reduced to 501 for the Baseline analysis weight matrix and 500 for the AR12 designs. These reduced size weight matrices were the starting weight data for the flutter survey as well as for all other PADS analyses.

The aerodynamics to be used for the flutter survey were produced by FLUT1. FLUT1 produces unsteady doublet lattice type aerodynamics for the left hand side of the aircraft. The unsteady aerodynamic model consisted of planar representations which extended to the centerline for the wing and horizontal stabilizer. The keypoints for the division of the planform into panels is depicted in figure 3.8-1. Spanwise divisions were selected in a 'looks reasonable' manner with biasing in areas where the pressure gradient was expected be the greatest. Chordwise divisions were made at 3, 10, 30, 55, 80, and 88 percent of the chord.

FLUT1 produced output geometry matrices for the force application and downwash points defined at the one-quarter chord and three-quarter chord points on the panel respectively. Unsteady aerodynamic matrices were produced for reduced frequencies of 0., 0.15, 0.3, 0.44, 0.76, 1.33, 1.95, 3.5, AND 6.8.

Transformation and sign change matrices necessary to obtain mass and aerodynamic matrices in the analysis degrees of freedom for the right hand side of the aircraft with a right hand rule sign convention were produced by FLUT2, the Grid Transformation Module. FLUT2 requires input of a PADS type geometry matrix for the weight, aerodynamic, and stiffness matrices. The stiffness geometry matrix is used for the formulation of rigid body mode shapes. Mode shapes for rigid body plunge and pitch are used for the symmetric analysis.

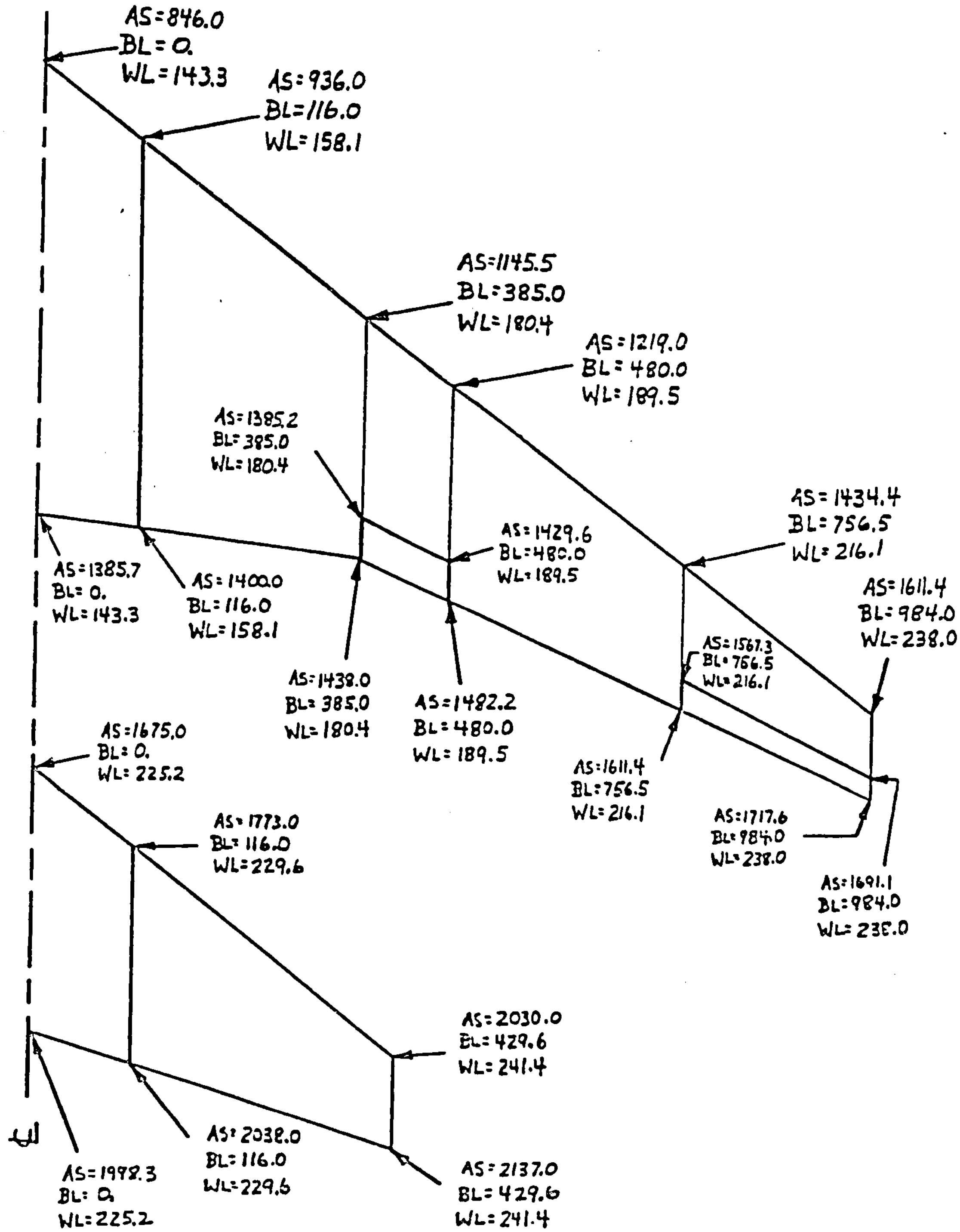


FIGURE 3.8-1 Keypoints for Unsteady Aero Panels

3.8.2 Flutter methodology - Flutter analyses in the PADS design process use vibration modes for an aircraft in a vacuum to modalize the problem. Modalization of the problem reduces the order of the problem to a size that can be handled.

The sign convention used for the vibration (and flutter) analyses is shown in figure 3.8-2.

The vibration equation of motion is:

$$(-w^2 [MM] + [KS])\{qSS\} = \{ 0 \}$$

where

$$[MM] = 1/386 * ([SHI] * [DI])^T * [W] * ([SHI] * [DI])$$

$$[KS] = [SHS]^T * [K] * [SHS]$$

[qSS] = analysis DOF for right side of aircraft using right hand rule

[W] = weight matrix defined for left side of aircraft using left hand rule

[K] = stiffness matrix defined for left side of aircraft using left hand rule

[DI] = transformation which relates weight DOF to analysis DOF

[SHI] = sign change matrix for weight matrix

[SHS] = sign change matrix for stiffness matrix

w = circular frequency

FLUT5, a PADS module, was used for the half airplane vibration analyses. Output from FLUT5 consists of the airplane vibration modes, natural frequencies, and the modalized mass and stiffness matrices.

The flutter equation of motion for the p-k method is:

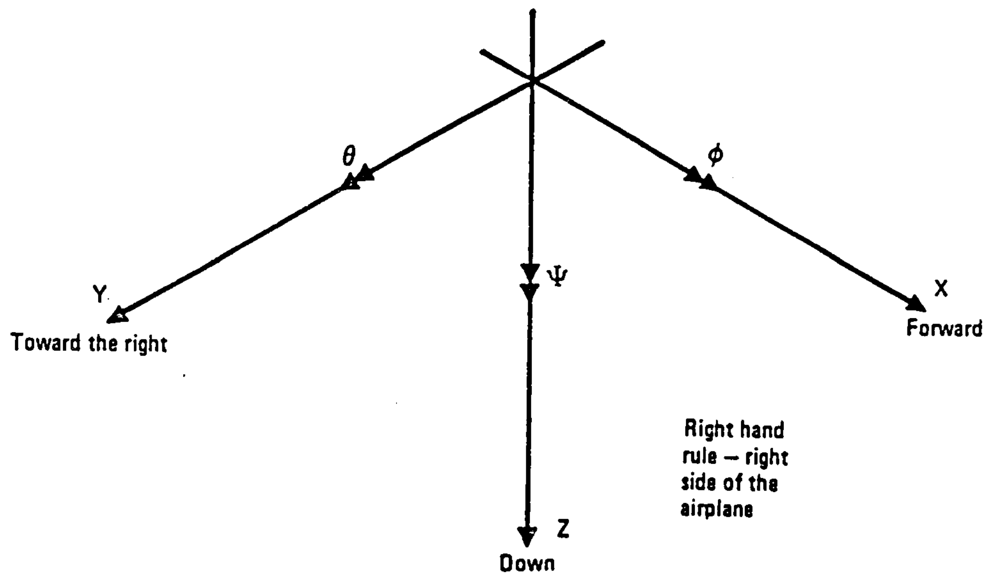


FIGURE 3.8-2 Flutter Sign Convention

$$[T]^T * \left((V/b_0)^2 * [MM] * p^2 + (1+ig) * [KS] \right.$$

$$\left. \frac{1}{2}(\rho_0) V^2 (\sigma) * [A(k)] \right) * [T] * \{q\} = \{0\}$$

where

$$[A(k)] = [DFZ]^T * [SHQ]^T * [A] *$$

$$\left(i(k/b_0) * [SHA] * [DAZ] + [SHA] * [DATH] \right)$$

[A] = FLUT1 aerodynamic matrix

$[DFZ]^T$ = transformation which relates aero force DOF to analysis DOF

[DAZ] = transformation which relates aero panel displacements to analysis DOF

[DATH] = transformation which relates aero panel rotations to analysis DOF

[T] = vibration modes with synthetic rigid body modes replacing zero frequency modes

$[SHQ]^T$ = sign change for aero forces

[SHA] = sign change for aero displacements

{q} = modal displacements

p = nondimensional operator $(b_0/V)(d/dt)$

g = structural damping

k = reduced frequency $(\omega)(b_0/V)$

V = true velocity

b_0 = reference length

omega = circular frequency

rho_0 = air density at sea level

sigma = air density ratio, rho/rho_0

FLUT6, a PADS module, was used for the flutter analyses. Output from FLUT6 consists of vfg plots. (freq. vs. velocity and damping vs. velocity)

3.8.3 Vibration results - Vibration solutions were run for the Baseline aircraft and compared to ground vibration results as well as results from the final flutter model (Stick model) used for the study of the L-1011-500 aircraft (Job 5352 Sect 701). Table 3.8-2 compares symmetric vibration result for the first ten natural frequencies for a full fuel weight condition. PADS stiffness model section 9002 refers to the stiffness representation where wing panel property dimensions were taken from drawings and entered into NASTRAN for computation of a stiffness matrix (production stiffness). PADS stiffness model section 300 (Job 5580) is from the second flex sizing (no margins). PADS vibration results are from Job 5750 RUN046 for stiffness section 9002 and Job 5699 RUN016 for stiffness section 300. The fuselage for PADS section 300 stiffness representation has buckled skin whereas the other two stiffness representations do not.

TABLE 3.8-2 - FREQUENCY COMPARISON FOR VIBRATION RESULTS

Mode #	Description	PADS Model S9002	PADS Model S300	GVT	Job 5352 Sect 701
1	Rigid body z	0.0	0.0		0.0
2	Rigid body theta	0.0	0.0		0.0
3	Wing 1st bending	1.24	1.13	1.24	1.20
4	Wing eng. up & out	2.32	2.17	2.33	2.31
5	Wing eng. down & out	2.72	2.67	2.78	2.80
6	Fus. 1st bending & wing 2nd bending	3.11	2.95		3.01
7	Wing 2nd z	3.81	3.63	4.20	3.97
8	Horiz. stab 1st	4.27	4.22	4.88	4.81
9	Wing eng. yaw/roll	6.44	6.41		6.26
10	Wing eng. pitch, wing 3rd bending	6.28		6.82	6.73

* note - GVT results are from LR 24045 Aug. 1971 ,Rev. B

3.8.4 Flutter analysis results - Flutter analyses were run on a selective basis. Full and min fuel conditions were run for the final PADS sized design (SECT 300) as well as for the final production stiffness design (SECT 9002). The flutter equation was modalized with 20 vibration modes plus 2 synthetic rigid body modes for most of the PADS flutter work. Structural damping is represented as being equal to 2 percent of the stiffness.

The flutter survey for both of the Baseline models did not show any unstable roots below 1.2Vd. Flutter plots (vfg) for the the full fuel production stiffness PADS model (Job 5750 RUN046) are shown in figures 3.8-3 through 3.8-5. Figure 3.8-3 is a plot of frequency (cycles per second) versus velocity (KEAS) for the first ten modes. Figures 3.8-4 and 3.8-5 are plots of damping versus velocity.

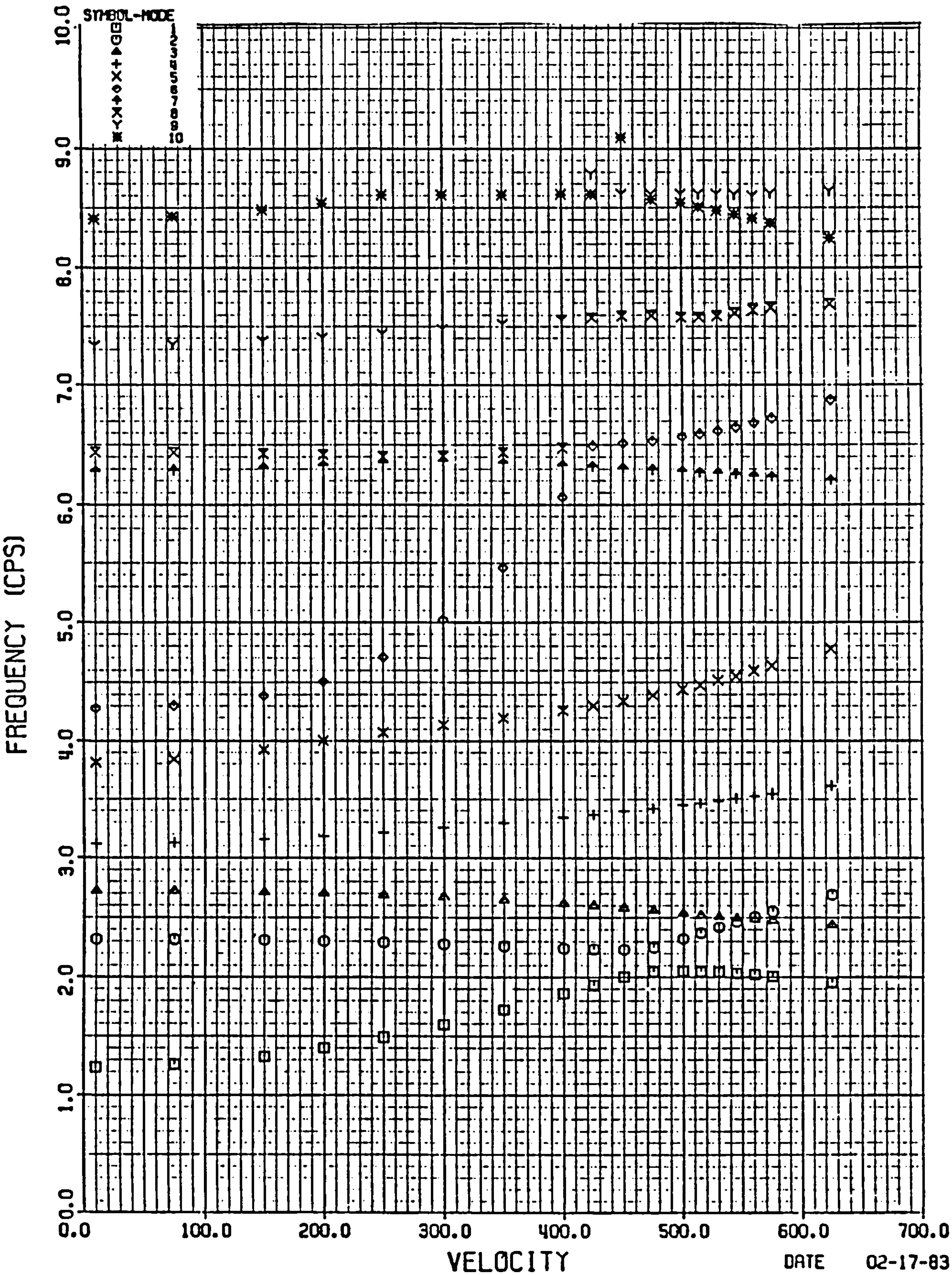


FIGURE 3.8-3 Full Fuel Flutter Plot - Frequency vs. Velocity

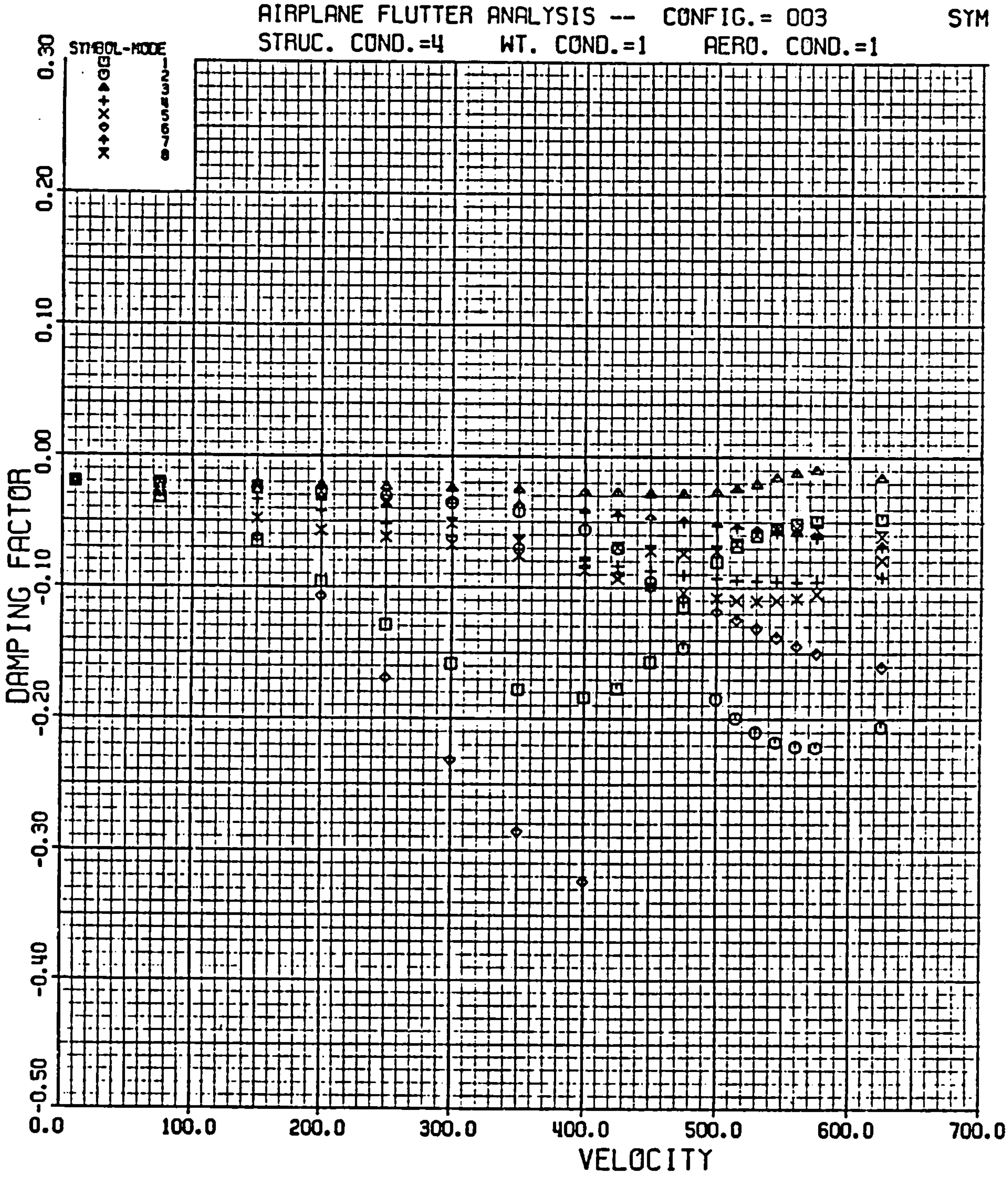


FIGURE 3.8-4 Full Fuel Flutter Plot - Damping vs. Velocity for Modes 1-8

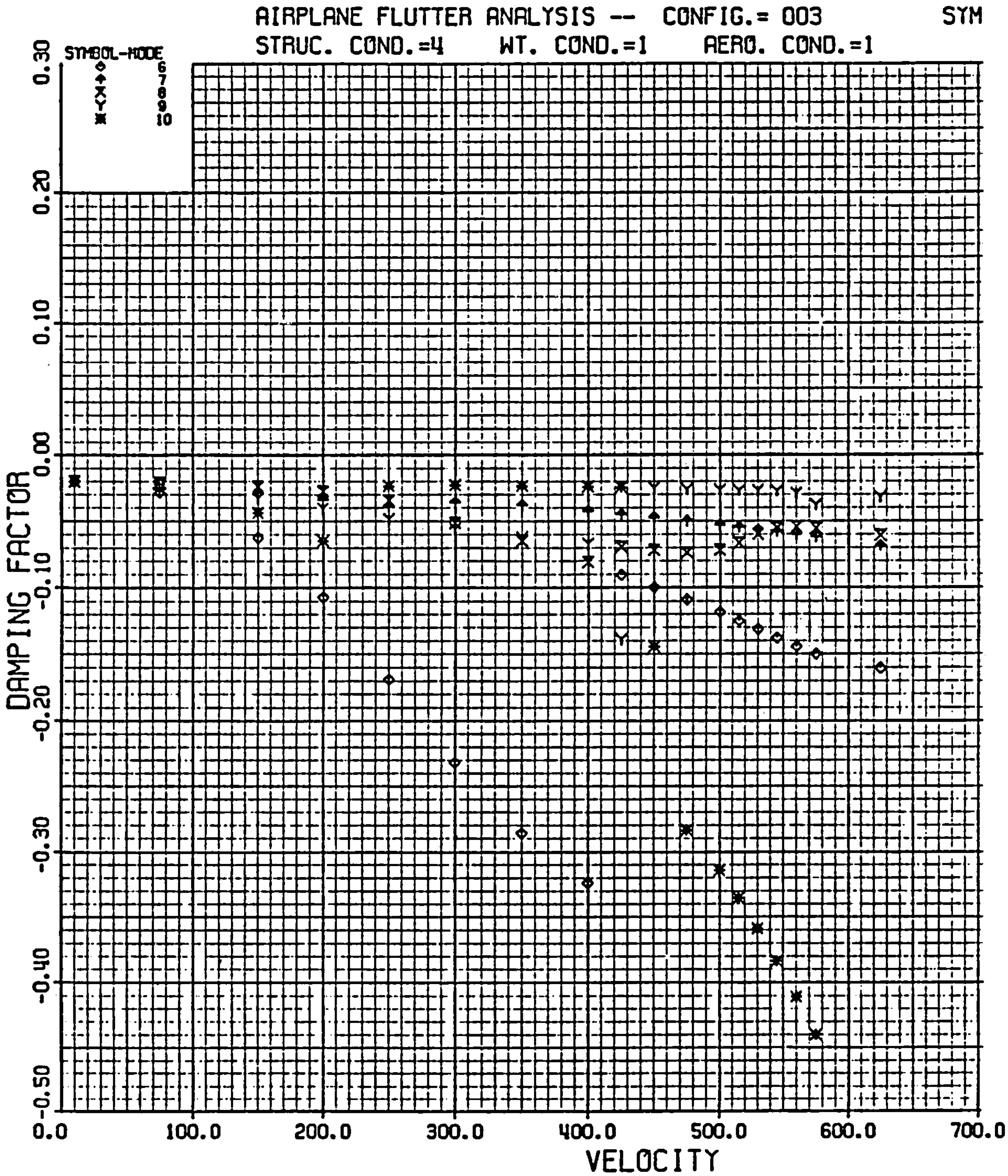


FIGURE 3.8-5 Full Fuel Flutter Plot - Damping vs. Velocity for Modes 6-10

3.9 Discussion Of Dynamic Gust Loads

Many airplanes (such as large transports) can be assumed to be potentially gust-critical. If the sizing in a preliminary design analysis is influenced by consideration of turbulence, this may affect (the work of) other disciplines. It is therefore important to include loads due to turbulence early in the design.

The method for computation of loads due to turbulence uses a gust input defined as a power spectral density at an rms value of 1 ft/sec of the gust velocity. This defines the power spectrum of the responses, which is integrated over the frequency domain to give the rms of the responses due to the 1 ft/sec rms gust. For a final design, mission profiles can be established, and the responses in these missions can be computed, using specified gust environment for the profile altitudes.

For a preliminary design, or in the absence of defined missions, the responses can be computed for a number of points of the design speed-altitude envelope, much in the same way that static gust analysis is done. The design envelope contains the critical values of weight distribution, flight speed, and altitude. In this case the responses are found by multiplying the rms values of responses due to 1 ft/sec gust by the gust intensity factor. This gust intensity factor is a function of altitude (see Reference 4), comparable to specified gust velocities used in static analyses. To properly account for the phasing between the various responses, correlation coefficients between these responses are also computed. From these correlation coefficients and using the rms values of the responses, ellipses of equal probability of selected response combinations are formed.

These are biased by the response due to 1-g steady flight for the pertinent design envelope point. In order to limit the number of conditions to be analyzed for stress, the ellipses are circumscribed by octagons. Thus each load combination results in eight points of an octagon of equal probability as shown in figure 3.9-1.

In the presently used methods the loads computed in the gust loads analysis are external loads. These are shears, bending moments, and torsion moments relative to selected load axes for the various airplane components, at selected cross sections. Octagons are formed for these external loads. The usual external load combinations selected are bending moments combined with torsion moments and/or torsion moments combined

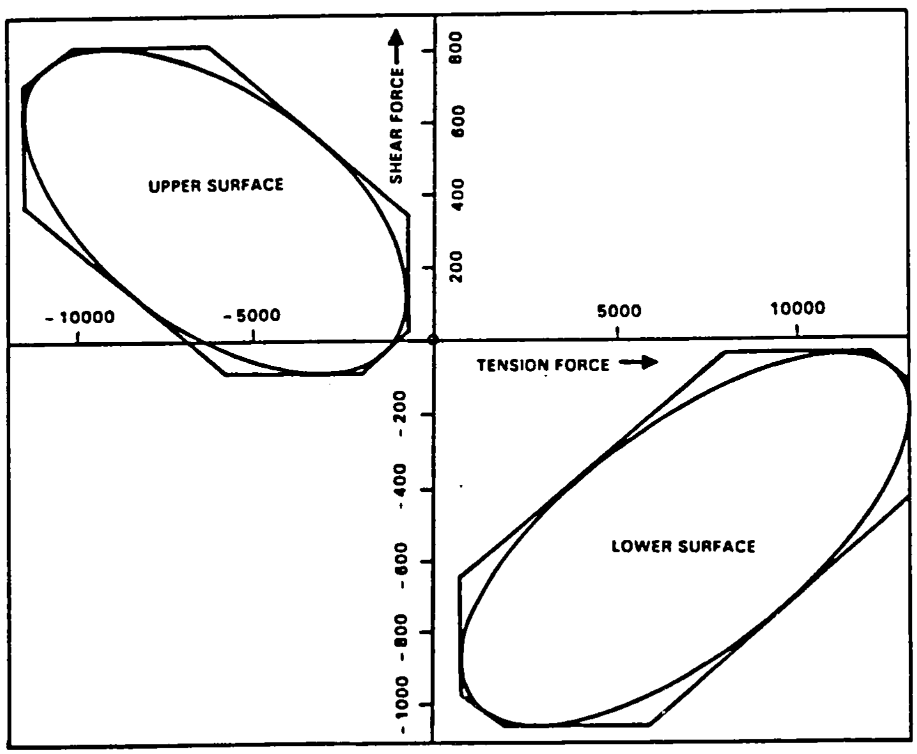


FIGURE 3.9-1 Octagons of Equal Probability

with shear forces. Note that load octagons at different stations are not correlated.

The points of the octagons are used to determine the critical conditions. Stress analysis by finite element models must use sets of balanced loads at the SIC grid points as conditions. To represent the loads combinations found in the gust load analysis as sets of SIC grid point loads' a set of elementary load distributions at the SIC grid points is defined. The selection of these elementary conditions can be rather arbitrary. The only requirement is that all pertinent points of the equal probability octagons can be reasonably well simulated by linear combinations of the elementary load distributions. Each of these sets of elementary loads must be balanced (that is, they must be in self-equilibrium). This way linear combinations of the elementary distributions, resulting in external loads which match exactly, or conservatively, the critical points of the equal probability octagons, will also be balanced. Thus a number of balanced load conditions, consisting of sets of loads on the SIC grid points, is created which in combination with other such inputs (e.g., from static loads or ground loads), are used to compute internal loads acting on the structural elements.

The new method described here, to be used in the PADS system, is identified as "INTERnal LOad Procedure", (INTERLOP). The basic feature of INTERLOP is that the internal loads combinations are computed directly in the gust analysis. This requires the input of all internal loads of interest due to all unit loads at each SIC grid point separately. The computer cost to obtain internal loads for each SIC node unit external load has been found to be acceptable. It should be noted that these loads by themselves would not be balanced; each of these unit loads is reacted at the support points. However, when applied in combination in the gust analysis program the totality of these loads will be balanced by definition, and the reactions at the supports cancel.)

The advantages of the use of INTERLOP are substantial. Not only is the costly and time consuming process of matching conditions avoided, but also, the internal vs external loads matrix is used for other loads analyses besides gusts. Furthermore, since the internal loads are computed directly, no additional conservatisms need to be introduced, which is next to unavoidable if matching conditions are used.

The modified version of GLP5 which is part of the procedure, here named INTERLOP includes accepting AIC matrices as in the flutter analysis. (These include AIC's at several reduced frequencies.) The program is extended to compute the eight corners of the octagons of equal probability, and include the bias formed by the 1-g steady loads. The latter is obtained using the 1-g (external) SIC grid point loads computed by STATICS. The dimension of the octagons is determined by an input value, which is the predicted gust intensity factor in feet/second. This is based on the use of the design envelope approach rather than a mission profile analysis. The former, because of its simplicity, is considered to be more appropriate for a preliminary design effort, in particular because the missions may not yet be completely defined.

Figure 3.9-2 shows the input and output structure of the two programs, which can be combined in a single PADS run.

3.9.1 Results - GLP5K has been used to obtain octagons of equal probability for one condition for the Baseline and AR12, 35 Degree Sweep airplane configurations. This condition was selected as cruise at high altitude with low fuel weight. This was expected to be the most critical condition for gust response. The PADS sizing procedures were exercised for applying Dynamic loading conditions simultaneously with final second flex Static loads on the Baseline aircraft. The finite element model design regions for the Baseline aircraft wing are represented in figure 3.7-5. Internal loads for the wing were computed in a NASTRAN static solution run. Panel sizing and stress allowable (PSASA) module was used for sizing of the surface panels.

The loads applied to the structure for sizing include combining the Static loads for the second flex aircraft as shown in table 3.4-2 with the eight loads of equal probability formed by the gust analysis. Figure 3.9-3 shows the load conditions which have the minimal margin of safety for each panel for upper and lower surfaces, respectively. Gust conditions were found to be the designing factor for sections near the root rib and wing tip for both upper and lower surfaces. These areas are marked with gs. Braking conditions determined the sizing around the main gear. Wing midsection panels were designed by a 2.5g maneuver and these panels are identified by symbol x.

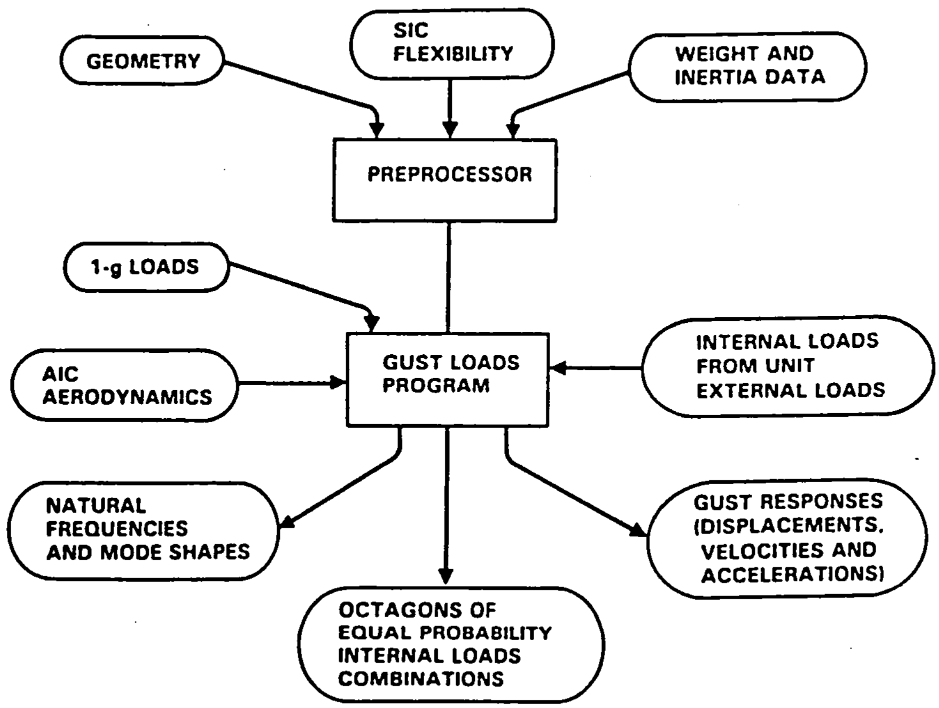


FIGURE 3.9-2 INTERLOP Flow Chart

SYM.	COND#	g's	MACH	ACTIVE CONTROLS
G	1-8	GUST LOADS		ON
+	124	+2.0	.478	OFF
X	132	+ 2.5	.82	ON
*	133	- 1.0	.82	ON
●	142	+ 2.5	.88	ON
▬	143	0.0	.88	ON
B	411	BRAKE		
T	451	TAXI		

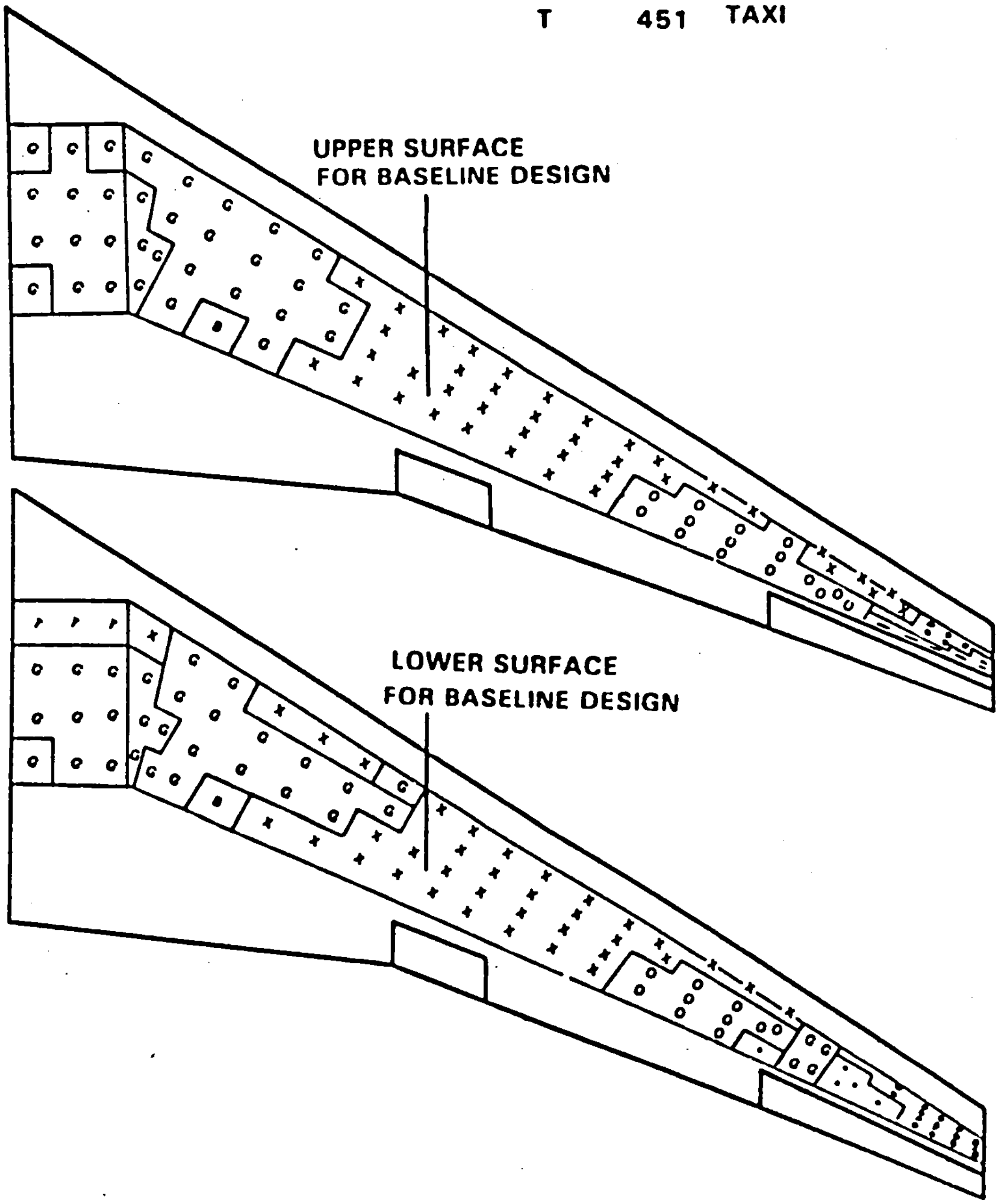


FIGURE 3.9-3 Load Conditions Designing Baseline Design (Combined Gust and Static Loads)

4.0 ASSET CONFIGURATION STUDY FOR MINIMUM BLOCK FUEL

ASSET is an upper level optimizer which uses parametric data and/or small models to represent various disciplines, such as propulsion, aerodynamics, vehicle geometry, weights, and subsystems. ASSET integrates these inputs and derives candidate vehicles which satisfy given mission requirement. A schematic presentation of the primary input and output data in the ASSET synthesis cycle is shown in figure 4.0-1. A generalized schematic illustrating key elements and the flow of information through the ASSET program is shown in figure 4.0-2. Further information on ASSET is available in reference 7, chapter 3.

The ASSET configuration study was performed using the baseline airplane as a starting point. The design variables were aspect ratio and thickness to chord. First, an aspect ratio and t/c was defined which produced minimum block fuel. Then ASSET was exercised to establish the block fuel sensitivity to the sweep angle. Reference 7 provided some guidance in this area and a sweep of 25 degrees was selected for the study. Since airfoil technology was to remain fixed during this design process, block fuel sensitivity to mach number study was performed at the reduced sweep angle.

The above plan for the initial ASSET study was extremely limited in its scope. The primary aeroelastic objective of the wing optimization study is to generate point designs in PADS and use that data to update the ASSET parametric weight data. The weight team members were concerned with simple changes to the baseline configuration for input to the PADS process. There was never any attempt to fully optimized an ASSET configuration and then use the ASSET optimized configuration for a PADS design. The ASSET studies were used only as one of many inputs to the PADS design configuration selection process. After the ASSET studies were performed, a consensus was formed to generate two PADS designs. The first design changed the aspect ratio from 7.64 to 12, relative to the baseline airplane. The second design changed the sweep from 35 degrees to 25 degrees, relative to the first PADS design. These PADS designs were made in accordance with the basic function accorded PADS in its interface with ASSET, namely; PADS supports the ASSET's weight database for the sizing of the vehicle.

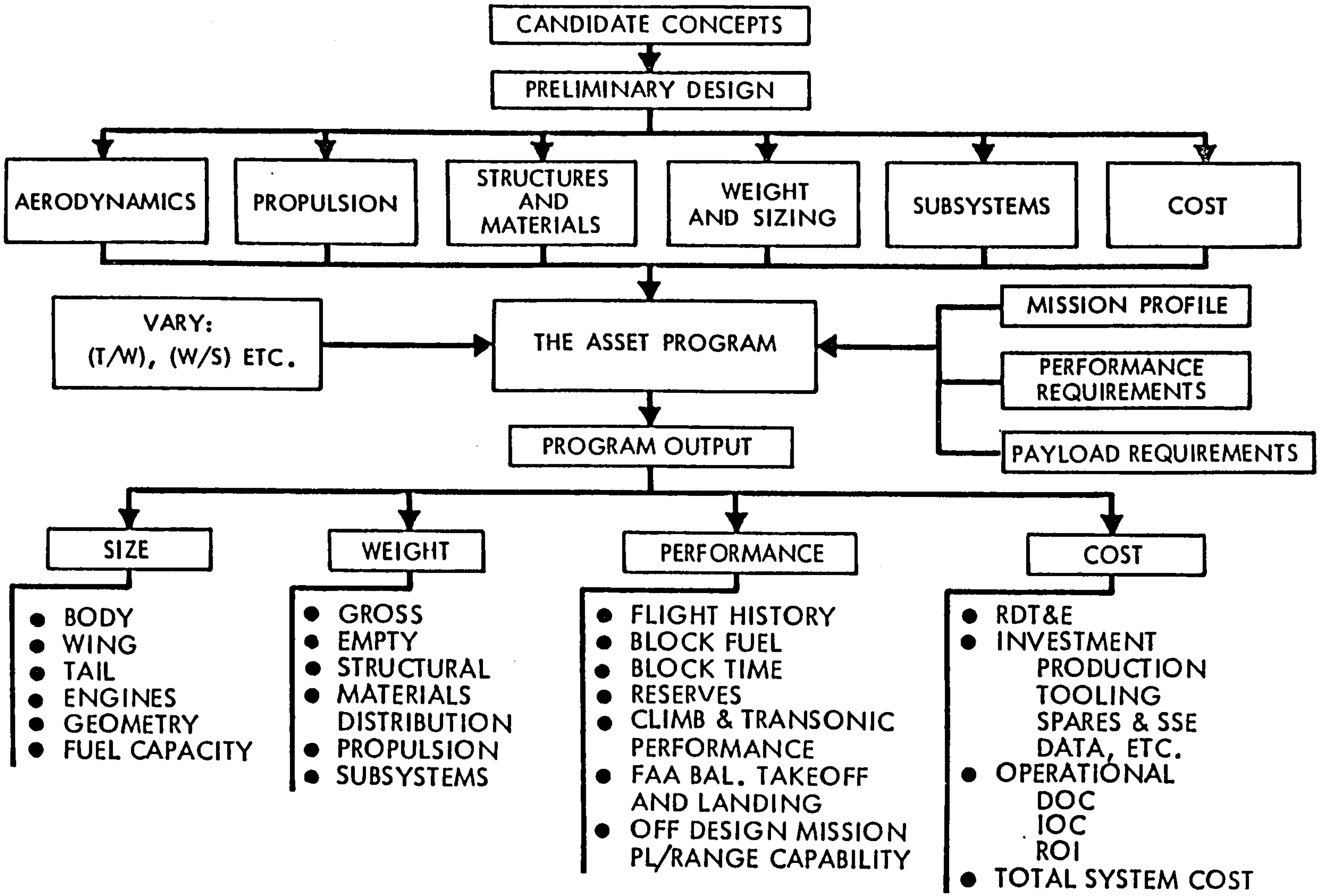


FIGURE 4.0-1 The ASSET synthesis cycle

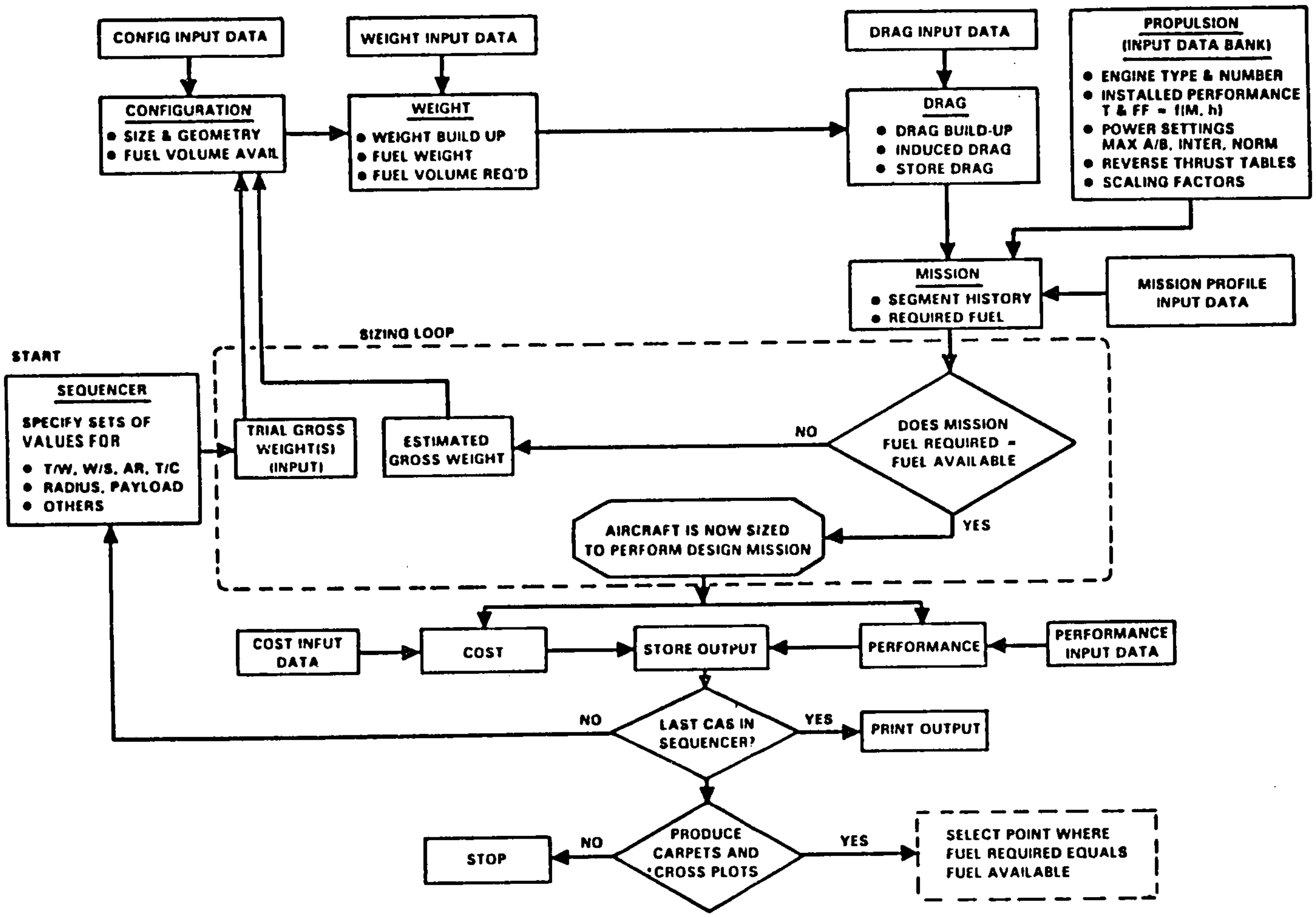


FIGURE 4.0-2 ASSET program schematic

4.1 Baseline Airplane Parametric Study At 35 Degrees Of Sweep

The ASSET computer program was given input data that corresponded to the Baseline airplane model. Using the standard weight equation, the Baseline model on ASSET was run with the following variations in wing parameters:

AR = 10, 11, 13

t/c = 9, 10, 11, 12

For each configuration the aircraft was sized to perform the design range of 4780 n.mi.

Block fuel at the design range is shown in the carpet plot of figure 4.1-1. The same data are shown as a knot-hole plot in figure 4.1-2. It shows that the optimum wing parameters for minimum block fuel are (approximately):

AR = 12

t/c = 10.75

A listing of aircraft characteristics is given in table 4.1-1. Note that in keeping thrust to weight ratio (T/W) and wing loading (W/S) fixed, the field length constraints become variables. To be more precise, T/W and W/S should be adjusted to meet fixed constraints. The approach used in this exercise will not significantly affect results.

A summary of ASSET runs is shown in table 4.1-2.

Plots of fuel weight, wing weight, zero fuel weight, OEW weight, wing area, body weight, and engine scale versus aspect ratio at constant t/c are shown in figures 4.1-3 through 4.1-9. These plots show variations of key aircraft parameters in addition to fuel weight. Engine scale is a factor on the engine thrust requirements relative to the baseline airplane. The definition of the other variables are either contained in the other sections or assumed to be known through accepted usage.

Selected ASSET outputs for parameters, mission summary, and weight summary are shown in table 4.1-3 through 4.1-5 for AR = 12 and t/c = 10.

The results of this study indicated that PADS computer run should be made for AR = 12. It was decided to hold t/c constant in going from the baseline AR = 7.6 to AR = 12.

From the advanced transport study found in reference 7, it was also found that the sweep = 25 degrees and AR = 12 was a possible optimized configuration. It was then decided to vary sweep from 35 degrees to 25 degrees for the second derivative configuration. The design of wings for AR = 12, sweep = 25 degrees and 35 degrees, are the subject of the following sections, after which another ASSET study is performed.

ASSET uses aerodynamic data in parametric form which is based on windtunnel data generated for a specific technology wing. The database also reflects studies performed on 3-D aerodynamic flow codes. A number of drag buildup computer output sheets were forwarded to NASA for used in tuning NASA's performance module to better reflect Lockheed's database in ASSET.

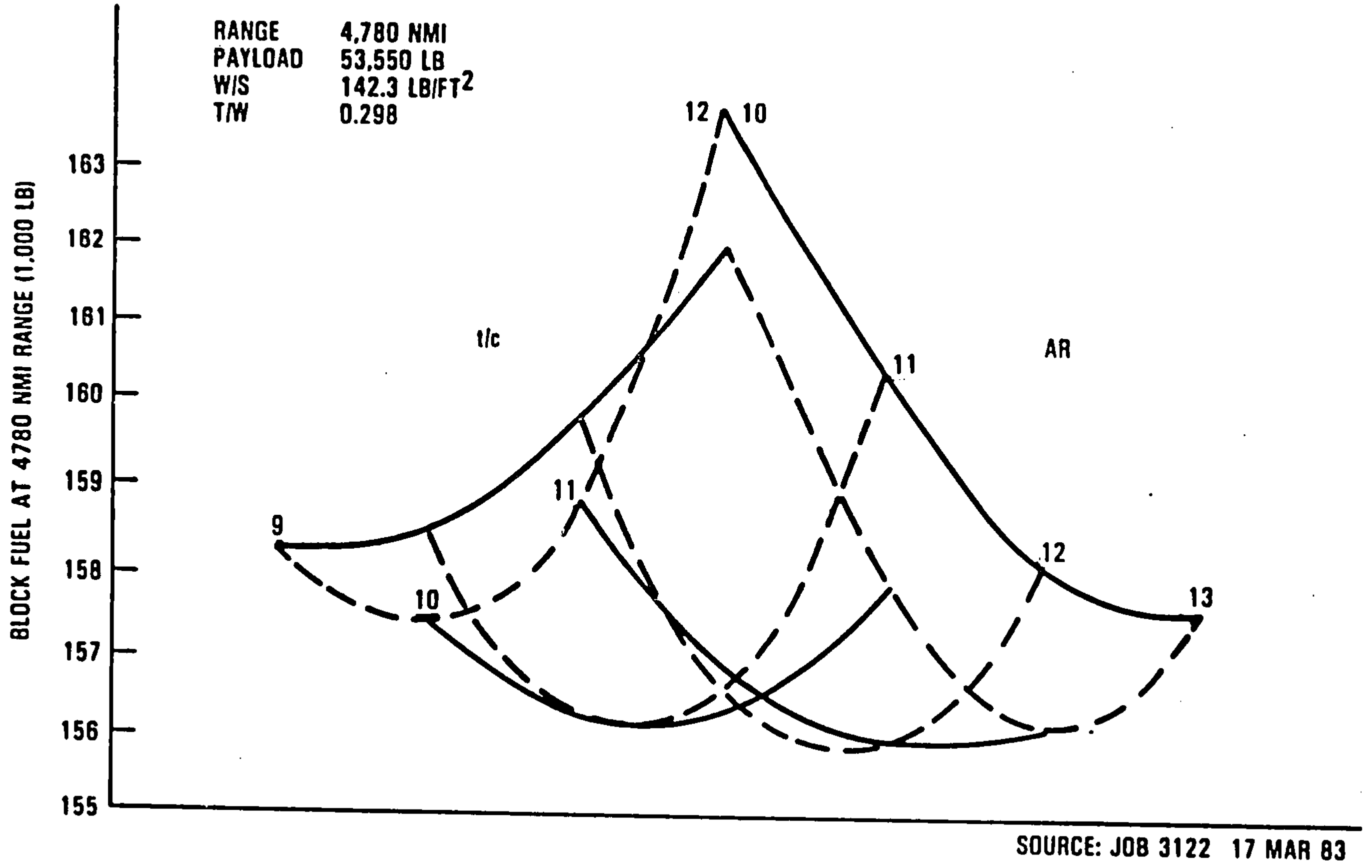


FIGURE 4.1-1 Wing parameter variation for std. weight equation

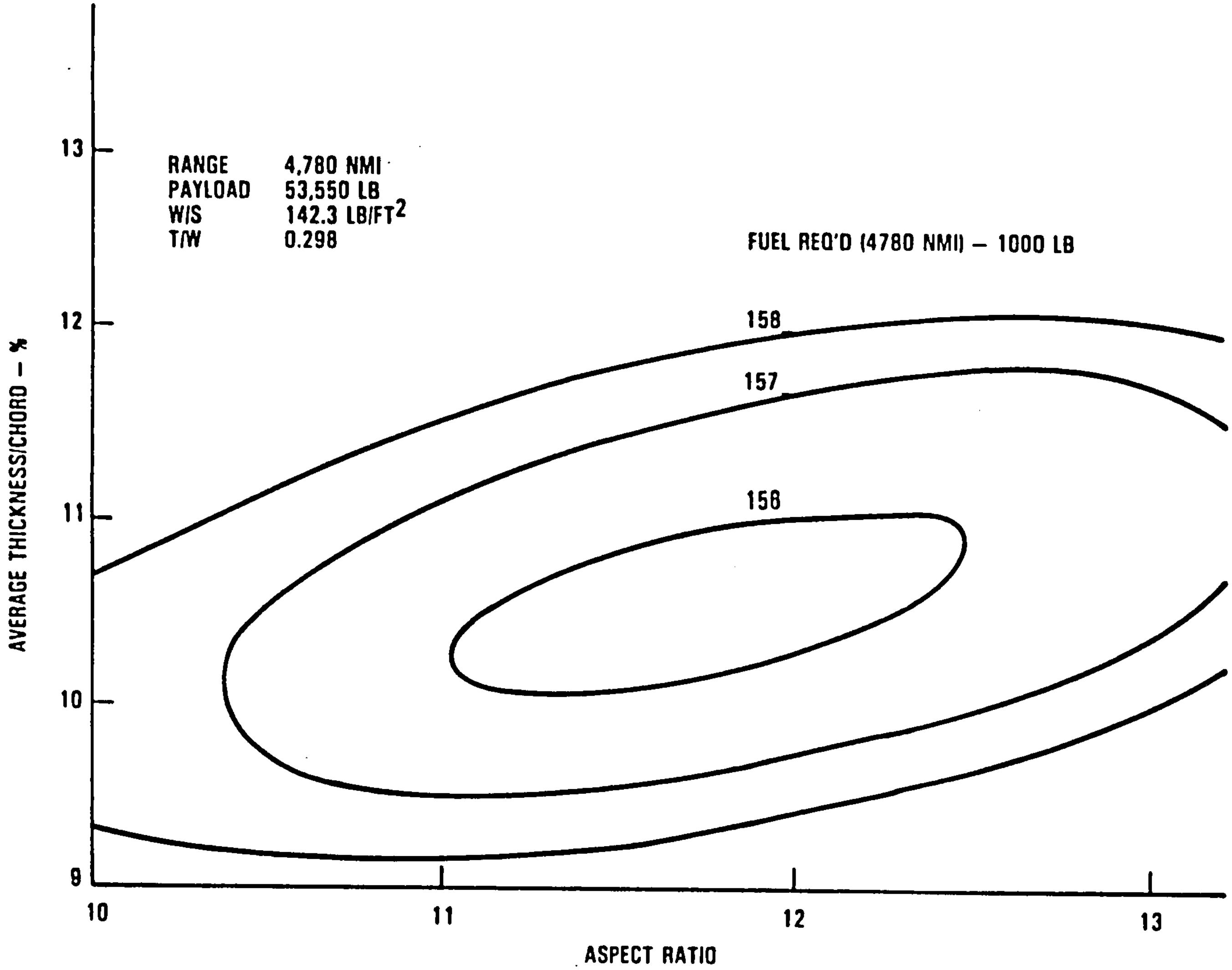


FIGURE 4.1-2 Block fuel (lb) knothole for std. weight equation

TABLE 4.1-1 BASELINE MODEL PARAMETRIC ANALYSES AIRCRAFT CHARACTERISTICS

Payload	53550 lb
Range	4780 n.mi.
W/S	142.3 lb/ft^2
T/W	0.298
Sweep	35 degrees
Cruise Mach	0.83
Aerodynamics:	L-100 (Wing 37B)
Systems:	L-1011 Active Controls
Materials:	L-1011
Propulsion:	RB211-524B4

Constraints:

All-engine takeoff distance	=	7749 - 7967 ft (84 deg.F SL) @ Design Range
Engine-out takeoff distance	=	7674 - 7891 ft (84 deg.F SL) @ Design Range
Approach speed	=	151.1 - 154.1 kt (TAS)

TABLE 4.1-2 Asset parametric analyses for baseline and std weight equation

AIRCRAFT MODEL
I.O.C. DATE --1980
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 140000
SLS SCALE 1.0 = 50000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 35.00 DEG
WING TAPER RATIO = 0.259

1 W/S	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3	142.3
2 T/W	0.290	0.298	0.298	0.298	0.298	0.298	0.298	0.298	0.290	0.298	0.298	0.290	0.290	0.298	0.290	0.298
3 AR	10.00	11.00	12.00	13.00	10.00	11.00	12.00	13.00	10.00	11.00	12.00	13.00	10.00	11.00	12.00	13.00
4 T/C	9.00	9.00	9.00	9.00	10.00	10.00	10.00	10.00	11.00	11.00	11.00	11.00	12.00	12.00	12.00	12.00
5 SWEEP	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 HPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4700	4780	4700	4700	4780	4780	4700	4700	4700	4700	4700	4780	4700	4700	4700	4780
12 GROSS WEIGHT	521145	534723	551070	570459	510174	519043	532442	548291	504636	511558	521115	532730	505368	509615	515570	525105
13 FUEL WEIGHT	187165	187068	189779	192950	185709	184045	185516	187815	187039	185083	184610	185134	192101	188927	186623	186524
14 OP. WT. EMPTY	280430	293305	307741	323960	270916	281448	293377	306926	264047	272925	282755	294046	257637	267130	275397	285031
15 ZERO FUEL WT.	333700	346855	361291	377510	324466	334998	346927	360476	317597	326475	336505	347596	313187	320680	328947	338581
16 ENGINE SCALE	1.034	1.061	1.093	1.132	1.012	1.031	1.056	1.088	1.001	1.015	1.034	1.057	1.003	1.011	1.023	1.042
17 THRUST/ENGINE	51698	53044	54666	56590	50609	51568	52818	54370	50060	50747	51695	52847	50133	50554	51145	52090
18 WING AREA	3662.	3757.	3872.	4008.	3584.	3652.	3741.	3852.	3546.	3594.	3661.	3743.	3551.	3581.	3622.	3689.
19 WING SPAN	191.4	203.3	215.5	228.3	189.3	200.4	211.9	223.8	188.3	198.8	209.6	220.6	188.4	198.5	208.5	219.0
20 H. TAIL AREA	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0
21 V. TAIL AREA	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0
22 ENG. LENGTH	10.31	10.60	10.95	11.36	10.08	10.28	10.55	10.89	9.96	10.11	10.31	10.56	9.98	10.07	10.19	10.40
23 ENG. DIAMETER	7.27	7.36	7.48	7.61	7.19	7.26	7.35	7.46	7.15	7.20	7.27	7.35	7.16	7.19	7.23	7.30
24 BODY LENGTH	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2
25 WING FUEL LIMIT	1.079	1.070	1.065	1.064	1.169	1.157	1.148	1.143	1.255	1.240	1.228	1.220	1.336	1.317	1.304	1.292
COST DATA																
26 RDTE - BIL.	2.739	2.863	3.004	3.164	2.647	2.748	2.864	2.996	2.581	2.666	2.762	2.870	2.538	2.610	2.689	2.782
27 FLYAWAY - MIL.	61.00	63.13	65.53	68.23	59.44	61.17	63.14	65.40	58.34	59.79	61.44	63.28	57.71	58.91	60.25	61.84
28 INVESTMNT-BIL.	21.387	22.119	22.943	23.872	20.848	21.441	22.119	22.876	20.474	20.968	21.534	22.165	20.260	20.670	21.127	21.673
29 DOC - C/SM	4.286	4.337	4.410	4.505	4.236	4.250	4.306	4.380	4.232	4.234	4.260	4.304	4.209	4.268	4.264	4.294
30 IOC - C/SM	2.388	2.398	2.411	2.426	2.380	2.386	2.396	2.408	2.376	2.380	2.387	2.396	2.379	2.381	2.384	2.391
31 ROI A.T. - 0/0	-28.09	-27.96	-28.03	-28.27	-28.02	-27.62	-27.52	-27.67	-28.43	-27.84	-27.55	-27.44	-29.56	-28.72	-28.08	-27.87
MISSION PARAMETERS																
32 MISN V1(1,1)	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000	31000
33 MISN V2(1,1)	158346	158534	159786	162096	157403	156202	156414	157965	158078	156733	155944	156073	163750	160474	158068	157670
CONSTRAINT OUTPUT																
34 TAKEOFF DST(1)	7952	7851	7777	7749	7960	7860	7787	7760	7966	7867	7795	7769	7967	7870	7800	7774
35 CLIMB GRAD(1)	0.1426	0.1473	0.1512	0.1545	0.1424	0.1470	0.1509	0.1542	0.1422	0.1468	0.1507	0.1539	0.1421	0.1467	0.1505	0.1537
36 TAKEOFF DST(2)	7890	7782	7711	7695	7887	7776	7702	7683	7888	7775	7690	7676	7891	7777	7698	7674
37 CLIMB GRAD(2)	0.0573	0.0615	0.0648	0.0672	0.0574	0.0617	0.0651	0.0677	0.0574	0.0618	0.0653	0.0680	0.0572	0.0617	0.0653	0.0681
38 CTOL LDG D(1)	7028	7077	7131	7191	6990	7028	7073	7126	6970	6999	7036	7077	6973	6992	7017	7052
39 AP SPEED-KT(1)	151.9	152.6	153.3	154.1	151.4	151.9	152.5	153.2	151.1	151.5	151.9	152.5	151.1	151.4	151.7	152.1

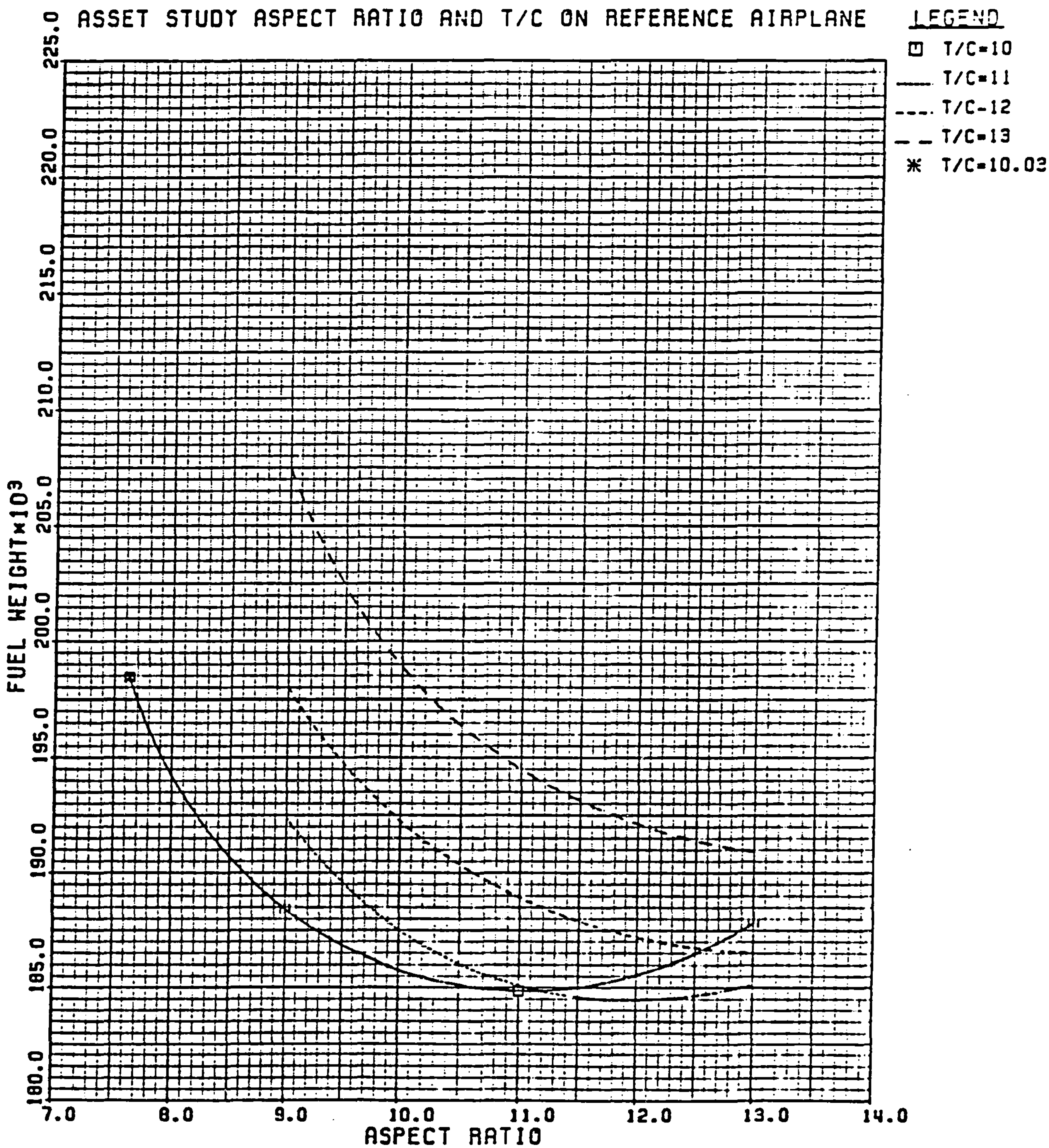


FIGURE 4.1-3 Block fuel weight (lb) vs aspect ratio for std weight equation

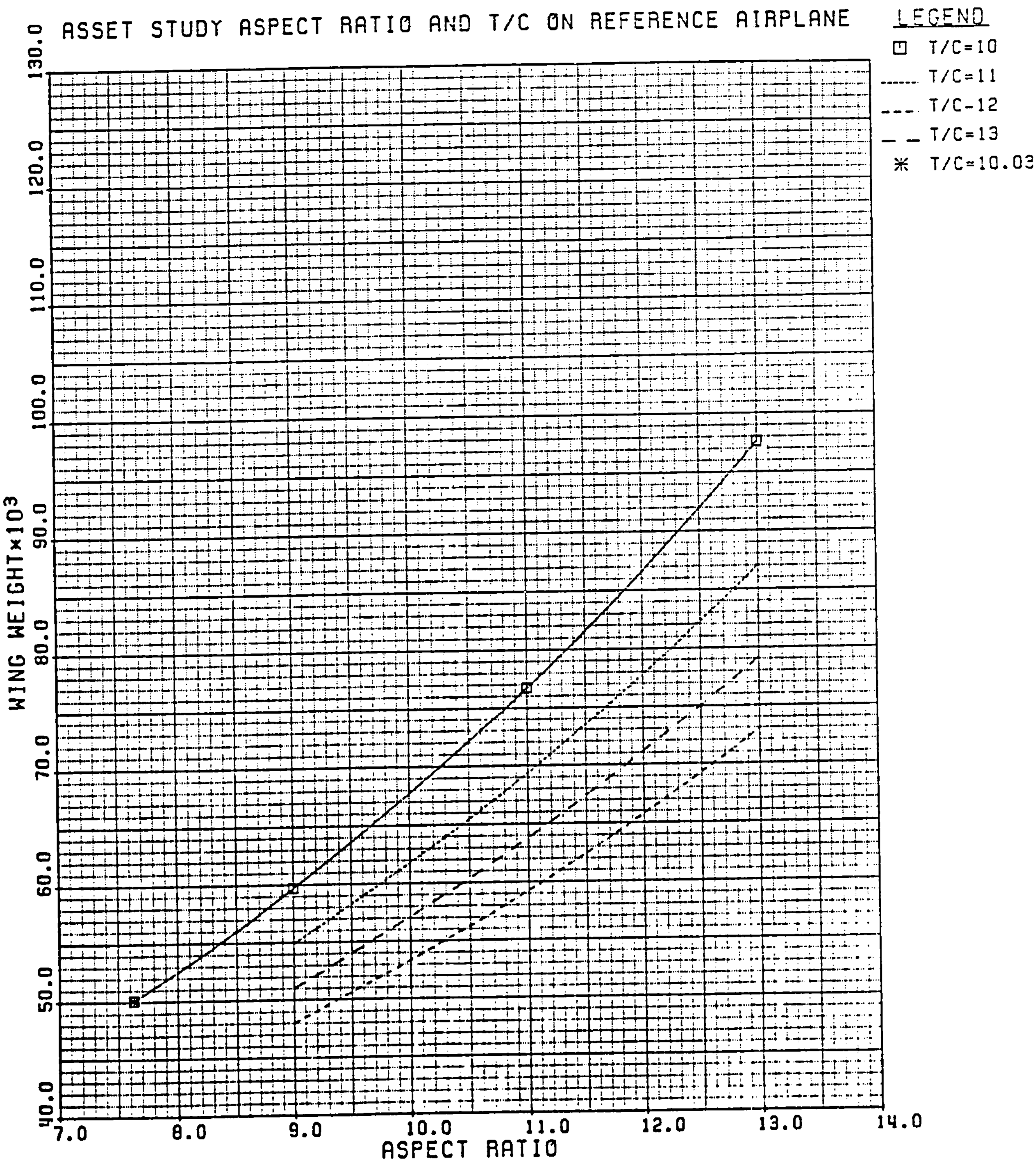


FIGURE 4.1-4 Wing weight (lb) vs aspect ratio for std weight equation

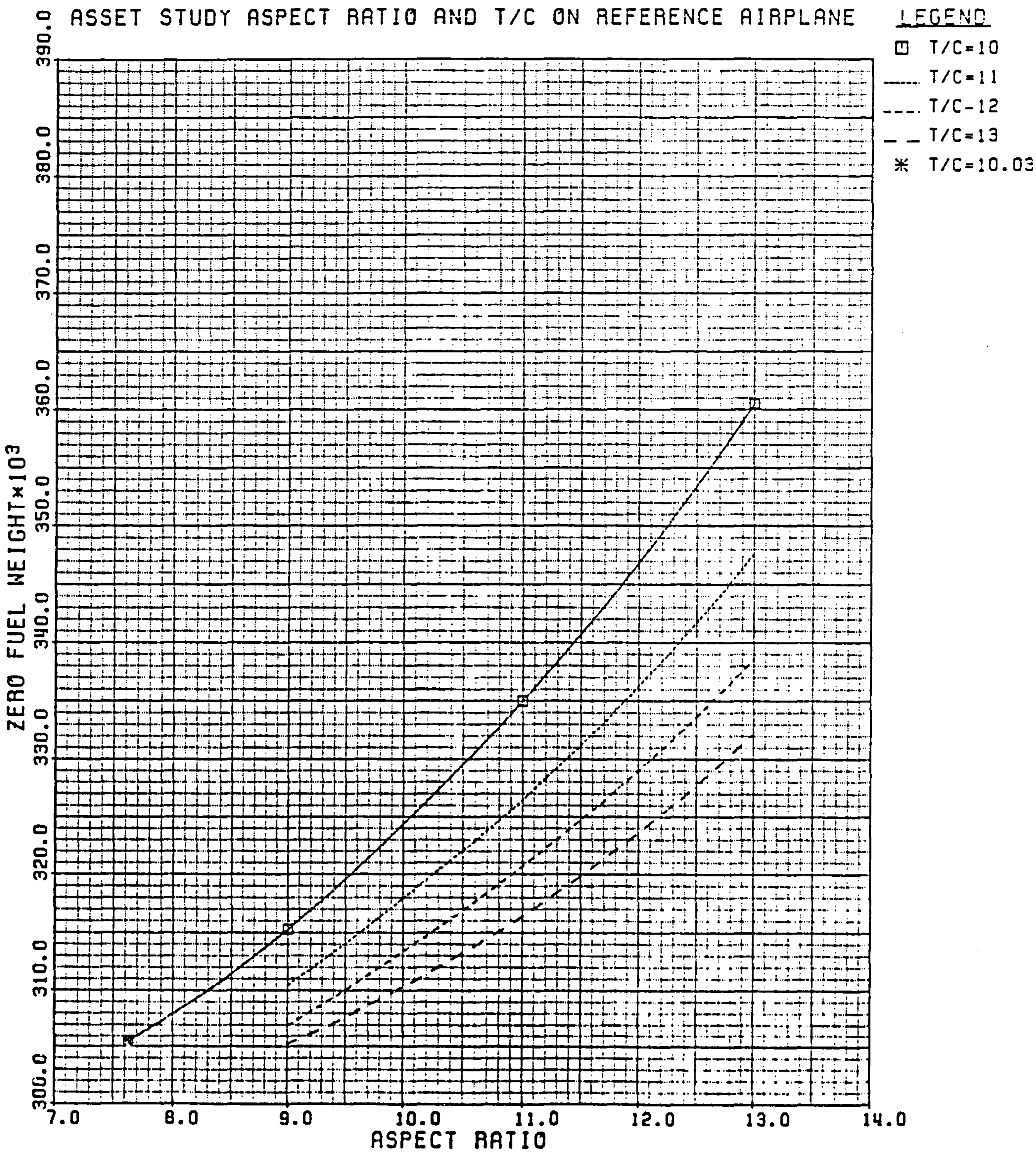


FIGURE 4.1-5 Zero fuel weight (lb) vs aspect ratio for std weight equation

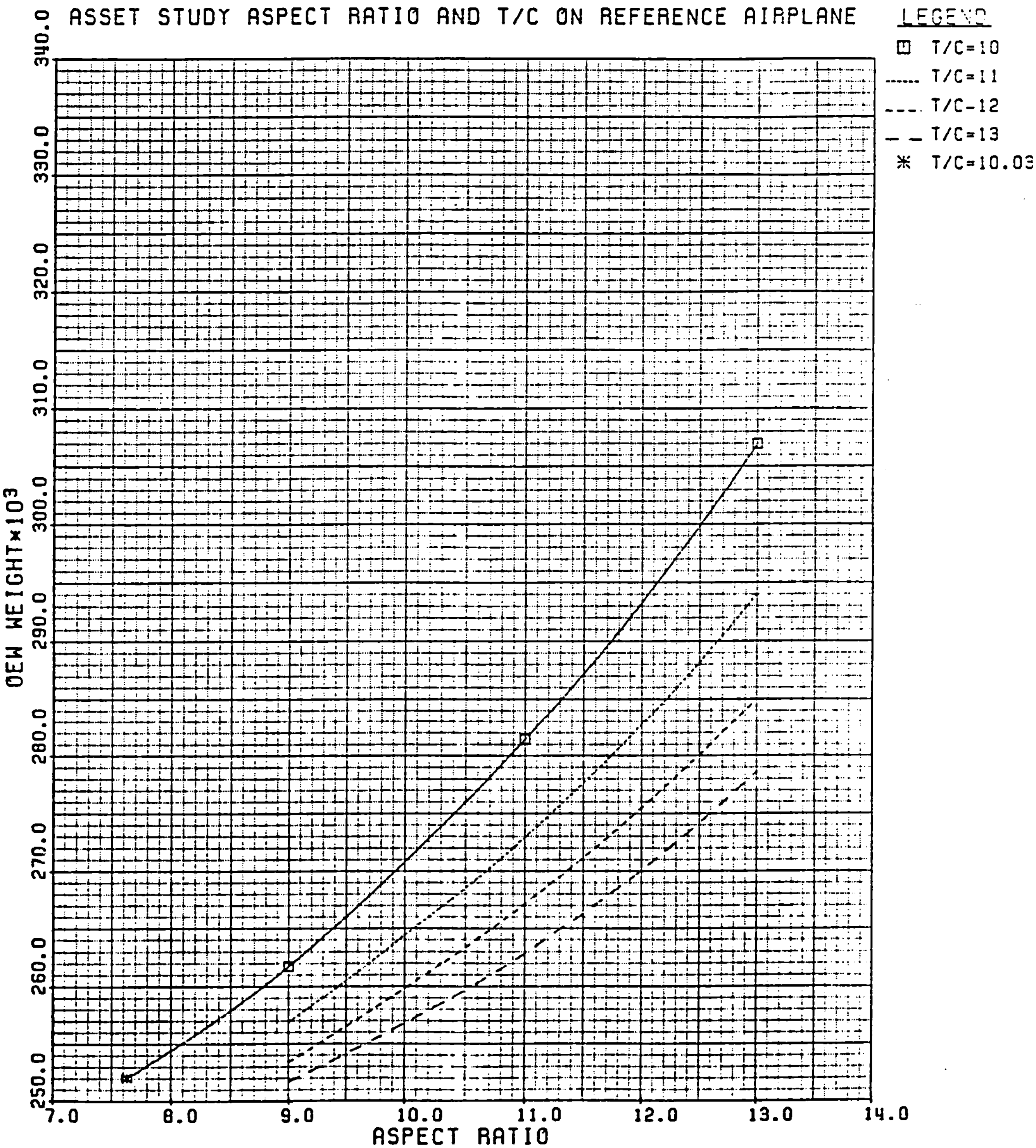


FIGURE 4.1-6 OEW weight vs aspect ratio for std weight equation

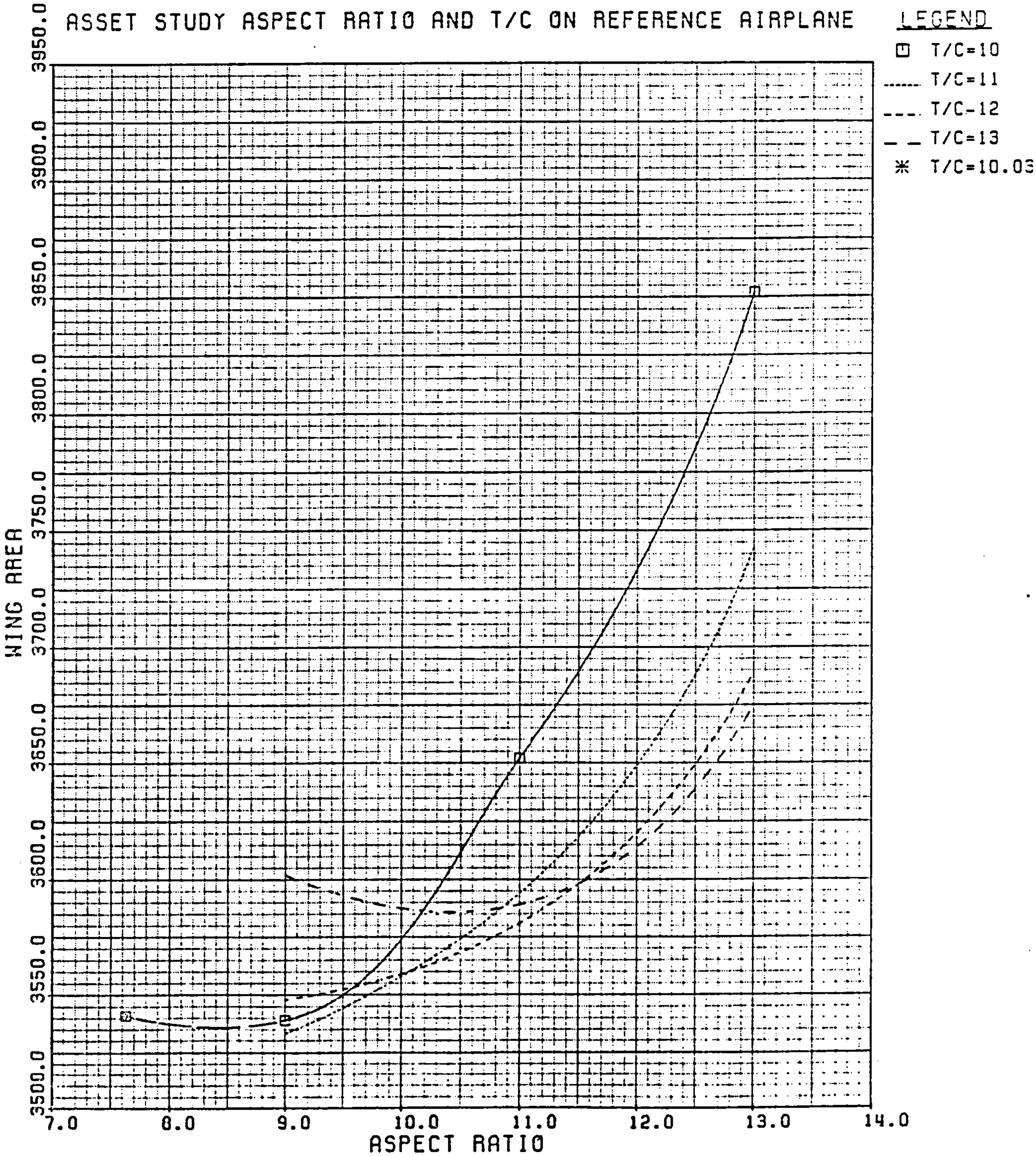


FIGURE 4.1-7 Wing area (ft**2) vs aspect ratio for std weight equation

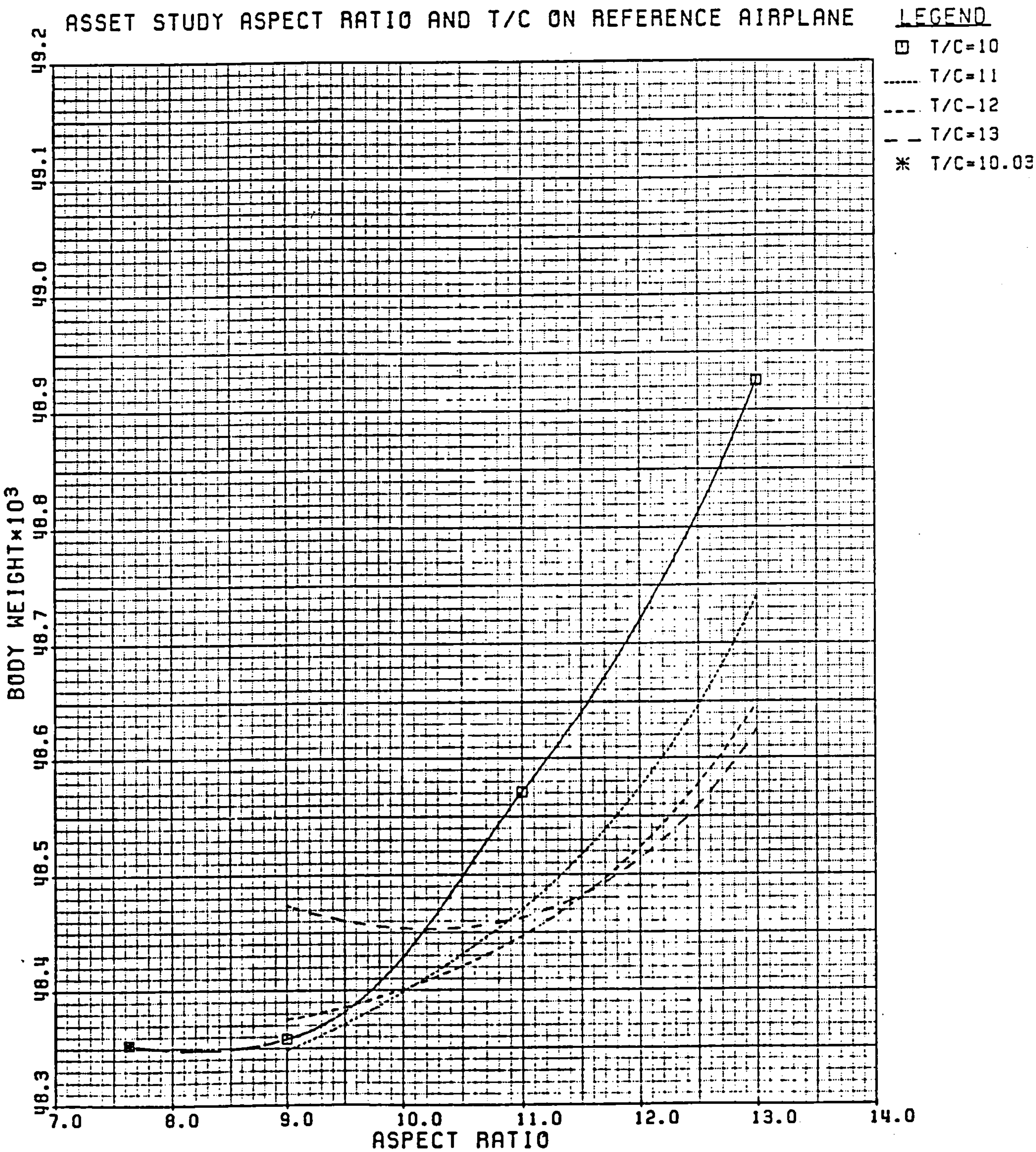


FIGURE 4.1-8 Body weight (lb) vs aspect ratio for std weight equation

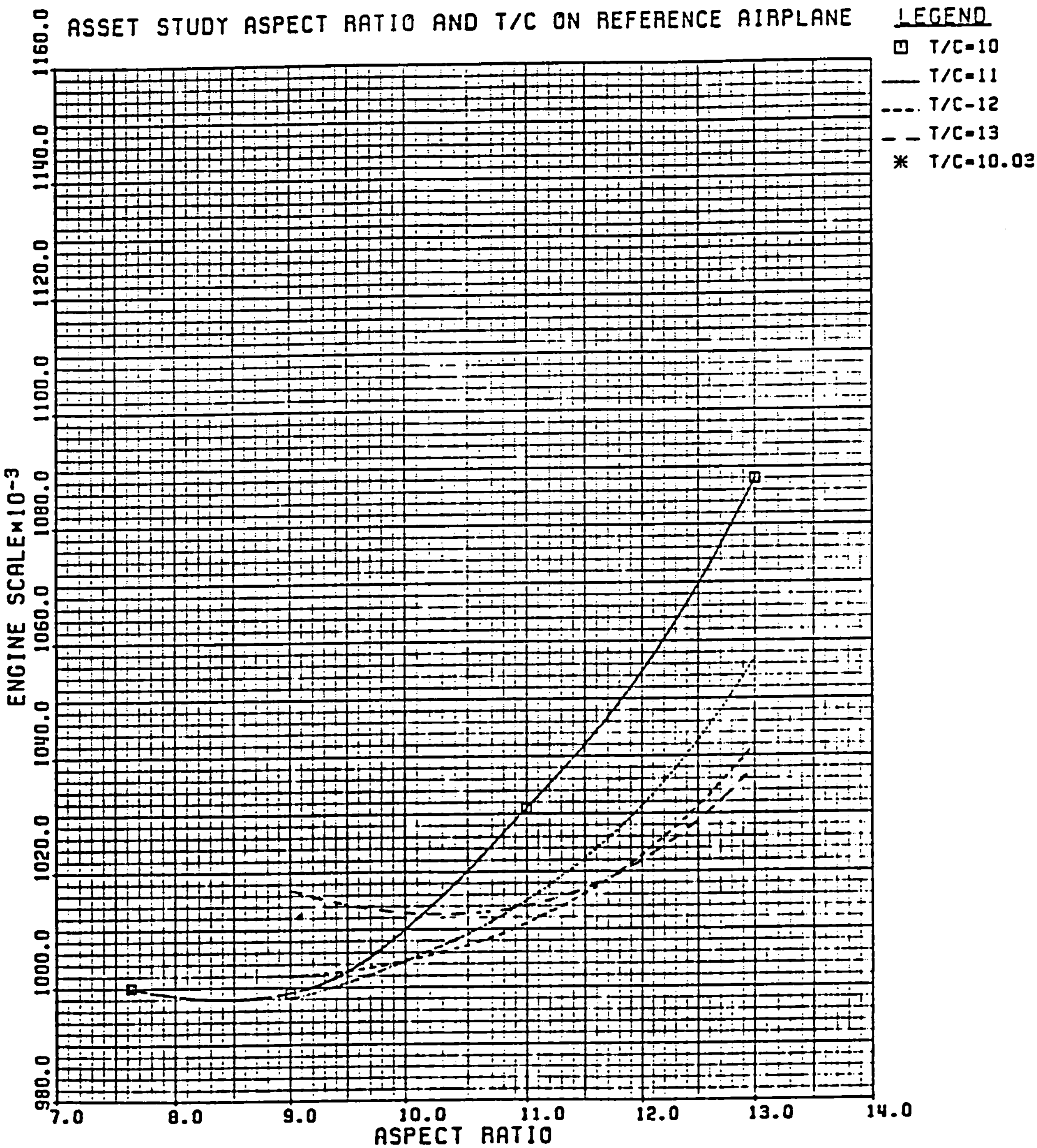


FIGURE 4.1-9   Engine scale vs aspect ratio for std weight equation

SUMMARY ID NO. 1 A S S E T   P A R A M E T R I C   A N A L Y S I S MARCH 13 1984

AIRCRAFT MODEL --L-1011-3
I.O.C. DATE --1980
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 140000
SLS SCALE 1.0 = 50000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 35.00 DEG
WING TAPER RATIO = 0.259

1 W/S	140.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.286	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.03	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	35.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 MPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4749	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	524100	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	181763	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	280787	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	342337	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	1.000	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	50000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3541.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	206.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	1282.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	550.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	9.95	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	7.15	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	164.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.080	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	2.819	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	62.03	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	21.713	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	4.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.395	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - O/O	-27.07	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	31000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	153277	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
34 TAKEOFF DST(1)	8458	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 CLIMB GRAD(1)	0.1400	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
36 TAKEOFF DST(2)	8409	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(2)	0.0609	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 CTOL LNDG D(1)	7245	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 AP SPEED-KT(1)	155.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

TABLE 4.1-3 ASSET design parameters for AR12 35 Degree Sweep

M I S S I O N   S U M M A R Y

DASH 500 L1011-3,238 PAX, M=.83,RANGE=OUTPUT,INTERNATIONAL

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF															
POWER 1	0.	0.0	524100.	0.	0.	0.	0.	0.0	0.0	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	524100.	933.	933.	0.	0.	1.0	1.0	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.378	523167.	2965.	3898.	18.	18.	4.1	5.1	0.	140201.	0.	0.697	19.40	0.558
ACCEL	10000.	0.456	520202.	881.	4779.	8.	26.	1.3	6.4	0.	140201.	0.	0.491	20.31	0.608
CLIMB	10000.	0.638	519320.	6976.	11755.	101.	128.	12.8	19.3	0.	140201.	0.	0.380	18.85	0.680
CRUISE	31000.	0.830	512345.	0.	11755.	0.	128.	0.0	19.3	0.	-140101.	0.	0.499	20.13	0.666
ACCEL	31000.	0.830	512345.	0.	11755.	0.	128.	0.0	19.3	0.	140201.	0.	0.499	20.13	0.680
ACCEL	31000.	0.830	512345.	0.	11755.	0.	128.	0.0	19.3	0.	140201.	0.	0.499	20.13	0.680
CRUISE	31000.	0.830	512344.	10857.	22613.	314.	442.	38.7	58.0	0.	-140101.	0.	0.493	20.09	0.667
CLIMB	31000.	0.830	501487.	1446.	24059.	29.	471.	3.6	61.6	0.	140201.	0.	0.540	19.97	0.677
CRUISE	35000.	0.830	500041.	86055.	110113.	2755.	3226.	345.4	407.0	0.	-140101.	0.	0.535	19.85	0.651
CLIMB	35000.	0.830	413987.	1273.	111387.	30.	3256.	3.8	410.8	0.	140201.	0.	0.541	19.71	0.676
CRUISE	39000.	0.830	412713.	36927.	148313.	1346.	4602.	169.6	580.4	0.	-140101.	0.	0.559	19.56	0.649
DESCENT	39000.	0.830	375786.	1237.	149550.	101.	4703.	12.9	593.3	0.	140301.	0.	0.346	16.99	-18.799
DECEL	10000.	0.638	374550.	186.	149736.	11.	4714.	1.9	595.2	0.	140301.	0.	0.365	18.30	-5.214
DESCENT	10000.	0.450	374364.	839.	150575.	35.	4749.	7.8	603.0	0.	140301.	0.	0.500	20.66	3.365
CRUISE	39000.	0.830	373524.	0.	150575.	0.	4749.	0.0	603.0	0.	-140101.	0.	0.530	19.68	0.654
LOITER	1500.	0.274	373524.	746.	151321.	0.	4749.	3.0	606.0	0.	-140101.	0.	1.000	15.51	0.620
CRUISE	1500.	0.378	372778.	502.	151823.	0.	4749.	2.0	608.0	0.	-140101.	0.	0.525	20.83	0.845
RESET	0.	0.0	372276.	0.	151823.	-4749.	0.	0.0	608.0	0.	0.	0.	0.0	0.0	0.0
CRUISE	39000.	0.830	372276.	12374.	164197.	0.	0.	60.8	668.8	0.	-140101.	0.	0.520	19.66	0.656

TABLE 4.1-4   ASSET mission summary for AR12 35 Degree Sweep

TAKEOFF POWER 1	0.	0.0	359902.	0.	164197.	0.	0.	0.0	668.8	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	359902.	933.	165130.	0.	0.	1.0	669.8	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.378	358969.	1713.	166843.	11.	11.	2.4	672.2	0.	140201.	0.	0.473	20.36	0.558
ACCEL	10000.	0.456	357256.	246.	167090.	2.	13.	0.4	672.6	0.	140201.	0.	0.396	19.12	0.585
CLIMB	10000.	0.547	357009.	3521.	170611.	46.	58.	6.7	679.3	0.	140201.	0.	0.337	17.75	0.630
CRUISE	30000.	0.760	353488.	1207.	171818.	41.	100.	5.5	684.8	0.	-140101.	0.	0.391	18.83	0.698
DESCENT	30000.	0.750	352281.	813.	172631.	69.	168.	10.2	695.0	0.	140301.	0.	0.333	17.62	13.445
DECEL	10000.	0.547	351468.	100.	172730.	5.	174.	1.0	696.0	0.	140301.	0.	0.393	19.05	5.983
DESCENT	10000.	0.450	351369.	830.	173560.	35.	209.	7.7	703.7	0.	140301.	0.	0.469	20.31	3.364
CRUISE	30000.	0.760	350539.	251.	173811.	9.	217.	1.2	704.9	0.	-140101.	0.	0.389	18.76	0.698
CRUISE	1500.	0.378	350287.	7897.	181708.	0.	217.	32.0	736.9	0.	-140101.	0.	0.488	20.50	0.876

WTO = 524100.0 FUEL A=181763.1 FUEL R=181708.1

TABLE 4.1-4 (cont.) ASSET mission summary for AR12 35 Degree Sweep

DASH 500 L1011-3,238 PAX, M=.83,RANGE=OUTPUT,INTERNATIONAL

T/C=10.03 AR=12.00 W/S=148.01 T/W=0.286

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(524100.)		
FUEL AVAILABLE	181763.	FUEL	34.68
EXTERNAL	0.		
INTERNAL	181763.		
ZERO FUEL WEIGHT	342337.		
PAYLOAD	53550.	PAYLOAD	10.22
PASSENGERS	40460.		
BAGGAGE	13090.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	288787.		
OPERATIONAL ITEMS	8599.	OPERATIONAL ITEMS	3.29
STANDARD ITEMS	8638.		
EMPTY WEIGHT	271550.		
STRUCTURE	174795.	STRUCTURE	33.35
WING	85578.		
ROTOR	0.		
TAIL	9042.		
BODY	48587.		
ALIGHTING GEAR	21903.		
ENGINE SECTION AND NACELLE	9684.		
PROPULSION	40459.	PROPULSION	7.72
CRUISE ENGINES	32693.		
LIFT ENGINES	0.		
THRUST REVERSER	4923.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	214.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2091.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	56297.		
FLIGHT CONTROLS	6254.		
AUXILIARY POWER PLANT	1202.		
INSTRUMENTS	998.		
HYDRAULIC AND PNEUMATIC	2758.		
ELECTRICAL	5797.		
AVIONICS	2827.	SYSTEMS	10.74
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	30267.		
AIR CONDITIONING	5811.		
ANTI-ICING	383.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

TABLE 4.1-5 ASSET weight summary for AR12 35 Degree Sweep

4.2 ASSET Run At 25 Degrees Of Sweep And Aspect Ratio 12

The second perturbation from the baseline design involved the changing of the wing sweep angle from 35 degrees to 25 degrees. All other design variables, including the aspect ratio 12, were held constant from the aspect ratio 12 and 35 degrees of sweep design.

Since the sweep angle was being reduced and the airfoil technology was constraint to be constant, ASSET generated data to study the drag rise problem at 25 degrees of sweep. Figure 4.2-1 shows block fuel as a function of mach number and range. The airplane at this sweep would have a cruise mach number of .76 to .78 for maximum fuel efficiency.

Tables 4.2-1 through 4.2-3 are selected ASSET computer run output from the mach number and block fuel study. The tables summarized the design parameters, mission, and weight statement for mach .76. The weight analysis (fuel, cg, etc) for the aeroelastic design was based on Mach .76. The lower mach number was selected to compensate for increases in t/c from 10.03. Work required to form a new loads design envelope however did not justify the expected small differences in wing design results. Therefore, the loads work was based on a cruise mach of .83.

4.3 Aeroelastic Wing Design Configuration Definition.

PADS is a loosely coupled design tool for vehicle section process. In this study, two prime design variables were judge to be important to validating the existing parametric weight equation for high aspect ratio wing designs; namely aspect ratio and sweep. At the beginning of the study, it was anticipated that the trends would follow the weight equation and that only a possible change in the magnitude of some coefficients would be required. The plan therefore included PADS design with t/c and wing area changes relative to the baseline. However, when the results were opposite to expectations, the t/c and wing area studies were cancelled and those resources were allocated to the investigation of the aspect ratio and sweep PADS designs results.

The two PADS designs selected for support of the ASSET weight equation was as follows:

1) Baseline with Aspect Ratio 12

2) Baseline with Aspect Ratio 12 and Sweep of 25 degrees

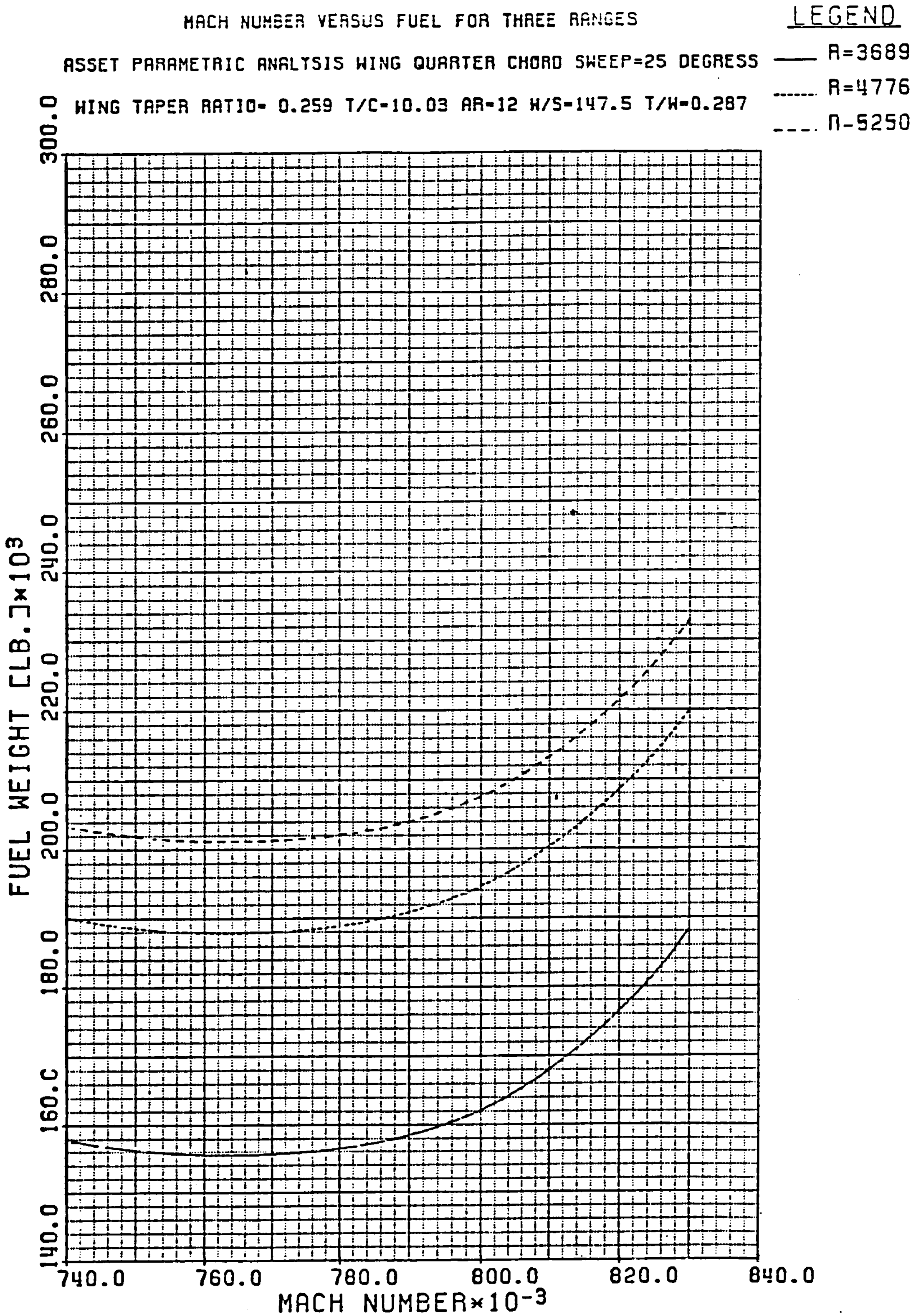


FIGURE 4.2-1 Block fuel versus Mach number for AR12 25 Degree Sweep aircraft

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S MARCH 13 1984

AIRCRAFT MODEL --L-1011-3
I.O.C. DATE --1980
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 140000
SLS SCALE 1.0 = 50000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 25.00 DEG
WING TAPER RATIO = 0.259

1 W/S	146.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.280	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.03	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 BPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4786	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	520273	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	188089	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	270634	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	332184	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	1.000	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	50000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3541.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	206.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	1282.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	550.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	9.95	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	7.15	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	164.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.044	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	2.723	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	60.55	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	21.215	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	4.428	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.449	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - O/O	-29.25	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	31000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	159630	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
34 TAKEOFF DST(1)	7775	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 CLIMB GRAD(1)	0.1420	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
36 TAKEOFF DST(2)	7771	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(2)	0.0600	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 CTOL LNDG D(1)	7194	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 AP SPEED-KT(1)	154.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

TABLE 4.2-1 ASSET design parameters for AR12 25 Degree Sweep

M I S S I O N   S U M M A R Y

DASH 500 L1011-3,238 PAX, M=.76,RANGE=OUTPUT,INTERNATIONAL

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	520273.	0.	0.	0.	0.	0.0	0.0	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	520273.	933.	933.	0.	0.	1.0	1.0	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.378	519340.	2886.	3819.	18.	18.	4.0	5.0	0.	140201.	0.	0.691	19.99	0.558
ACCEL	10000.	0.456	516454.	868.	4687.	8.	26.	1.3	6.3	0.	140201.	0.	0.487	20.42	0.608
CLIMB	10000.	0.638	515586.	6255.	10942.	88.	113.	11.7	18.0	0.	140201.	0.	0.422	19.17	0.660
CRUISE	31000.	0.760	509331.	0.	10942.	0.	113.	0.0	18.0	0.	-140101.	0.	0.591	19.78	0.634
ACCEL	31000.	0.760	509331.	0.	10942.	0.	113.	0.0	18.0	0.	140201.	0.	0.591	19.78	0.648
ACCEL	31000.	0.760	509331.	0.	10942.	0.	113.	0.0	18.0	0.	140201.	0.	0.591	19.78	0.648
CRUISE	31000.	0.760	509331.	66997.	77939.	1950.	2064.	262.3	280.4	0.	-140101.	0.	0.552	19.99	0.645
CLIMB	31000.	0.760	442333.	1057.	78997.	20.	2084.	2.8	283.1	0.	140201.	0.	0.568	19.78	0.647
CRUISE	35000.	0.760	441276.	75740.	154736.	2560.	4644.	350.4	633.6	0.	-140101.	0.	0.563	19.68	0.635
CLIMB	35000.	0.760	365536.	57.	154793.	1.	4645.	0.2	633.7	0.	140201.	0.	0.524	19.79	0.646
CRUISE	39000.	0.760	365479.	0.	154793.	0.	4645.	0.0	633.7	0.	-140101.	0.	0.619	19.08	0.622
DESCENT	39000.	0.760	365479.	1144.	155937.	96.	4740.	12.9	646.7	0.	140301.	0.	0.388	17.35	-9.331
DECEL	10000.	0.638	364335.	184.	156121.	11.	4751.	1.9	648.5	0.	140301.	0.	0.356	18.09	-5.124
DESCENT	10000.	0.450	364152.	841.	156961.	35.	4786.	7.8	656.3	0.	140301.	0.	0.486	20.62	3.365
CRUISE	39000.	0.760	363311.	0.	156961.	0.	4786.	0.0	656.3	0.	-140101.	0.	0.615	19.11	0.623
LOITER	1500.	0.270	363311.	718.	157679.	0.	4786.	3.0	659.3	0.	-140101.	0.	1.000	16.25	0.643
CRUISE	1500.	0.378	362593.	497.	158176.	0.	4786.	2.0	661.3	0.	-140101.	0.	0.511	20.85	0.861
RESET	0.	0.0	362096.	0.	158176.	-4786.	0.	0.0	661.3	0.	0.	0.	0.0	0.0	0.0
CRUISE	39000.	0.760	362096.	12774.	170950.	0.	0.	66.1	727.5	0.	-140101.	0.	0.602	19.23	0.627

TABLE 4.2-2 ASSET mission summary for AR12 25 Degree Sweep

TAKEOFF															
POWER 1	0.	0.0	349322.	0.	170950.	0.	0.	0.0	727.5	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	349322.	933.	171883.	0.	0.	1.0	728.5	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.378	348389.	1650.	173534.	10.	10.	2.3	730.7	0.	140201.	0.	0.458	20.26	0.558
ACCEL	10000.	0.456	346739.	238.	173771.	2.	12.	0.4	731.1	0.	140201.	0.	0.384	18.90	0.585
CLIMB	10000.	0.547	346501.	3411.	177183.	44.	57.	6.5	737.6	0.	140201.	0.	0.327	17.28	0.631
CRUISE	30000.	0.725	343090.	1249.	178432.	43.	100.	6.1	743.7	0.	-140101.	0.	0.420	19.13	0.688
DESCENT	30000.	0.750	341840.	794.	179226.	67.	167.	9.9	753.6	0.	140301.	0.	0.323	17.18	13.445
DECEL	10000.	0.547	341046.	99.	179324.	5.	172.	1.0	754.7	0.	140301.	0.	0.382	18.83	5.976
DESCENT	10000.	0.450	340948.	828.	180152.	35.	207.	7.7	762.3	0.	140301.	0.	0.455	20.21	3.364
CRUISE	30000.	0.720	340119.	195.	180347.	7.	214.	1.0	763.3	0.	-140101.	0.	0.420	19.13	0.688
CRUISE	1500.	0.378	339925.	7823.	188170.	0.	214.	32.0	795.3	0.	-140101.	0.	0.473	20.43	0.892

WTO = 520273.0 FUEL A=188088.8 FUEL R=188170.1

TABLE 4.2-2 (cont.) ASSET mission summary for AR12 25 Degree Sweep

DASH 500 L1011-3,238 PAX, M=.76,RANGE=OUTPUT,INTERNATIONAL

T/C=10.03 AR=12.00 W/S=146.93 T/W=0.288

WEIGHT STATEMENT

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(520273.)		
FUEL AVAILABLE	188089.	FUEL	36.15
EXTERNAL	0.		
INTERNAL	188089.		
ZERO FUEL WEIGHT	332184.		
PAYLOAD	53550.	PAYLOAD	10.29
PASSENGERS	40460.		
BAGGAGE	13090.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	278634.		
OPERATIONAL ITEMS	8599.	OPERATIONAL ITEMS	3.31
STANDARD ITEMS	8647.		
EMPTY WEIGHT	261389.		
STRUCTURE	164652.	STRUCTURE	31.65
WING	75556.		
ROTOR	0.		
TAIL	9042.		
BODY	48549.		
ALIGHTING GEAR	21821.		
ENGINE SECTION AND NACELLE	9684.		
PROPULSION	40464.	PROPULSION	7.78
CRUISE ENGINES	32693.		
LIFT ENGINES	0.		
THRUST REVERSER	4923.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	214.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2097.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	56273.		
FLIGHT CONTROLS	6248.		
AUXILIARY POWER PLANT	1202.		
INSTRUMENTS	998.		
HYDRAULIC AND PNEUMATIC	2740.		
ELECTRICAL	5797.		
AVIONICS	2827.	SYSTEMS	10.82
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	30267.		
AIR CONDITIONING	5811.		
ANTI-ICING	383.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

TABLE 4.2-3 ASSET weight summary for AR12 25 Degree Sweep

These two PADS designs will be the subject of the next section.

5.0 DEFINITION OF ASPECT RATIO 12 DESIGNS

The PADS system was used to formulate aeroelastic analysis models for aspect ratio 12 wing designs (AR12) of 35 and 25 degree sweeps. With the incorporation of the new capabilities for the finite element generation, the grid load distribution scheme, and the fuel tank loading procedures, PADS built the necessary database and processed the high aspect ratio designs through final sizing. Figure 2.0-2 shows the PADS design process overview.

Aspect Ratio 12, 35 Degree Sweep Design:

A NASTRAN model was created by the finite element model generator using the generic FEM model which was formed for the Baseline reference design. New planform and airfoil definition datasets were created for the aspect ratio 12.0 wing. Figure 5.0-1 shows the contrast between wing planforms for aspect ratios 7.64 and 12. The 25 percent chord location at the mean aerodynamic chord was constrained to the same fuselage station to keep center of lift for the aircraft approximately the same. This constraint forced the AR12's wing to be 22 inches forward of the Baseline wing leading edge at the centerline of the aircraft. Fuselage representation for the AR12 , 35 Degree Sweep design is the same as the Baseline.

Geometric parameters, including thickness/chord ratio, wing area, taper ratio, and 25 percent chord sweep angle between the two wings, were kept the same. Figures 5.0-2 and 5.0-3 show comparisons of cuts through the chord at the same percentage span for both wings. The airfoil with the smaller chord length is the AR12, 35 Degree Sweep airfoil. Figure 5.0-2 depicts the root rib airfoil and figure 5.0-3 depicts the midspan airfoil.

The NASTRAN structural representation had 3741 degrees of freedom (DOF). The load reference or SICs locations numbered 228. The weight distribution module generated 500 panel weights and the Static Loads grid was defined with 289 load points. Weight and maneuver conditions were chosen to be on a comparable basis as the Baseline design for covering the extremes of the design region. Twenty-five static loads conditions were formed for the initial wing panel sizing and stiffness generation. These 25 conditions include various weight configurations, velocities, and altitudes for maneuvers, as well as ground handling conditions as discussed in section 3.4.2. The initial loads were computed for a rigid airplane and applied to the structural model for computation of the first sizing. Subsequent loads included the effects of flexibility.

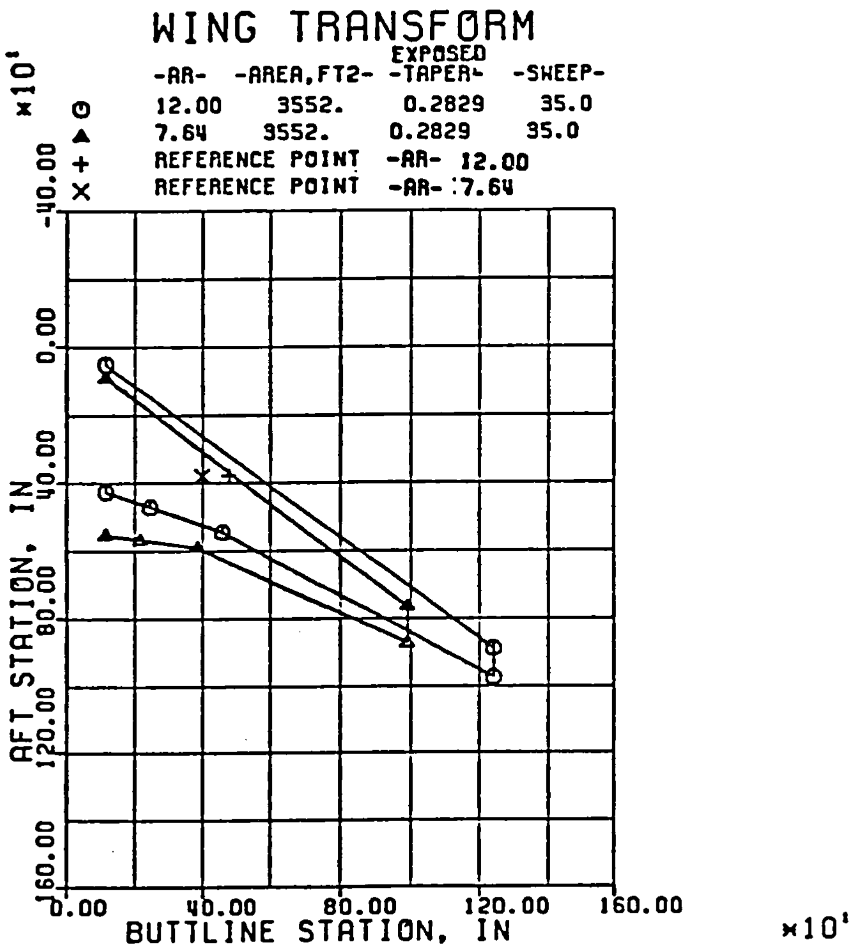


FIGURE 5.0-1 Baseline Versus AR 12 35 Degree Sweep Wing Geometry

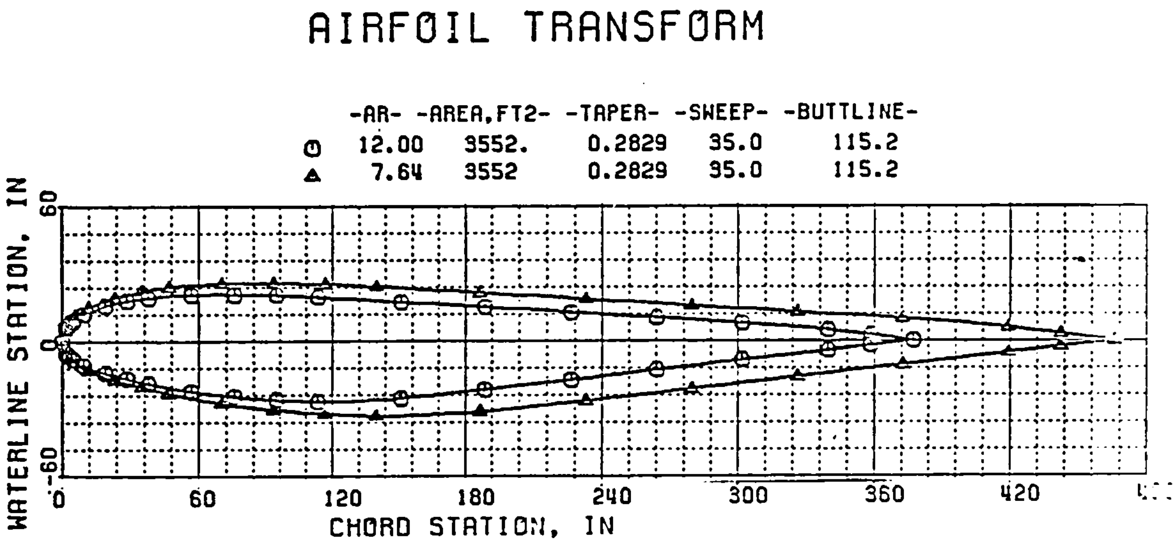


FIGURE 5.0-2 Root Airfoil Comparison

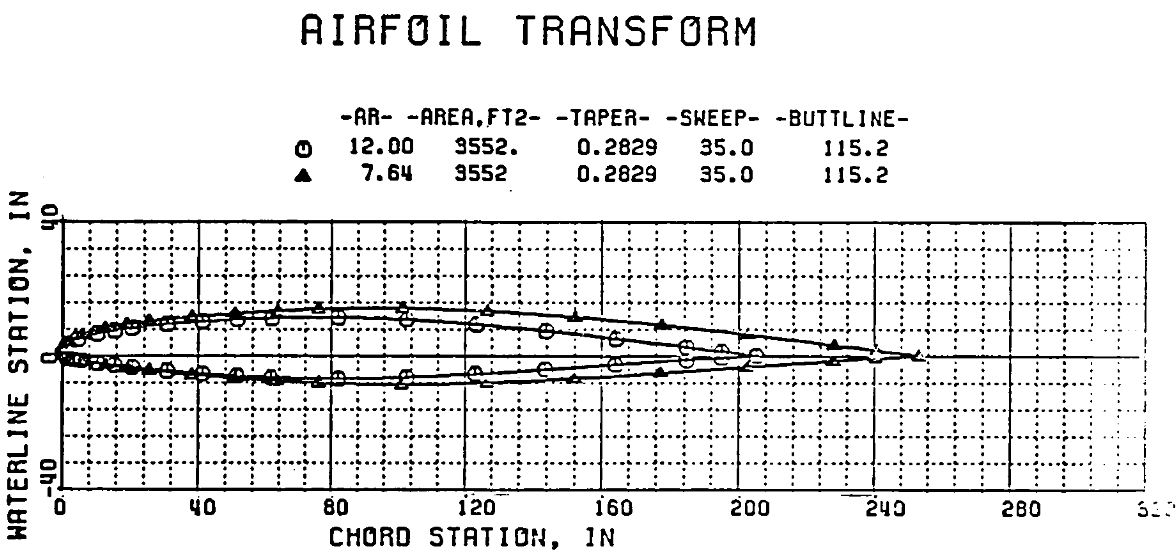


FIGURE 5.0-3 Midspan Airfoil Comparison

The sizing procedure for the AR12, 35 Degree Sweep design incorporated the new methodology for computing the design element internal loads for applied unit external loads at the SIC points. The initial internal loads were formed for an arbitrary starting sizing. Internal loads, due to the 25 constructed load conditions, were formed by multiplying the load condition matrix and the internal load matrix for unit external loads together.

Sizing for the internal loads was then accomplished with the PADS data base approach for selection of optimal panel dimensions. Stress margins of safety as described in Appendix B were supplied for all AR12, 35 Degree Sweep sizing runs. Sizing runs for the AR12, 35 Degree Sweep aircraft were run for the standard ACS gain of -11.33 degrees per g along with variations. The following upper and lower surface panel weights for half an airplane were computed as direct NASTRAN output:

NASTRAN WEIGHT CALCULATION FOR PADS SIZING ON AR12 SWEEP 35.

External Loads	Properties to be Updated	ACS Gain degrees/g	Upper Cover Weight	Lower Cover Weight
rigid	starting	-11.33	7511.42	12358.22
1st flex	rigid	-11.33	6512.45	10745.62
2nd flex	1st flex	-11.33	6335.90	10695.23
15 deg/g	2nd flex	-15.00	6211.04	10748.68
2 g no ACS	2nd flex	0.00	5154.62	8028.01
2.5 g no ACS	2nd flex	0.00	6725.68	11876.96
20 deg/g	2nd flex	-20.00	6181.05	10655.39
30 deg/g	2nd flex	-30.00	6319.27	10984.63
2nd flex & gust loads	2nd flex	-11.33	6418.52	10881.62

The NASTRAN calculated weights (as described in more detail in section 5.1.1.5) get factored to account for effects not represented in the finite element. Load cases, which yield the minimum margins of safety after consideration in the optimization process for effects such as fatigue and fail-safe conditions, are shown in figure 5.0-4 for the 2nd flex (Static load) external load condition. Each symbol corresponds to the load condition yielding the minimum margin for the corresponding panel.

Aspect Ratio 12, 25 Degree Sweep Design:

LEGEND

Letter	Cond#	g's	Mach	Active Controls
A.H	132	+2.5	.82	ON
B.I	411	BRAKE		
C.J.L	122	+2.5	.478	ON
D.K	124	+2.5	.478	OFF
E	133	-1.0	.82	ON
F. M	142	+2.5	.88	ON
G	113	-1.0	.804	ON
N	112	+2.5	.804	ON
O	143	0.0	.88	ON

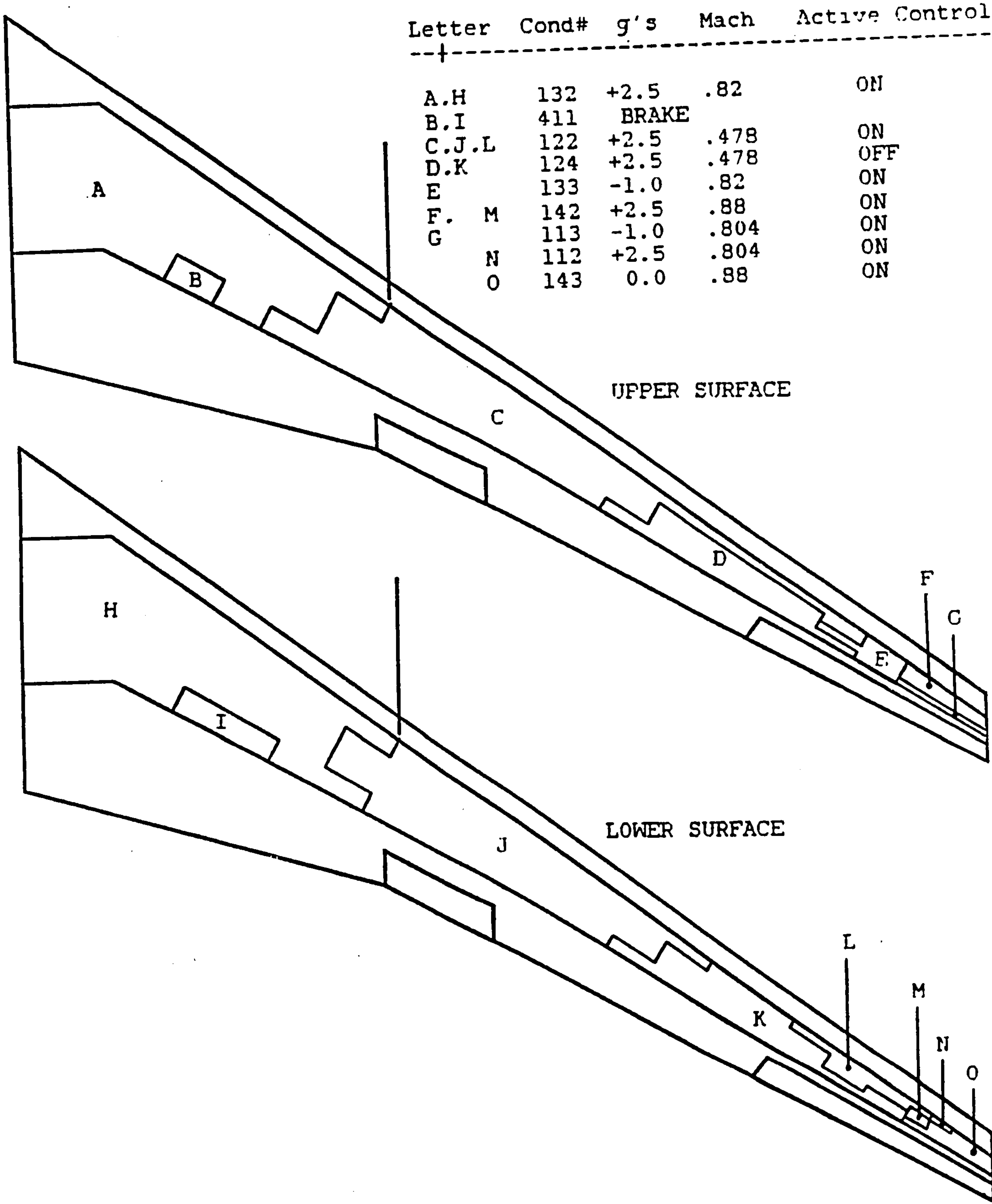


FIGURE 5.0-4 AR12 35 Degree Sweep 2nd Flex Min Margin Loads

A new NASTRAN model was created by the finite element model generator. New planform and airfoil definition datasets were created for the change from an aspect 7.64 wing with a 35 degree sweep to an aspect ratio 12.0 wing with a 25 degree sweep. Figure 5.0-5 shows the contrast between wing planforms. The 25 percent chord location at the mean aerodynamic chord was constrained to the same fuselage station to keep center of lift for the aircraft approximately the same. This constraint forced the AR12's wing to be 63 inches aft of the Baseline wing leading edge at the centerline of the aircraft. Fuselage representation for the AR12 25 Degree Sweep design is the same as the Baseline except for the aft movement of the wing. Geometric parameters, including thickness/chord ratio, wing area, and taper ratio between the two wings, were kept the same.

The NASTRAN structural representation, weight grid, and load point grid were constructed similar in nature as to the AR12, 35 degree design. Weight and maneuver conditions were chosen to be on a comparable basis as the Baseline design for covering the extremes of the design region. Twenty-five static loads conditions were formed for the initial wing panel sizing and stiffness generation. These 25 conditions include various weight configurations, velocities, and altitudes for maneuvers, as well as ground handling conditions as discussed in section 3.4.2.

Sizing for the internal loads was then accomplished similar to the AR12, 35 Degree design. Stress margins of safety as described in Appendix B were supplied for all AR12, 25 Degree Sweep sizing runs. The following upper and lower surface panel weights for half an airplane were computed as direct NASTRAN output:

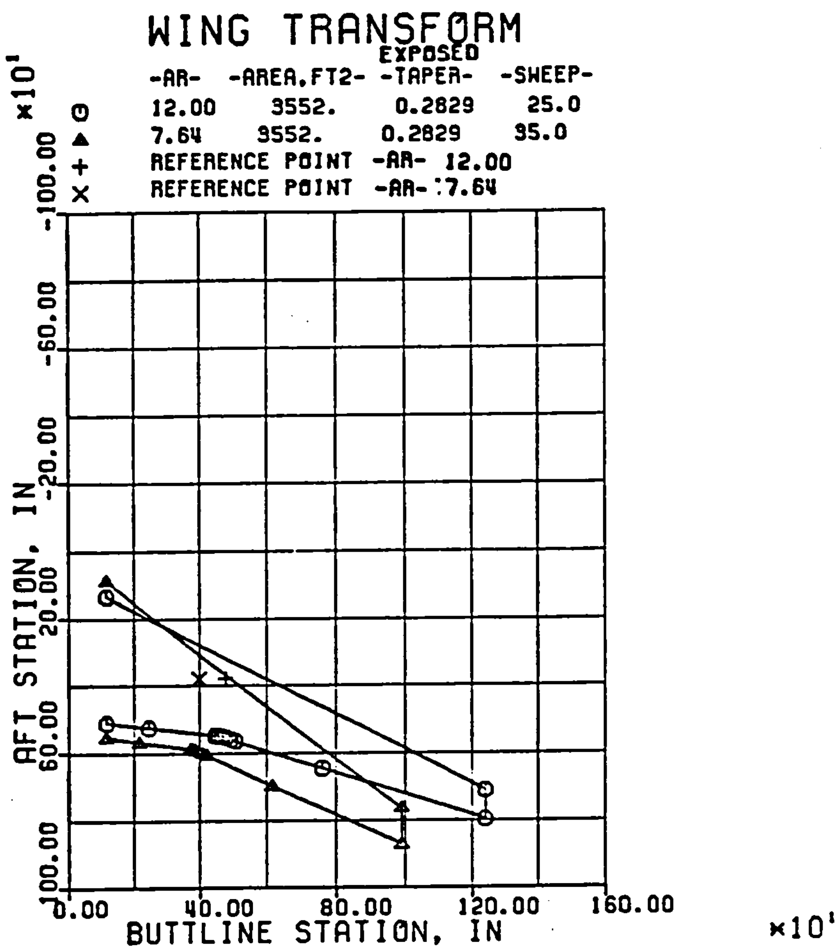


FIGURE 5.0-5 Baseline Versus AR12 25 Degree Sweep Wing Geometry

NASTRAN WEIGHT CALCULATION FOR PADS SIZING ON AR12 SWEEP 25.

External Loads	Properties to be Updated	ACS Gain degrees/g	Upper Cover Weight	Lower Cover Weight
3rd flex	2nd flex	-11.33	6591.19	11642.60
2.5 g no ACS	2nd flex	0.00	7011.35	11902.33
9 deg/g	2nd flex	- 9.00	6648.10	11792.96
18 deg/g	2nd flex	-18.00	6645.31	11776.91

* Note - Initial pass sizing results are not included because panel areas for these sizing passes were adjusted after computation of weight.

The NASTRAN calculated weights are factored to account for overall PADS' sizing differences with production airplane model sizings and to account for non-structural items such as sealants and rivets. Load cases which yield the minimum margins of safety are shown in figure 5.0-6 for the 3rd flex (Static load) external load condition. Each symbol corresponds to a load condition which generated the minimum margin for the corresponding panel.

Flutter Analysis:

A limited flutter analysis was performed for the aspect ratio 12, 35 degree sweep design. The stiffness matrix was based on the second flex loads which also included the effects of dynamic gust loads. The frequency and damping versus velocity plots as shown in figures 5.0-7 and -8 respectively were computed using fifty vibration modes. Figure 5.0-8 shows the flutter speed at 430 KEAS, which is close to the flutter dive velocity (VD) of 418 KEAS. The current FAA proposal on active control design permits flutter deficiencies below 1.2 VD but above VD for the airplane without active controls. This proposal will permit the satisfying of the 1.2 VD flutter free requirement by the application of active control technology. The weight increment necessary to satisfy the 1.2 VD requirement is being investigated as part of a flutter optimization methodology study. With the flutter speed sensitivity to aspect ratio, it is recommended that flutter be an active constraint in any future high aspect ratio wing design study.

LEGEND

Letter	Cond#	g's	Mach	Active Controls
A.D.F. M	132	+2.5	.82	ON
B. G. K. P	142	+2.5	.88	ON
C. L	411	BRAKE		
E. N	124	+2.5	.478	OFF
H.J. Q	113	-1.0	.804	ON
I. O	133	-1.0	.82	ON

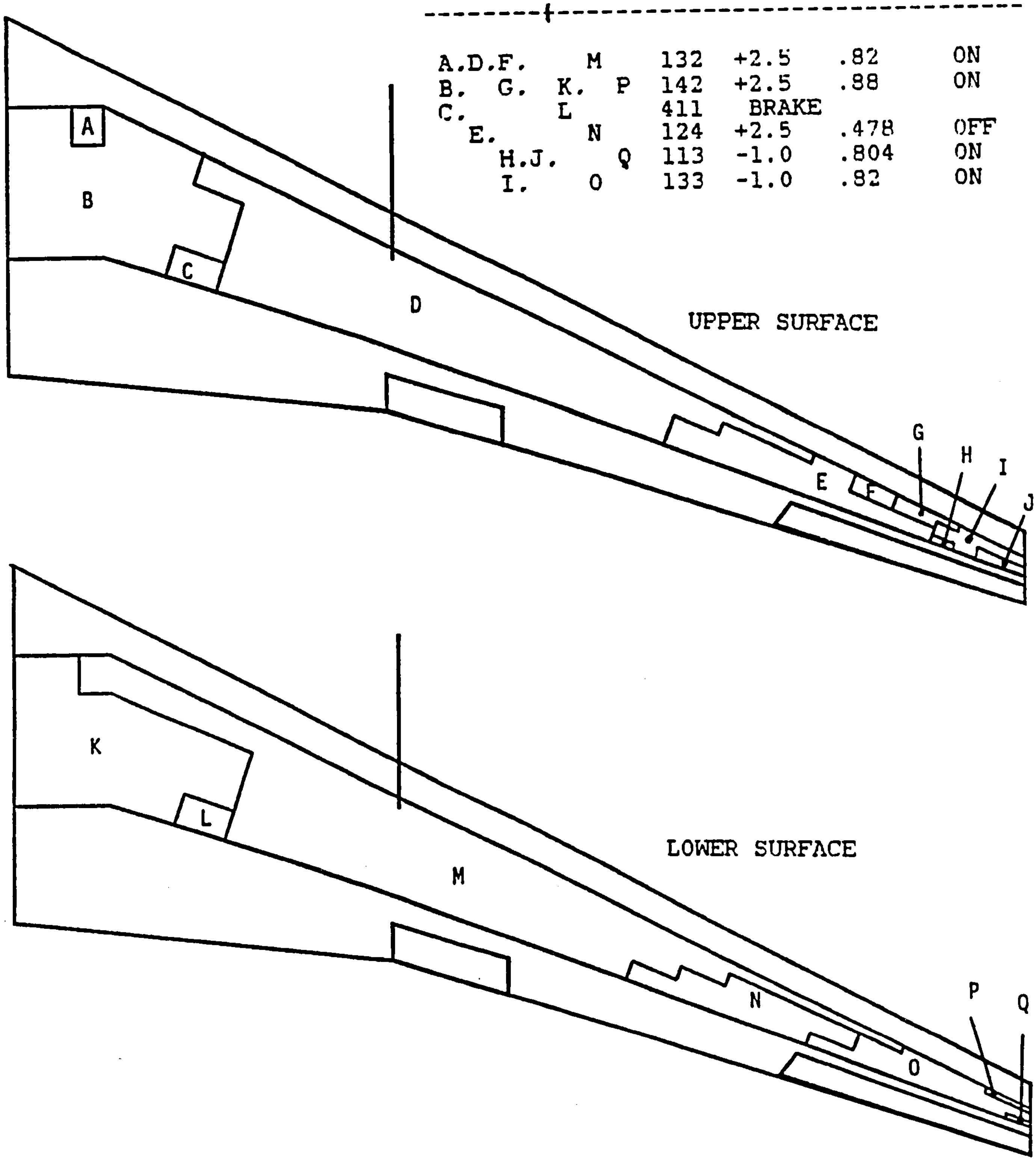


FIGURE 5.0-6 AR12 25 Degree Sweep 3rd Flex Min Margin Loads

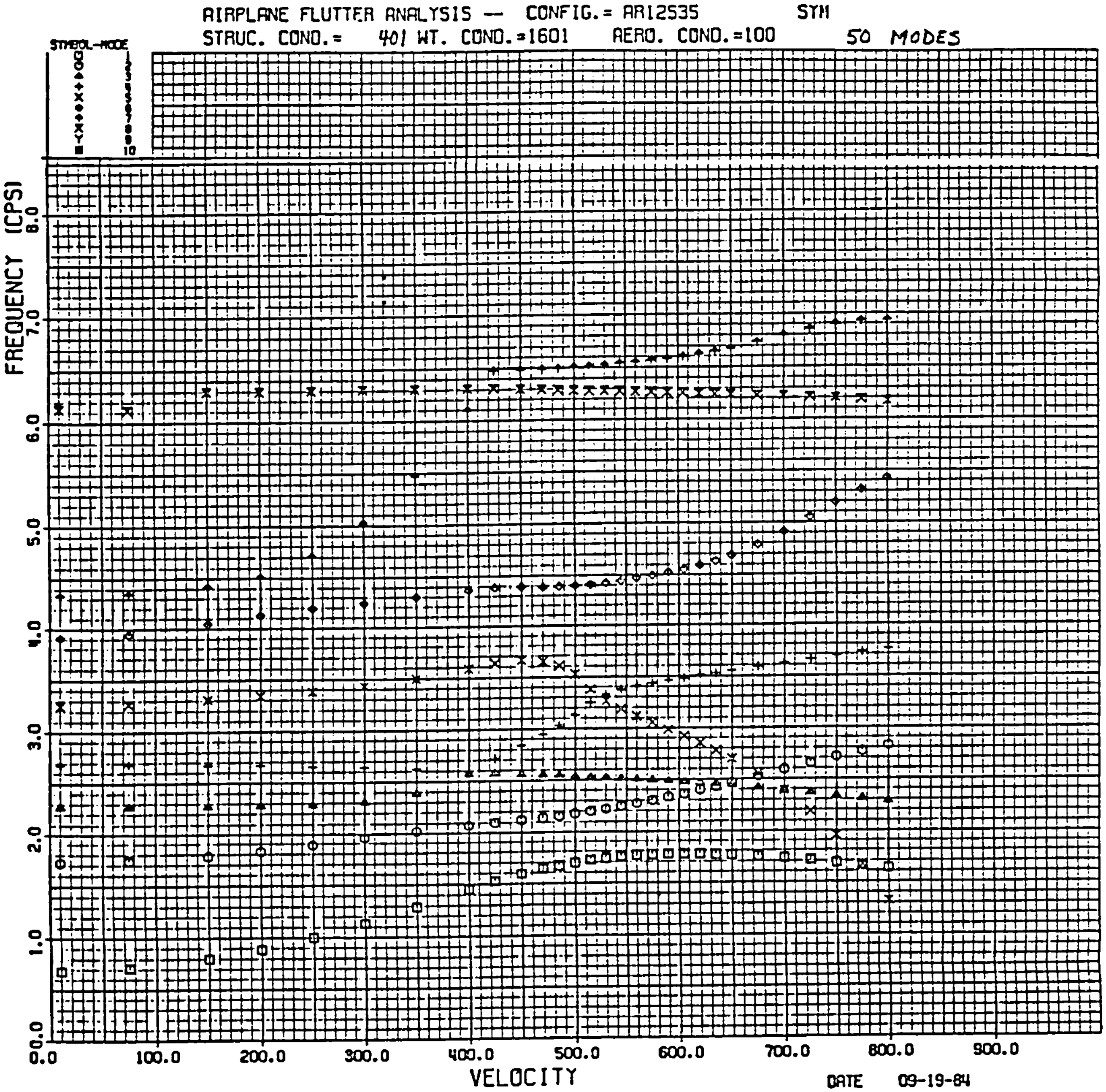


Figure 5.0-7 AR12 35 Deg. Sweep full fuel flutter plot - frequency vs. velocity

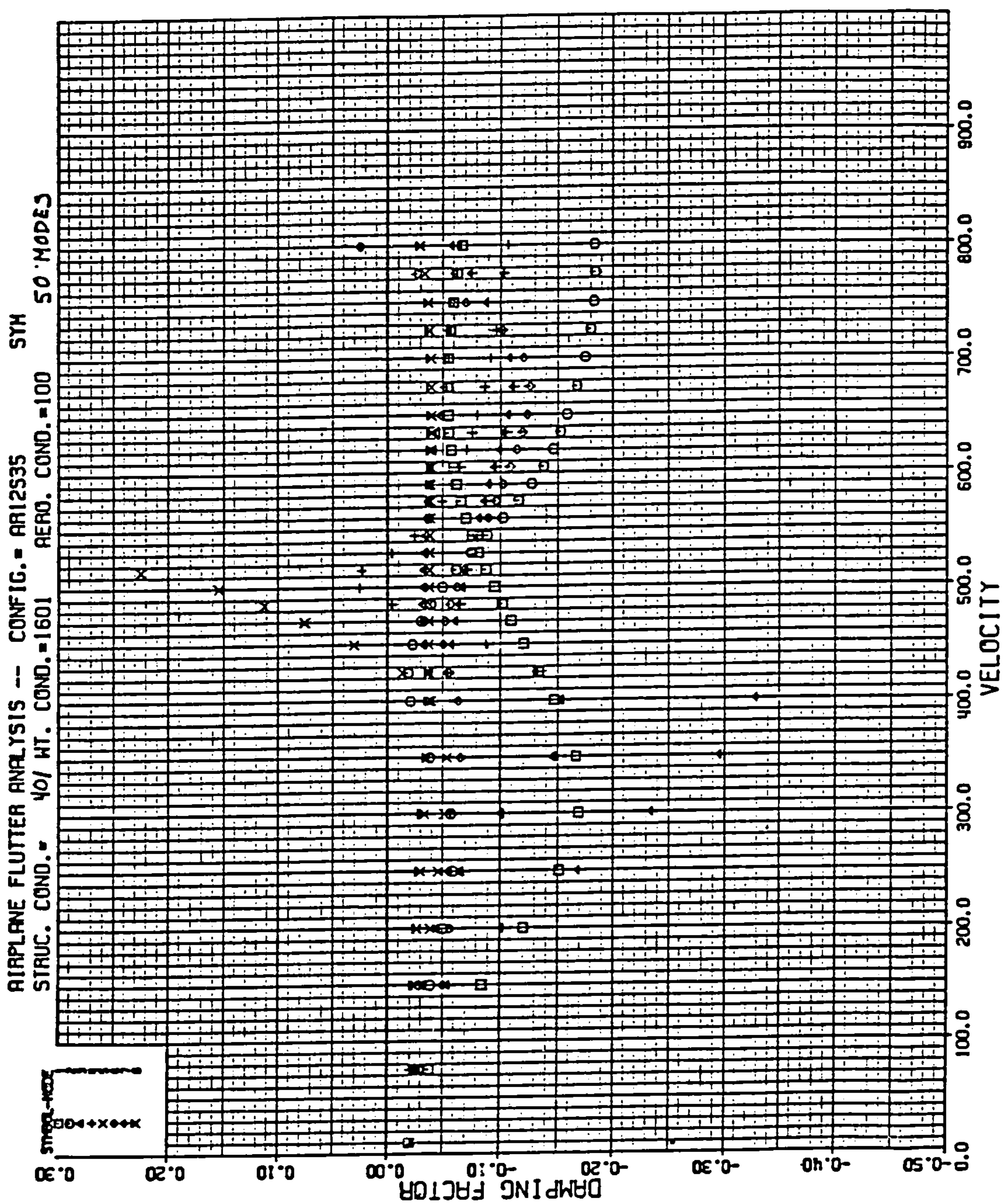


Figure 5.0-8 AR12 35 Deg. Sweep full fuel flutter plot - damping vs. velocity

5.1 Weight Analysis

The use of a finite element analysis as a weight estimating tool has presented the weight engineering community an opportunity to verify or improve statistically derived algorithms relating parameters such as aspect ratio (AR), sweep, thickness to chord ratio, and area to wing weight.

The primary method used at Lockheed-California Company to perform conceptual design studies is the Advanced Systems Synthesis and Evaluation Technique (ASSET) program. This program is capable of performing perturbation studies on a wide range of aircraft.

This automated technique was used to perform optimization studies on the Baseline aircraft varying aspect ratio, sweep, thickness to chord ratio, and wing area. Maintaining the baseline payload and range, the aircraft was optimized for minimum fuel.

A wing with an aspect ratio of 12, and a sweep of 25 degrees was determined in reference 7 to be the optimal configuration for the PADS study. In order to clearly examine the effects of the variation of individual geometric parameters on wing weight, a step by step change in these parameters was performed.

The parameters were adjusted one step at a time, with the first variation being a change in wing aspect ratio, and the second being wing sweep. In each case the baseline wing area, longitudinal center-of-lift, and thickness to chord ratio were maintained.

Section 5.1.1 specifically addresses the effect of an increase in aspect ratio on wing weight.

Section 5.1.2 addresses the further effects of then unsweeping the wing to 25 degrees.

5.1.1 Aspect Ratio 12, 35 Degree Sweep aircraft - Maintaining the Baseline longitudinal center-of-lift line required a forward movement of the wing on the fuselage for an increase in only wing aspect ratio.

This forward movement of the wing resulted in a nonrealistic location for the main landing gear. Realizing that the final wing would be able to accommodate the landing gear in a reasonable location, and that the present study was merely one iteration on the way to the final configuration, the effects of this unreasonable location were neglected.

5.1.1.1 ASSET weight statement - A mass properties estimation model was created on ASSET to determine the effects of increasing the baseline aircraft aspect ratio to 12. The weight statement generated for this configuration is presented in table 5.1-1.

5.1.1.2 ASSET group weight to aircraft component distribution -

These group weights were allocated to the major aircraft components used as input to the Mass Distribution Module (MDM). This distribution is based on the Baseline aircraft. It was assumed that the horizontal tail, vertical tail, and body mass distributions would remain the same as in the baseline. The nose landing gear and center engine remain in the same location, while the main gear and wing engines remain in the same relative location with reference to the wing geometry. The panel distribution on the wing was kept at the same percentage span and chord as the Baseline aircraft.

Fuel volume data for the new wing were generated and estimates of fuel fill and burn sequences were calculated. This was used to determine the forward center-of-gravity limit, and to prepare the load case data.

Distribution of the calculated fuel weight into the mass properties grid provided direct input to the mass distribution module.

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(524354.)		
FUEL AVAILABLE	182537.	FUEL	34.81
EXTERNAL	0.		
INTERNAL	182537.		
ZERO FUEL WEIGHT	341817.		
PAYLOAD	53550.	PAYLOAD	10.21
PASSENGERS	40460.		
BAGGAGE	13090.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	288267.		
OPERATIONAL ITEMS	8599.	OPERATIONAL ITEMS	3.29
STANDARD ITEMS	8639.		
EMPTY WEIGHT	271029.		
STRUCTURE	174271.	STRUCTURE	33.24
WING	85046.		
ROTOR	0.		
TAIL	9042.		
BODY	48590.		
ALIGHTING GEAR	21909.		
ENGINE SECTION AND NACELLE	9684.		
PROPULSION	40459.	PROPULSION	7.72
CRUISE ENGINES	32693.		
LIFT ENGINES	0.		
THRUST REVERSER	4923.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	214.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2092.		
DRIVE SYSTEM	0.		
SYSTEMS	56299.		
FLIGHT CONTROLS	6255.		
AUXILIARY POWER PLANT	1202.		
INSTUMENTS	998.		
HYDRAULIC AND PNEUMATIC	2759.		
ELECTRICAL	5797.		
AVIONICS	2827.	SYSTEMS	10.74
ARMAMEMT	0.		
FURNISHINGS AND EQUIPMENT	30267.		
AIR-CONDITIONING	5811.		
ANTI-ICING	383.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

TABLE 5.1-1 AR12 35 DEGREE SWEEP ASSET WEIGHT STATEMENT

5.1.1.3 Center-of-gravity envelope - The center-of-gravity envelope used for the PADS AR12 35 Degree Sweep aircraft is presented in figure 5.1-1. The following assumptions were made in the derivation of the envelope:

- Maximum zero fuel weight and maximum landing weight of the AR12 would be determined by increasing the baseline design weights by the structural weight increment required for the increased aspect ratio. This is approximately 37,000 pounds. Thus, the AR12 aircraft design weights are:

  Maximum Zero Fuel Weight (MZFW) 375,000 lb

  Maximum Landing Weight (MLW) 405,000 lb

- The Maximum Takeoff Gross Weight was determined as the difference between the increase in structural weight and the decrease in fuel required added to the baseline maximum takeoff gross weight. Maximum taxi weight allows for a fuel allowance of 2000 pounds for taxi.

  Maximum Takeoff Gross Weight (MTOW) 524,354 lb

  Maximum Taxi Weight (MTW) 526,354 lb

- The problem with landing gear location would be ignored, realizing that the problem encountered with a high aspect ratio aircraft reduces with less sweep angle.

- The forward center-of-gravity envelope immediately above MZFW is determined by fuel loading.

- The forward center-of-gravity limit at MTOW would be derived as the center-of-gravity location that would maintain the same tail loading as the baseline at MTOW and forward center of gravity.

- The aft center-of-gravity limit at MTOW was assumed to be the same longitudinal location as on the baseline, neglecting the unrealistic landing gear location.

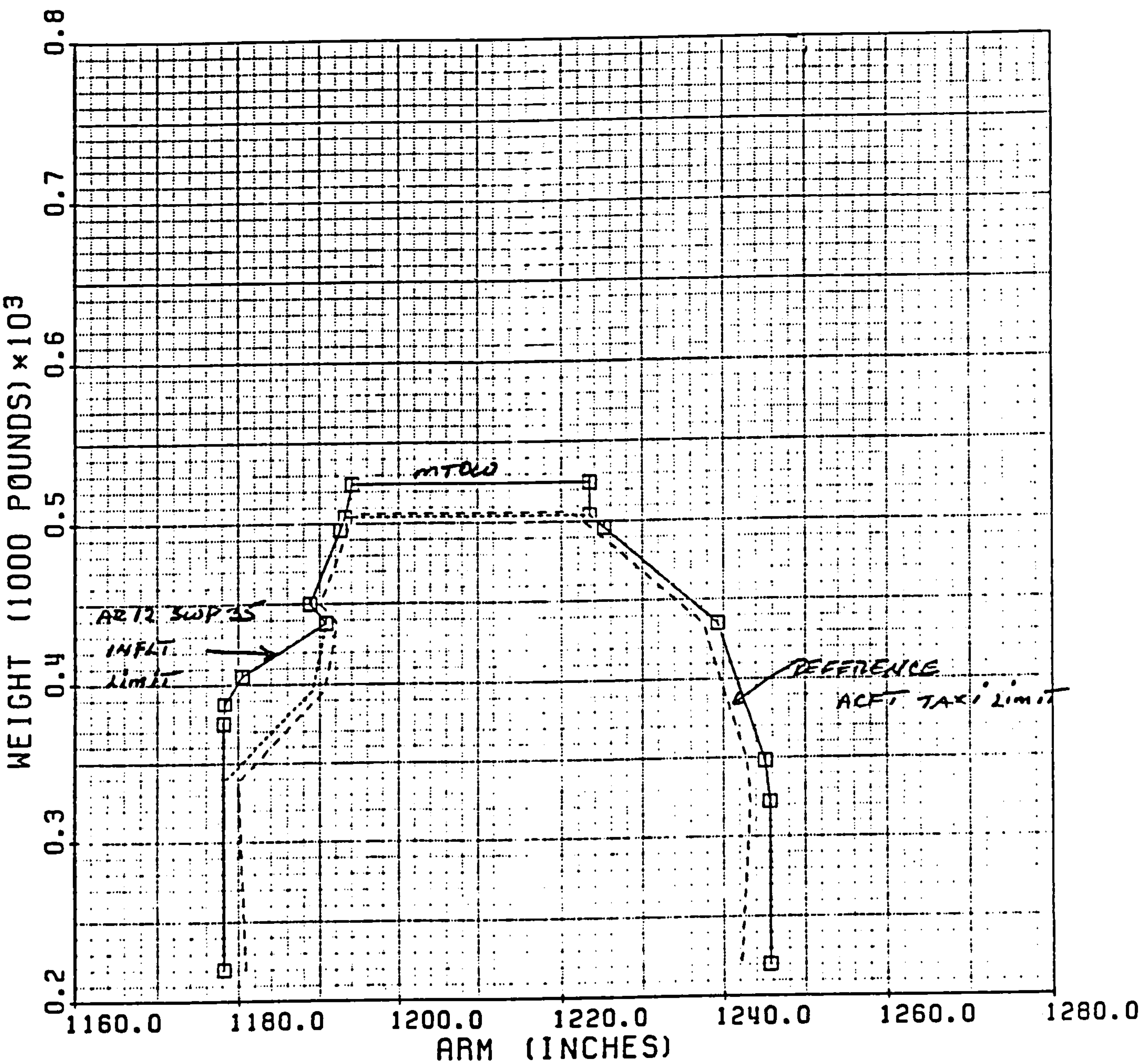


FIGURE 5.1-1 AR12 35 Degree Sweep Center of Gravity Envelope

5.1.1.4 Inertia conditions - Four inertia conditions were analysed on the baseline aircraft. These conditions represented weight distributions for the critical design loads for the Baseline aircraft. The same conditions were assumed to be critical for the PADS aspect ratio 12 aircraft. Table 5.1-2 presents a summary.

These conditions were:

Flight Conditions (gear up)

- o WUA - 901 Forward center of gravity, maximum takeoff weight, minimum wing bending relief, maximum zero fuel weight.
- o WUA - 910 Forward center of gravity, minimum fuel, maximum zero fuel weight.

Landing Conditions (gear down)

- o WDA - 963 Forward center-of-gravity, maximum landing weight, maximum zero fuel weight.
- o WDA - 965 Aft center-of-gravity, maximum zero fuel weight, at maximum taxi weight.

Plots of these conditions are presented in figure 5.1-2. The Fuel burn sequence presented in figure 5.1-3 assumes a normal tank to engine fuel feed, with only minor variations for tank equalization. It is similar to the sequence used by the Baseline aircraft for normal flight.

These load cases were stored under the following Panvalet names:

WUA - 901	E873W901RD and E873W901RT
WUA - 910	E873W910RD and E873W910RT
WDA - 963	E873W963RD and E873W963RT
WDA - 964	E873W965RD and E873W965RT

Table 5.1-2 Inertia Conditions for AR12 35 Degree Sweep Design

Condition	Name	Description
1	Full fuel	Gear up, fwd cg limit, max takeoff weight, 149,354 lb. fuel, (WUA-901)
2	Low fuel	Gear up, fwd cg limit, 12,300 lb. fuel, (WUA-910)
3	WDA-963	Gear down, fwd cg limit, max landing wt (405k), 30,000 lb. fuel
4	WDA-965	Gear down, aft cg limit, max taxi wt., 151,354 lb. fuel

Aircraft Loading Condition
(all weight shown in pounds)

Condition	1	2	3	4
Op. empty wt.	279171.3	279171.3	279171.3	279171.3
Payload	95828.7	95828.7	95828.7	95830.6
Zero fuel wt.	375000.0	375000.0	375000.0	375001.9
Fuel wt.	149354.0	12300.0	30000.0	151354.0
Gross wt.	524353.9	387300.0	405000.0	526355.8
XCG	1193.9	1178.3	1180.5	1221.1
YCG	235.8	193.0	206.4	247.1
ZCG	197.3	199.9	195.9	194.0

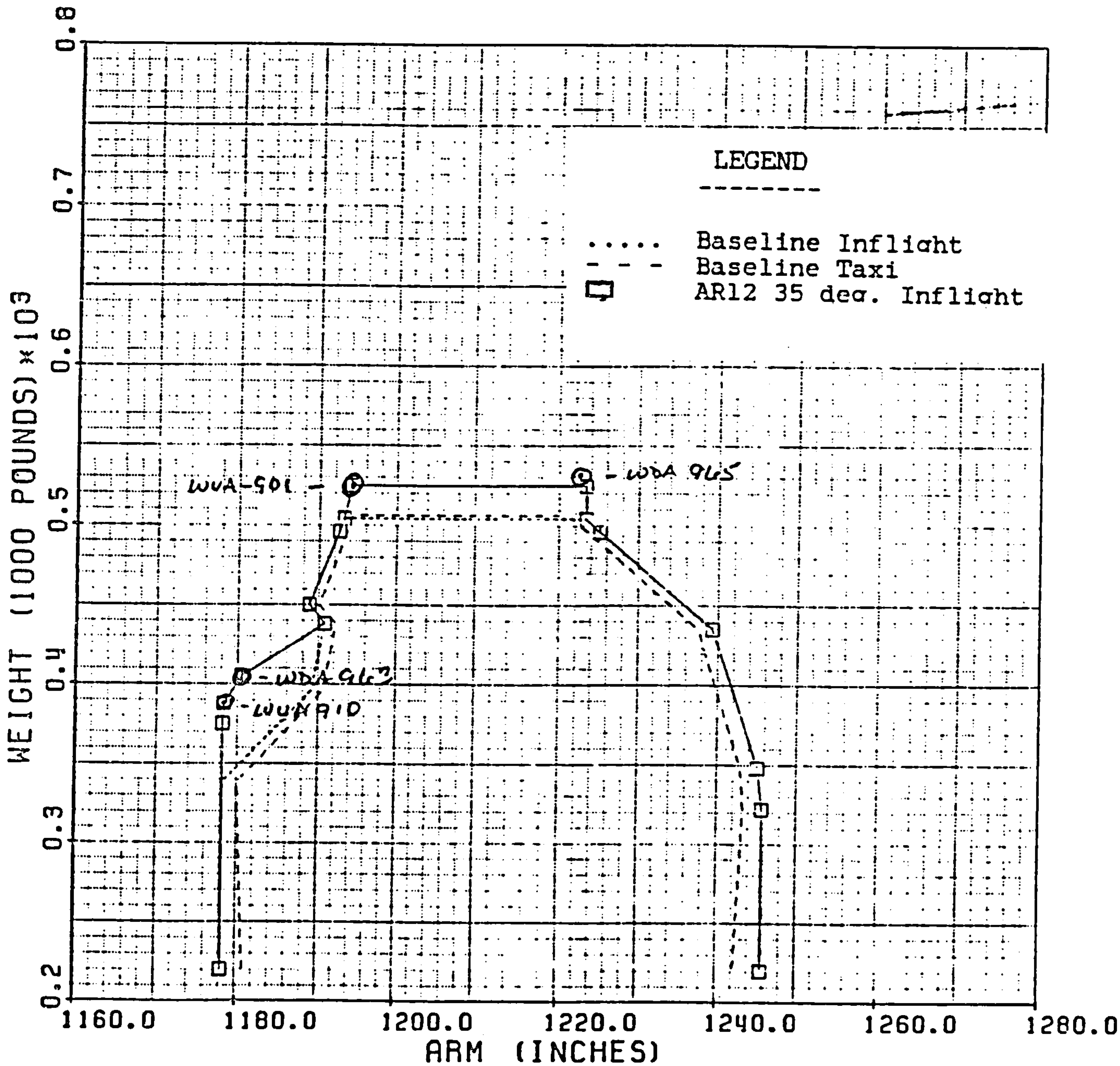


FIGURE 5.1-2 AR12 35 Degree Sweep Loading Conditions

Figure 5.1-3 Fuel Burn Sequence

ASPECT RATIO 12 FUEL BURN @ 7.1 lb/GAL.

Condition	Tank 2L Outboard	Tank 2L Inboard	Tank 1	Tank 3	Tank 2R Inboard	Tank 2R Outboard	Total
Full Wing Fuel	8023	15954	49711	49711	15954	8023	147376
14K Fuel Burned	8023	13621	45044	45044	13621	8023	133376
Tanks Evened	8023	13621	43288	43288	13621	8023	129864
Inb'd Sump Fuel Remng	8023	1060	18166	18166	1060	8023	54498
Burn to 30K	3940	1060	10000	10000	1060	3940	30000
Burn to 12.3K	990	1060	4100	4100	1060	990	12300

NOTE: For tank 2L or 2R, the inboard compartment is used until only the sump fuel remains, then the outboard compartment will gravity feed the inboard compartment.

Standard fuel Usage:

(1) Fuel usage is direct tank-to-engine during taxi, takeoff and climbout until total fuel load is reduced by 14,000 pounds.

(2) Cross-feed from tanks 1 and 3 until tanks 1, 2, and 3 are equal.

(3) When tanks are equalized, fuel usage is direct tank-to-engine until end of flight.

5.1.1.5 NASTRAN wing weight estimate - NASTRAN element weights were calculated as described in section 3.3.5 for the Aspect Ratio 12, Sweep 35 Aircraft. Special adjustment had to be made to the rib weight.

The PADS sizing module (PSASA) assumes constant rib spacing for sizing of the covers. This spacing was held constant between the Baseline and AR12 designs. However, the finite element representations for the AR12 designs have a rib spacing which is increased due to the streching of the wing.

In order to maintain constant rib spacing as assumed in the cover sizing, the NASTRAN model would have required an increased number of ribs. However, the stretching process used to derive the AR12 wing and NASTRAN geometry did not change the number of ribs, only their location and chord length. The Grid Point Weight Generator uses the NASTRAN geometry and thickness data to calculate weight, and would calculate weight only for the number of ribs represented in the model. This weight had to be adjusted to account for a larger number of ribs.

It was assumed that the tbar (average thickness) of the ribs would remain constant. The number of ribs was assumed to increase linearly with an increase in span.

Half span of Baseline aircraft = 988 inches
Half span for AR12 aircraft = 1233 inches

Ratio to be applied to rib regional factors = 1233/988 = 1.2480

This ratio was applied to the Baseline rib regional factors to form regional factors for the AR12 ribs.

RIB DESIGN REGIONS	BASELINE REGIONAL FACTORS	AR12 REGIONAL FACTORS
1	1.6609	2.0728
2	1.2880	1.6074
3	1.3265	1.6554
4	1.0588	1.3214
5	0.7580	0.9460
6	0.6372	0.7952
7	0.8710	1.0870

The span of both AR12 designs were the same, so these factors were applied to both.

The regional weight factors used on the NASTRAN estimated weights are presented in tables 5.1-3 thru 5.1-6. The design regions are shown in figure 3.3-9.

Table 5.1-3 Upper Cover Regional Weight Factors

DESIGN REGION	REGIONAL FACTORS
1	1.30000
2	1.22260
3	1.10119
4	1.08630
5	1.07580
6	1.18520
7	1.29705

Table 5.1-4 Lower Cover Regional Weight Factors

DESIGN REGION	REGIONAL FACTORS
1	1.32790
2	1.12800
3	1.05980
4	1.14470
5	1.01660
6	1.10940
7	1.26930

Table 5.1-5 Spar Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	0.64430
2	0.68060
3	0.83680
4	0.92640
5	0.74260
6	0.91520
7	1.19445

Table 5.1-6 Rib Weight Regional Factors

DESIGN REGION	REGIONAL FACTORS
1	2.07280
2	1.60740
3	1.65540
4	1.32140
5	0.94600
6	0.79520
7	1.08700

5.1.1.6 Comparison with conventional estimation - The wing weight breakdown derived from the PADS design cycle (adjusted FEM weights) is presented below:

Wing Component	Weight (lb)
Center Box Structure	(7795)
Upper Cover	2401
Lower Cover	3361
Spars and Ribs	2033
Outer Wing Structure	(61503)
Upper Cover	14421
Lower Cover	24417
Spars	3952
Ribs	5547
Secondary	13166
Total Wing Structural Weight	(69298)

The total wing weight predicted by conventional methods for this configuration was 85046.

5.1.2 Aspect Ratio 12, 25 Degree Sweep aircraft - Sweeping the aspect ratio 12 wing forward to a 25 degree quarter chord sweep made the landing gear location a reasonable place. The reference longitudinal center of lift line was maintained, thus the wing was moved aft on the fuselage from the sweep 35 position.

5.1.2.1 ASSET weight statement - A mass properties estimation model was created in ASSET to determine the effects of increasing the Baseline aircraft aspect ratio to 12, and unsweeping the wing to 25 degrees. The weight statement generated for this configuration is presented in table 5.1-7.

5.1.2.2 ASSET group weight to aircraft component distribution -

These group weights were allocated to the major aircraft components used as input to the Mass Distribution Module (MDM). This distribution is based on the Baseline aircraft. It was assumed that the horizontal tail, vertical tail, and body mass distributions would remain the same as in the Baseline. The nose landing gear and center engine remain in the same location, while the main gear and wing engines remain in the same relative location with reference to the wing geometry.

Fuel volume data for the new wing were generated and estimates of fuel fill and burn sequences were calculated. This was used to determine the forward center-of-gravity limit, and to prepare the load case data.

Distribution of the calculated fuel weight into the mass properties grid provided direct input to the mass distribution module.

5.1.2.3 Center-of-gravity envelope - The center-of-gravity envelope used for the PADS AR12, Sweep 25 aircraft is presented in figure 5.1-4. The following assumptions were made in the derivation of the envelope:

- o Maximum zero fuel weight and maximum landing weight of the AR12 would be determined by increasing the baseline design weights by the structural weight increment required for the increased aspect ratio. This is approximately 26,000 pounds. Thus, the AR12 aircraft design weights are:

  Maximum Zero Fuel Weight (MZFW) 364,000 lb

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(520273.)		
FUEL AVAILABLE	188008.	FUEL	36.14
EXTERNAL	0.		
INTERNAL	188008.		
ZERO FUEL WEIGHT	332185.		
PAYLOAD	53550.	PAYLOAD	10.29
PASSENGERS	40460.		
BAGGAGE	13090.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	278635.		
OPERATIONAL ITEMS	8599.	OPERATIONAL ITEMS	3.31
STANDARD ITEMS	8647.		
EMPTY WEIGHT	261389.		
STRUCTURE	164652.	STRUCTURE	31.65
WING	75556.		
ROTOR	0.		
TAIL	9042.		
BODY	48549.		
ALIGHTING GEAR	21821.		
ENGINE SECTION AND NACELLE	9684.		
PROPULSION	40464.	PROPULSION	7.78
CRUISE ENGINES	32693.		
LIFT ENGINES	0.		
THRUST REVERSER	4923.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	214.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2097.		
DRIVE SYSTEM	0.		
SYSTEMS	56273.		
FLIGHT CONTROLS	6248.		
AUXILIARY POWER PLANT	1202.		
INSTUMENTS	998.		
HYDRAULIC AND PNEUMATIC	2740.		
ELECTRICAL	5797.		
AVIONICS	2827.	SYSTEMS	10.82
ARMAMEMT	0.		
FURNISHINGS AND EQUIPMENT	30267.		
AIR-CONDITIONING	5811.		
ANTI-ICING	383.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

TABLE 5.1-7 AR12 25 DEGREE SWEEP ASSET WEIGHT STATEMENT

Maximum Landing Weight (MLW) 394,000 lb

- o The maximum takeoff gross weight was determined as the difference between the increase in structural weight and the decrease in fuel required added to the Baseline maximum takeoff gross weight. Maximum taxi weight allows for a fuel allowance of 2000 pounds for taxi.

  Maximum Takeoff Gross Weight (MTOW) 520,275 lb

  Maximum Taxi Weight (MTW) 522,275 lb

- o The forward center-of-gravity envelope immediately above MZFW is determined by fuel loading.

- o The forward center-of-gravity limit at MTOW would be derived as the center of gravity location that would maintain the same tail loading as the baseline at MTOW and forward center of gravity.

- o The aft center of gravity limit at MTOW was assumed to be the same longitudinal location as on the baseline.

5.1.2.4 Inertia conditions - Four inertia conditions were analysed on the baseline aircraft for reference purposes. These conditions represented the conditions that determined many of the design loads for the baseline aircraft. The same conditions were assumed to be critical for the PADS aspect ratio 12 aircraft. Table 5.1-8 presents a summary.

These conditions were:

Flight Conditions (gear up)

- o WUA - 901 Forward center of gravity, maximum takeoff weight, minimum wing bending relief, maximum zero fuel weight.

- o WUA - 910 Forward center of gravity, minimum fuel, maximum zero fuel weight.

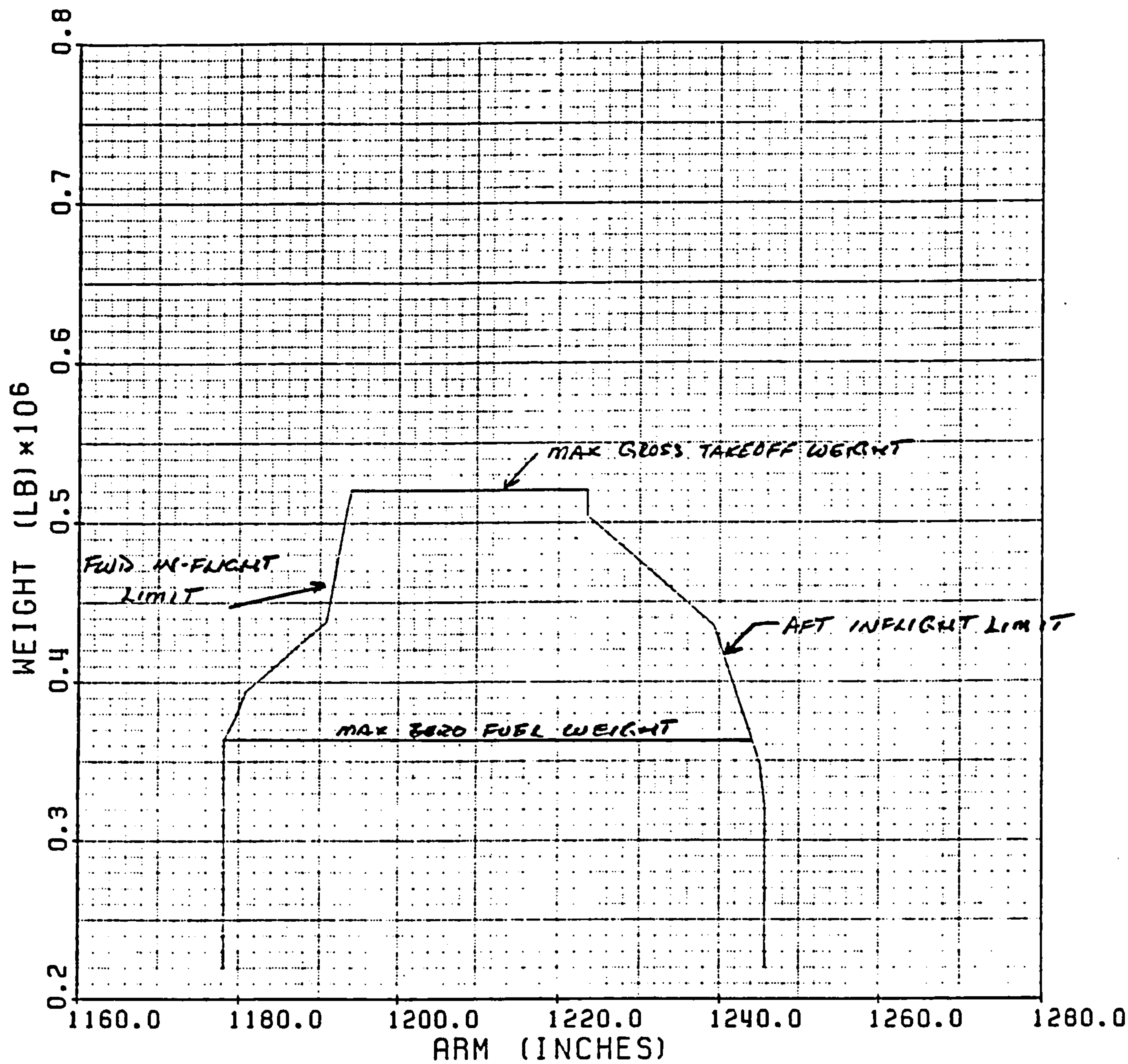


FIGURE 5.1-4 AR12 25 Degree Sweep Center of Gravity Envelope

Table 5.1-8 Inertia Conditions for AR12 25 Degree Sweep Design

Condition	Name	Description
1	Full fuel	Gear up, fwd cg limit, max takeoff weight, 156,276 lb. fuel, (WUA-901)
2	Low fuel	Gear up, fwd cg limit, 12,300 lb. fuel, (WUA-910)
3	WDA-963	Gear down, fwd cg limit, max landing wt (394k), 30,000 lb. fuel
4	WDA-965	Gear down, aft cg limit, max taxi wt., 158,276 lb. fuel

Aircraft Loading Condition
(all weight shown in pounds)

Condition	1	2	3	4
Op. empty wt.	269539.3	269539.3	269539.3	269539.3
Payload	94458.8	94461.6	94461.3	94459.1
Zero fuel wt.	363998.1	364000.9	364000.6	363998.4
Fuel wt.	156276.0	12300.0	30000.0	158276.0
Gross wt.	520274.1	376300.9	394000.5	522274.3
XCG	1193.9	1179.5	1180.8	1221.1
YCG	219.8	170.5	185.3	231.7
ZCG	196.2	198.7	194.7	192.9

Landing Conditions (gear down)

- o WDA - 963 Forward center-of-gravity, maximum landing weight, maximum zero fuel weight.

- o WDA - 965 Aft center-of-gravity, maximum zero fuel weight, at maximum taxi weight.

Plots of these conditions are presented in figure 5.1-5. The Fuel burn sequence presented in figure 5.1-6 assumes a normal tank to engine fuel feed, with only minor variations for tank equalization. It is similar to the sequence used by the baseline aircraft for normal flight.

These load cases were stored under the following Panvalet names:

WUA - 901	E907W901MD and E907W901MT
WUA - 910	E907W910MD and E907W910MT
WDA - 963	E907W963MD and E907W963MT
WDA - 964	E907W965MD and E907W965MT

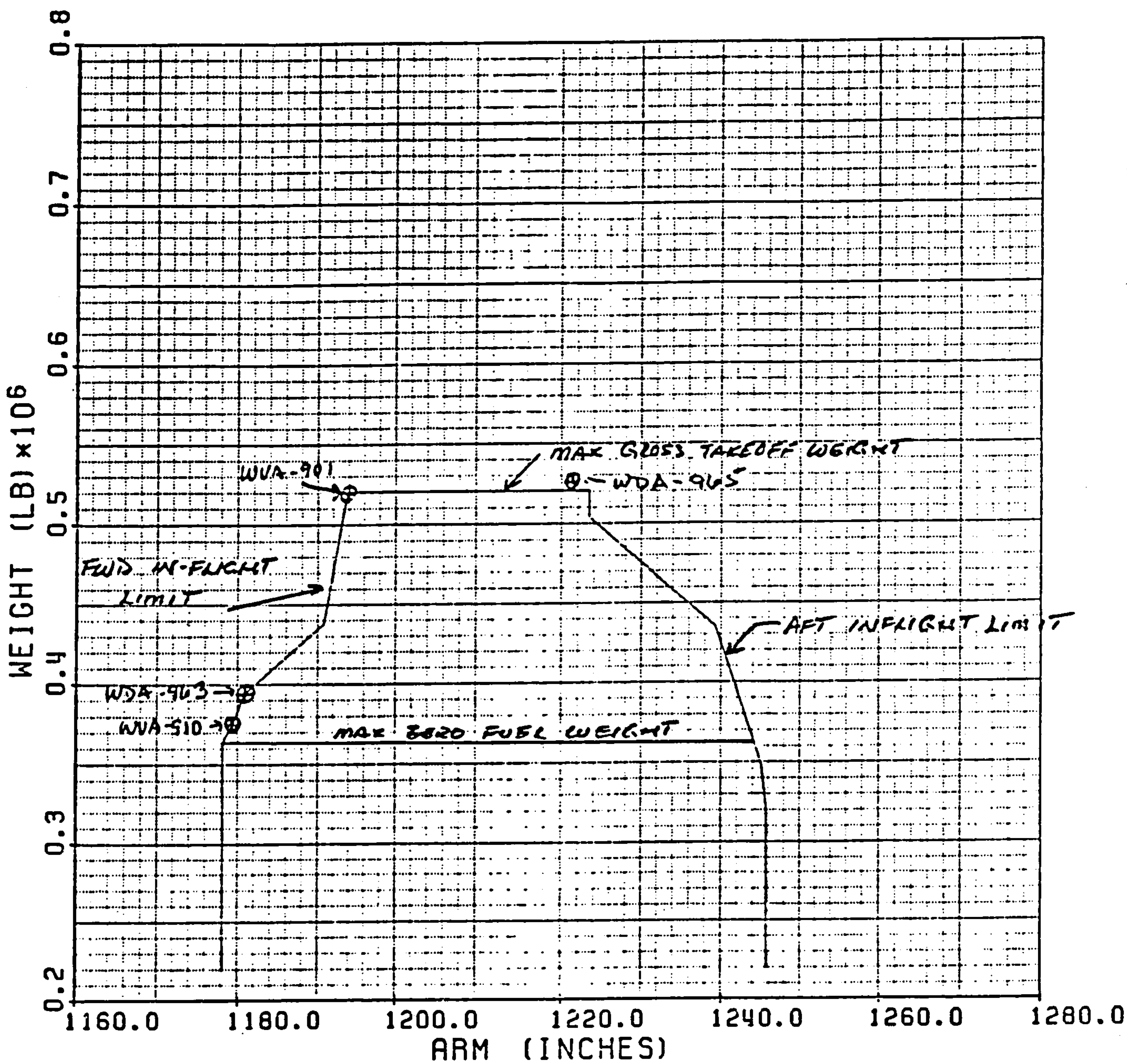


FIGURE 5.1-5 AR12 25 Degree Sweep Loading Conditions

Figure 5.1-6 Fuel Burn Sequence

ASPECT RATIO 12 FUEL BURN @ 7.1 lb/GAL.

Condition	Tank 2L Outboard	Tank 2L Inboard	Tank 1	Tank 3	Tank 2R Inboard	Tank 2R Outboard	Total
Full Wing Fuel	8148	16230	50593	50593	16230	8148	149942
14K Fuel Burned	8148	13897	45926	45926	13897	8148	135942
Tanks Evened	8148	13897	44090	44090	13897	8148	132270
Inb'd Sump Fuel Remng	8148	1060	18416	18416	1060	8148	55248
Burn to 30K	3940	1060	10000	10000	1060	3940	30000
Burn to 12.3K	990	1060	4100	4100	1060	990	12300

NOTE: For tank 2L or 2R, the inboard compartment is used until only the sump fuel remains, then the outboard compartment will gravity feed the inboard compartment.

Standard fuel Usage:

(1) Fuel usage is direct tank-to-engine during taxi, takeoff and climbout until total fuel load is reduced by 14,000 pounds.

(2) Cross-feed from tanks 1 and 3 until tanks 1, 2, and 3 are equal.

(3) When tanks are equalized, fuel usage is direct tank-to-engine until end of flight.

5.1.2.5 NASTRAN wing weight estimate - NASTRAN element weights were calculated as described in Section 3.3.5 for the Aspect Ratio 12, Sweep 25 Aircraft. The same adjustment had to be made to the rib weight as for the AR12 35 Degree Sweep design. The development of the regional factors for both AR12 designs is discussed in section 5.1.1.5.

5.1.2.6 Comparison with conventional estimation - The wing weight breakdown derived from the PADS design cycle (adjusted FEM weights) is presented below:

Wing Component	Weight (lb)
Center Box Structure	(8739)
Upper Cover	2716
Lower Cover	3996
Spars and Ribs	2027
Outer Wing Structure	(63466)
Upper Cover	14677
Lower Cover	25964
Spars	3578
Ribs	6081
Secondary	13166
Total Wing Structural Weight	(72205)

The total wing weight predicted by conventional methods for this configuration was 75556.

5.1.3 Weight assessment conclusion and summary - The estimation of wing structural weight through parametric techniques utilizing primarily statistical data is hazardous for high aspect ratio wing aircraft. Statistics are not available in this range, thus extrapolation from known designs becomes questionable. The utilization of active control systems in contemporary aircraft is also relatively new, and thus statistical weight benefits at high aspect ratios are not available. The Preliminary Aeroelastic Design Study attempts to address these problems through the utilization of finite element modeling analysis.

PADS presents the opportunity to examine analytically a vast variety of wing configurations. With refinement, PADS provides a potential method of adding to the statistical database in an age where the volume of aircraft being produced, and thus statistical data, is dwindling. More importantly, however, PADS enables detailed aeroelastic and structural sensitivity analysis far earlier in the conceptual design phase than was ever before possible.

6.0 PADS WING SIZING SUMMARY

The wing covers, represented as CMEMQ finite elements, which are designed in the PADS sizing process are listed in table 6.0-1. Table 6.0-2 summarizes the upper, lower, and total weights for the Baseline, AR12 35 Degree Sweep, and AR12 25 Degree Sweep half airplane covers. Table 6.0-3 presents half airplane weight increments as computed directly from the Grid Weight Point Generator in NASTRAN for various design configurations.

Table 6.0-1 - FEM IDS WHICH ARE SIZED

10103 - 10110	10203 - 10210
10303 - 10310	10403 - 10410
10503 - 10510	10603 - 10610
11003 - 11010	10803 - 10810
10903 - 10910	11003 - 11010
11103 - 11110	11203 - 11210
11303 - 11310	11403 - 11410
11503 - 11510	11603 - 11610
11703 - 11710	11803 - 11810
11903 - 11910	12003 - 12010
12103 - 12110	12203 - 12210
12303 - 12310	11403 - 11410
12503 - 12510	11603 - 11610
12703 - 12710	11803 - 11810

BASELINE DESIGN

#	PANNAME	NAME	EXTERNAL LOADS	PROPS TO BE UPDATED	ACS GAIN	STRESS MARGINS	UPPER	LOWER	TOTAL
1	E580MPRPM2	PRODUCTION PRPS					4618.62	6342.52	10961.
2	E750MPRHA1	RIGID	RIGID	ARBITRARY	11.33	NO	4147.75	5123.20	9271.
3	E750MPRHA2	1ST FLEX	1ST FLEX	RIGID	11.33	NO	3870.20	4845.49	8716.
4	E750MPRHA3	1ST ACS OFF	1ST ACS OFF	1ST FLEX	0.0	NO	4042.62	5221.34	9264.
5	E750MPRHA4	2ND ACS OFF	2ND ACS OFF	1ST ACS OFF	0.0	NO	4101.59	5390.61	9492.
6	E750MPRHA5	2ND FLEX	2ND FLEX	1ST FLEX	11.33	NO	3829.56	4847.12	8677.
7	E750MPRHA6	3RD ACS OFF	3RD ACS OFF	2ND ACS OFF	0.0	NO	4103.17	5418.23	9521.
8	E750MPRHA7	1ST WITH MARGINS	2ND FLEX	2ND FLEX	11.33	YES	4098.19	5530.03	9628.
9	E750MPRHA8	2ND WITH MARGINS	2ND FLEX	1ST W MARGS	11.33	YES	4106.54	5611.50	9718.
10	E750MPRHA9	PROD LOADS PRPS	PROD LOADS	2ND W MARG	11.33	YES	4223.02	5909.67	10133.
11	E750MPRPB1	2ND FLEX FSD PASS 1	2ND FLEX	1ST FLEX	11.33	NO	3901.18	4939.47	8841.
12	E750MPRPB2	2ND FLEX FSD PASS 2	2ND FLEX	2ND FLEX	11.33	NO	3904.61	4956.35	8861.
13	E750GPRHA1	GUST W MARGS NO NAC	2ND W MARGS	2ND FLEX	11.33	YES	4116.64	5642.52	9759.
			AND GUST		11.3				
14	E750GPRHA3	GUST W MARGS	2ND W MARGS	2ND FLEX	11.33	YES	4163.08	5752.55	9916.
			AND GUST	2ND FLEX	11.3				

AR12 35 DEGREE SWEEP DESIGN

#	PANNAME	NAME	EXTERNAL LOADS	PROPS TO BE UPDATED	ACS GAIN	STRESS MARGINS	UPPER	LOWER	TOTAL
15	E873MPRHA1	RIGID	RIGID	ARBITRARY	11.33	YES	7511.42	12358.22	19870.
16	E873MPRHA2	1ST FLEX	1ST FLEX	RIGID	11.33	YES	6512.45	10745.62	17258.
17	E873MPRHA3	2ND FLEX	2ND FLEX	1ST FLEX	11.33	YES	6335.90	10695.23	17031.
18	E873MPRHA4	15 DEG/G	15 DEG/G	2ND FLEX	15.0	YES	6211.04	10748.68	16960.
19	E873MPRHA5	2 G ACS ON	15 DEG/G	2ND FLEX	0.0	YES	5154.62	8028.01	13183.
		.4 * ACS OFF							
20	E873MPRHA6	2.5 G NO ACS	2.5 G NO ACS	2ND FLEX	0.0	YES	6725.68	11876.96	18603.
21	E873MPRHA7	20 DEG/G	20 DEG/G	2ND FLEX	20.0	YES	6181.05	10655.39	16836.
22	E873MPRHA8	30 DEG/G	30 DEG/G	2ND FLEX	30.0	YES	6319.27	10984.63	17304.
23	E873GPRHA1	0 DEG/G GUST	2ND FLEX	2ND FLEX	11.33	YES	6483.44	11113.16	17597.
			0. DEG/G GUST		11.3				
24	E873GPRHA2	11.3 DEG/G GUST	2ND FLEX	2ND FLEX	11.33	YES	6418.52	10881.62	17300.
			11.3 DEG/G GUST		11.3				
25	E873GPRHA3	15 DEG/G GUST	2ND FLEX	2ND FLEX	11.33	YES	6399.36	10830.03	17229.
			15 DEG/G GUST		11.3				

AR12 25 DEGREE SWEEP DESIGN

#	PANNAME	NAME	EXTERNAL LOADS	PROPS TO BE UPDATED	ACS GAIN	STRESS MARGINS	UPPER	LOWER	TOTAL
26	E907MPRHA4	9 DEG/G	9 DEG/G	2ND FLEX	9.0	YES	6648.10	11792.96	18441.
27	E907MPRHA5	2.5 G NO ACS	9 DEG/G	2ND FLEX	0.0	YES	7011.35	11902.33	18914.
28	E907MPRHA6	18 DEG/G	18 DEG/G	2ND FLEX	18.0	YES	6645.31	11776.91	18422.
29	E907MPRHA7	3RD FLEX	3RD FLEX	2ND FLEX	11.33	YES	6591.19	11642.60	18234.

TABLE 6.0-2 - WEIGHT COMPARISON FOR PADS SIZINGS

TABLE 6.0-3 HALF AIRPLANE COVER WEIGHT INCREMENTS COMPUTED ON PADS FINITE ELEMENT MODELS WITHOUT ADJUSTMENT.

	AIRCRAFT DESIGN BASELINE	AR12S35	AR12S25
A) INCREASE IN DESIGN WING WEIGHT WHEN THE ACTIVE CONTROL SYSTEM IS TURNED OFF. STRESS MARGINS OF SAFETY WERE NOT INCLUDED IN EITHER THE ON OR OFF CONDITION.	845.	-	-
B) INCREASE IN DESIGN WING WEIGHT WHEN THE ACTIVE CONTROL SYSTEM IS TURNED OFF. STRESS MARGINS OF SAFETY WERE INCLUDED IN BOTH ON AND OFF CONDITION.	-	1572.	680.
C) INCREASE IN WEIGHT FOR APPLYING STRESS MARGINS OF SAFETY FOR COVER DESIGN. THIS WEIGHT IS FOR A DESIGN WITH THE ACTIVE CONTROL SYSTEM ON.	1041.	-	-
D) INCREASE IN WEIGHT FOR INCLUSION OF GUST LOADS WITH 2ND FLEX STATIC LOADS. THIS WEIGHT IS FOR A DESIGN WITH THE ACTIVE CONTROL SYSTEM ON.	198.	269.	-
E) INCREASE IN WEIGHT FOR CHANGING WING GEOMETRY FROM BASELINE. THIS WEIGHT IS FOR FINAL FLEX DESIGN WITH THE ACTIVE CONTROL SYSTEM ON AND STRESS MARGINS OF SAFETY APPLIED.	-	7403.	8606.

7.0 WING OPTIMIZATION RESULTS FOR MINIMUM BLOCK FUEL

The ASSET computer program was given input data the Baseline airplane model. The weight equation was updated to match the PADS results for sweep/35 degrees and aspect ratio/12 designs. The PADS design for the 25 degrees of sweep was heavier than the 35 degees of sweep design at aspect ratio of 12. The weight equation matching was made at only 35 degrees of sweep for purposes of economy. While the designs at 25 and 35 degrees of sweep indicate that the optimum wing design may be greater than 35 degrees, for purposes of this study, it is assumed that a design constraint exists at 35 degrees of sweep. It should be noted however that the main landing gear constraint would force the sweep to be closer to 25 degrees than 35 degrees. For this study, in addition to the gear constraint, the flutter speed constraint was not active.

Another word of caution, the weight equation matching was made at aspect ratios 7.63 and 12. However, the way the standard weight equation was formulated, the modification due to matching PADS results at aspect ratio 12 produced a rather flat wing weight function with aspect ratio. It is suspected that the modified weight equation is outside the reasonable limits of the weight model except near aspect ratios 7.63 and 12.

The Baseline model on ASSET was run with the following variations in wing parameters:

AR = 10, 11, 12, 13, 14

t/c = 9, 10, 11, 12

For each configuration the aircraft was sized to perform the design range of 4780 n.mi.

Block fuel at the design range are shown in the carpet plot of figure 7.0-1. The same data is shown as a knot-hole plot in figure 7.0-2. It shows that the optimum wing parameters for minimum block fuel are (approximately):

AR = 14+

t/c = 11.00

A listing of aircraft characteristics is given in table 7.0-1. Note that in keeping T/W and W/S fixed, the field length constraints become variables. To be more precise, T/W and W/S should be adjusted to meet fixed constraints. The approach used in this exercise will not significantly affect results. The numbers for T/W and W/S are different from the numbers published in table 4.0-1. The differences are small and should not have a sigificant impact on the results in this section. The configuration parameters in this section are the actual parameters that made up the ASSET weight estimates used in the PADS analysis. It was judged that the differences did not justify the recomputation of data presented in section 4.0.

A summary of ASSET runs is shown in table 7.0-2.

Plots of fuel weight, wing weight, zero fuel weight, OEW weight, wing area, body weight, and engine scale versus aspect ratio at constant t/c are shown in figures 7.0-3 through 7.0-9. These plots show variations of key aircraft parameters in addition to fuel weight.

The mission summary for AR = 9 and t/c = 10, is shown in table 7.0-3.

These results may be compared to the ASSET study described in section 4.0. A summary of ASSET and PADS point designs and the optimum designs with and without PADS weight data update for 35 degrees of sweep are shown in table 7.0-4. The updating of the ASSET weight equation indicates the following:

1) inputting lower values of wing weight into ASSET results in higher aspect ratio designs for maximum fuel efficiency.

2) the optimum thickness to chord ratio appears to be a independent of the aspect ratio wing weight modification.

3) the block fuel improvement is 6.7 percent for increasing aspect ratio from 12 to 14 for standard weight equation.

4) the block fuel improvement is 5.1 percent at aspect ratio 12 when going from standard weight equation to the modified weight equation.

5) the block fuel improvement is 7.4 percent when going from standard weight equation to modified weight equation and from aspect ratio 12 to 14.

The optimum aspect ratio movement from 12 to 14+ appears to be a significant shift even for a paper exercise. Another PADS analysis would be adviseable, especially in the light of the deterioration of the flutter speed with increased aspect ratio.

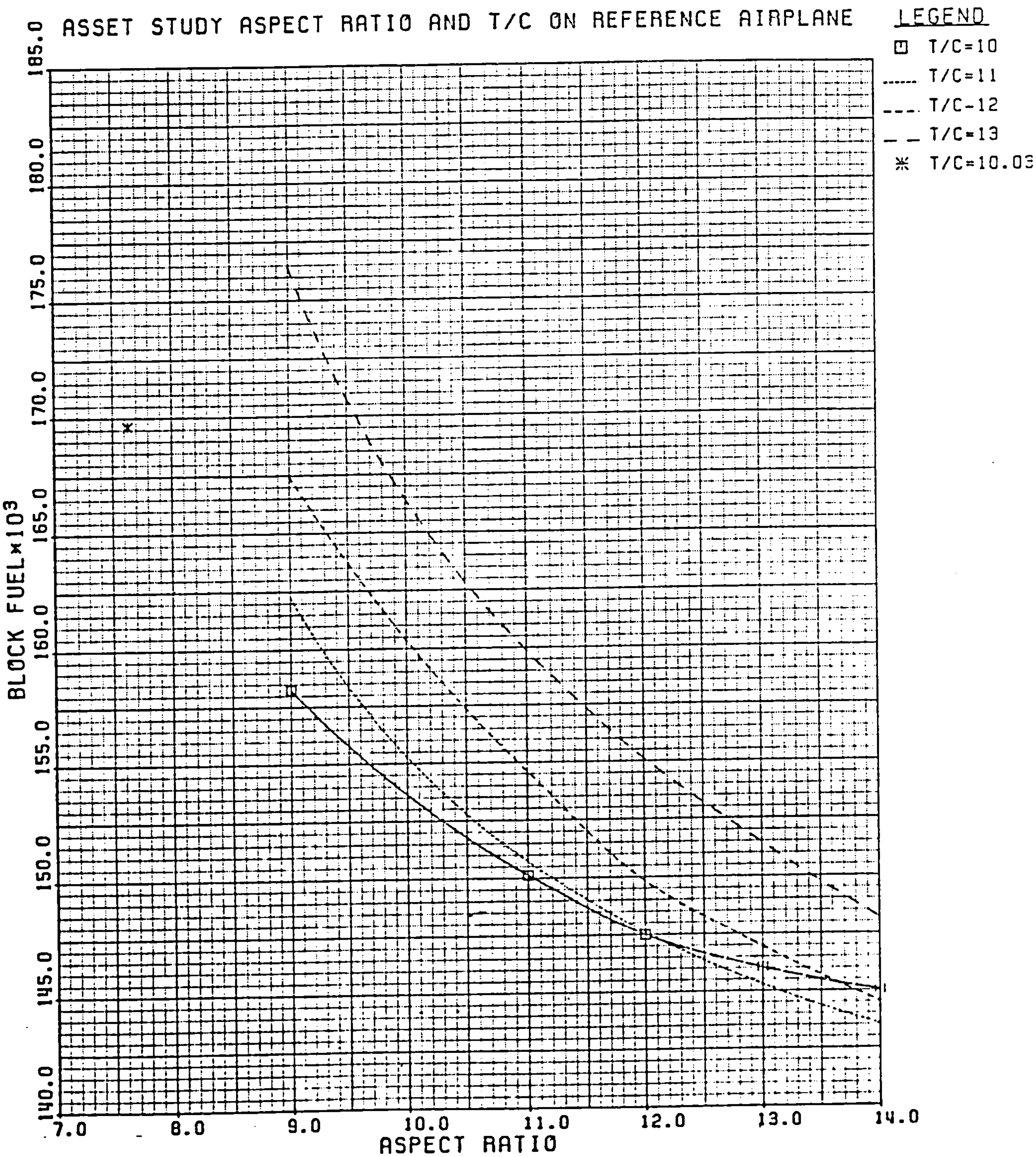


FIGURE 7.0-1 WING PARAMETER VARIATION
FOR MODIFIED WEIGHT EQUATION

Payload	53550 lb.
Range	4780 n.mi.
W/S	140.9 lb/ft^2
T/W	0.301
Sweep	35 Degrees
Cruise Mach	0.83
Aerodynamics:	L-100 (Wing 37B)
Systems:	L-1011 Active Controls
Materials:	L-1011
Propulsion:	RB211-524B4

Constraints for AR12=14 and t/c=11:

All-engine takeoff distance	=	7592 - 7918 ft (84 deg.F SL) @ Design Range
Engine-out takeoff distance	=	7457 - 7850 ft (84 deg.F SL) @ Design Range
Approach speed	=	149.5 - 150.6 kt. (TAS)

TABLE 7.0-1 BASELINE MODEL PARAMETRIC ANALYSES AIRCRAFT CHARACTERISTICS

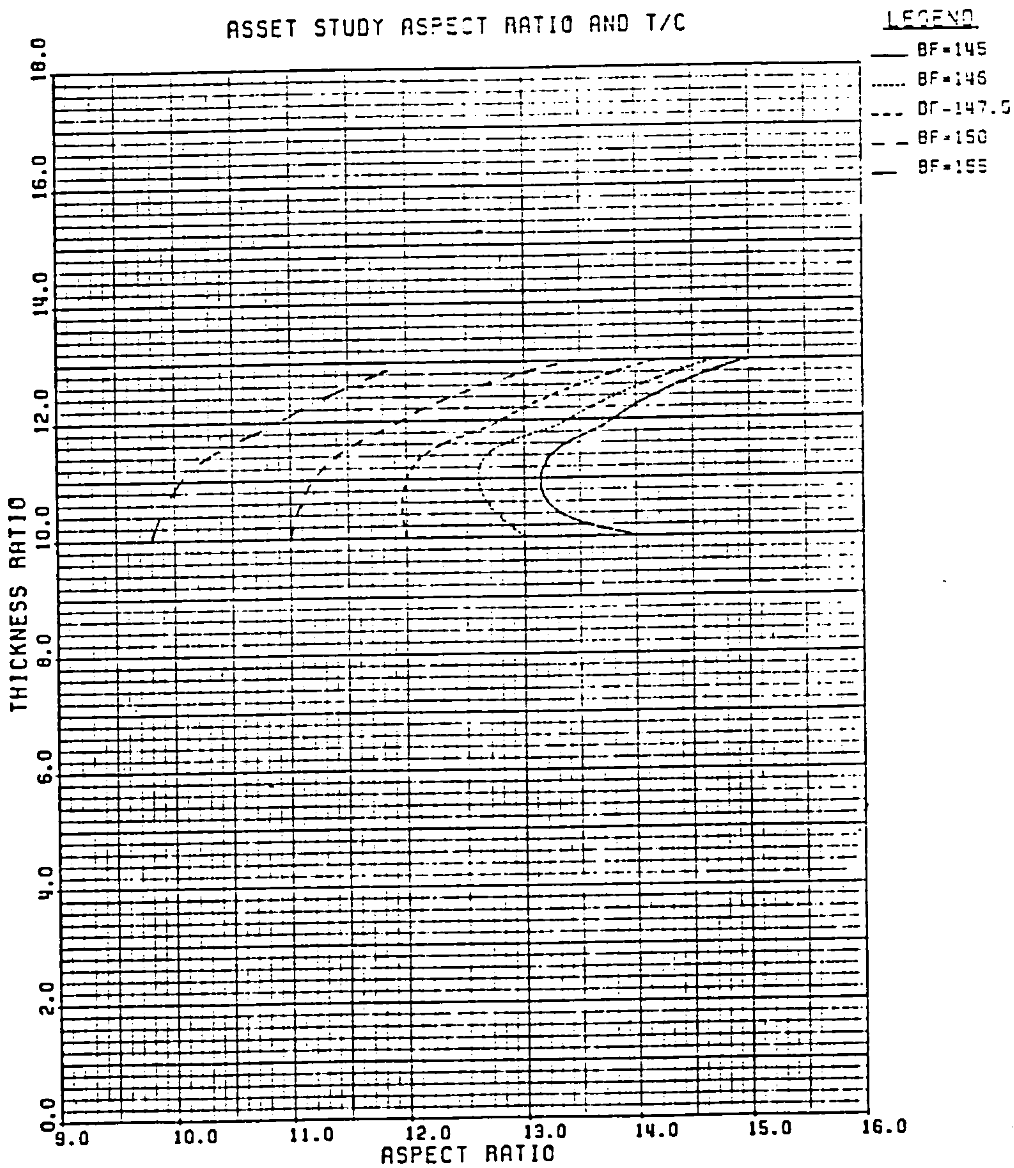


FIGURE 7.0-2 BLOCK FUEL (LB) KNOTHOLE
FOR MODIFIED WEIGHT EQUATION

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S NOVEMBER 26 1984

AIRCRAFT MODEL --L-1011-3
I.O.C. DATE --1980
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 140000
SLS SCALE 1.0 = 50000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 35.00 DEG
WING TAPER RATIO = 0.259

1 W/S	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9	140.9
2 T/W	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301	0.301
3 AR	9.00	11.00	12.00	13.00	14.00	9.00	11.00	12.00	13.00	14.00	9.00	11.00	12.00	13.00	14.00	9.00
4 T/C	10.00	10.00	10.00	10.00	10.00	11.00	11.00	11.00	11.00	11.00	12.00	12.00	12.00	12.00	12.00	13.00
5 SWEEP	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00	35.00
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778	4778
12 GROSS WEIGHT	496531	497107	499458	503706	508709	496658	490908	491912	494052	497132	499293	490741	488792	489651	491127	507851
13 FUEL WEIGHT	186107	177720	175068	173747	172926	190399	178148	174954	172598	170913	196049	182249	177215	174273	171799	205818
14 OP. WT. EMPTY	256874	265837	270840	276409	282233	252709	259210	263408	267904	272669	249694	254941	258027	261828	265778	248483
15 ZERO FUEL WT.	310424	319387	324390	329959	335783	306259	312760	316958	321454	326219	303244	308491	311577	315378	319328	302033
16 ENGINE SCALE	0.996	0.998	1.002	1.011	1.021	0.997	0.985	0.987	0.991	0.998	1.002	0.985	0.981	0.983	0.986	1.019
17 THRUST/ENGINE	49819	49876	50112	50538	51040	49831	49254	49355	49570	49879	50096	49238	49042	49128	49276	50954
18 WING AREA	3524.	3528.	3545.	3575.	3610.	3525.	3484.	3491.	3506.	3528.	3544.	3483.	3469.	3475.	3486.	3604.
19 WING SPAN	178.1	197.0	206.2	215.6	224.8	178.1	195.8	204.7	213.5	222.3	178.6	195.7	204.0	212.5	220.9	180.1
20 H. TAIL AREA	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0	1282.0
21 V. TAIL AREA	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0	550.0
22 ENG. LENGTH	9.91	9.92	9.97	10.06	10.17	9.91	9.79	9.81	9.86	9.92	9.97	9.79	9.75	9.76	9.80	10.15
23 ENG. DIAMETER	7.14	7.14	7.16	7.19	7.22	7.14	7.10	7.10	7.12	7.14	7.16	7.10	7.08	7.09	7.10	7.22
24 BODY LENGTH	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2	164.2
25 WING FUEL LIMIT	1.192	1.141	1.120	1.102	1.085	1.282	1.228	1.205	1.184	1.166	1.370	1.309	1.285	1.262	1.242	1.451
COST DATA																
26 RDTE - BIL.	2.514	2.601	2.650	2.704	2.760	2.474	2.537	2.578	2.622	2.668	2.444	2.496	2.526	2.563	2.602	2.431
27 FLYAWAY - MIL.	57.22	58.62	59.43	60.35	61.31	56.61	57.55	58.22	58.95	59.73	56.21	56.92	57.38	57.99	58.62	56.16
28 INVESTMNT-BIL.	20.093	20.566	20.841	21.154	21.485	19.888	20.199	20.425	20.673	20.939	19.754	19.987	20.139	20.344	20.560	19.748
29 DOC - C/SM	4.201	4.120	4.101	4.101	4.110	4.246	4.104	4.075	4.058	4.051	4.314	4.146	4.088	4.061	4.040	4.442
30 IOC - C/SM	2.371	2.370	2.371	2.373	2.377	2.373	2.366	2.365	2.366	2.368	2.378	2.367	2.364	2.364	2.364	2.389
31 ROI A.T. - 0/0	-28.42	-26.56	-25.95	-25.60	-25.35	-29.39	-26.75	-26.03	-25.49	-25.09	-30.62	-27.67	-26.57	-25.90	-25.34	-32.66
MISSION PARAMETERS																
32 MISN V1(1,1)	31000	35000	35000	35000	35000	31000	35000	35000	35000	35000	31000	35000	35000	35000	35000	31000
33 MISN V2(1,1)	158248	150130	147491	146029	145032	162259	150713	147602	145283	143533	167466	154608	149827	146948	144524	176435
CONSTRAINT OUTPUT																
34 TAKEOFF DST(1)	7916	7699	7632	7609	7592	7918	7704	7638	7616	7599	7918	7707	7641	7620	7604	7915
35 CLIMB GRAD(1)	0.1397	0.1498	0.1536	0.1567	0.1594	0.1396	0.1496	0.1534	0.1565	0.1592	0.1395	0.1495	0.1532	0.1563	0.1590	0.1395
36 TAKEOFF DST(2)	7843	7578	7501	7475	7457	7846	7578	7500	7472	7453	7850	7580	7501	7473	7453	7857
37 CLIMB GRAD(2)	0.0547	0.0649	0.0686	0.0717	0.0743	0.0546	0.0648	0.0686	0.0718	0.0744	0.0544	0.0647	0.0686	0.0717	0.0744	0.0542
38 CTOL LNDG D(1)	6887	6898	6910	6929	6949	6888	6875	6882	6894	6908	6897	6874	6871	6877	6886	6928
39 AP SPEED-KT(1)	149.9	150.0	150.1	150.3	150.6	149.9	149.6	149.7	149.8	150.0	150.1	149.6	149.5	149.5	149.6	150.5

TABLE 7.0-2 ASSET parametric analysis for modified weight equation

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S NOVEMBER 26 1984

AIRCRAFT MODEL --L-1011-3
I.O.C. DATE --1980
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 140000
SLS SCALE 1.0 = 50000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 35.00 DEG
WING TAPER RATIO = 0.259

1 W/S	140.9	140.9	140.9	140.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.301	0.301	0.301	0.301	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	11.00	12.00	13.00	14.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	13.00	13.00	13.00	13.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	35.00	35.00	35.00	35.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4778	4778	4778	4778	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	493387	491077	490380	490252	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	187948	182863	179027	175649	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	251889	254664	257803	261053	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	305439	308214	311353	314603	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	0.990	0.985	0.984	0.984	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	49503	49271	49201	49189	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3502.	3485.	3480.	3479.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	196.3	204.5	212.7	220.7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	1282.0	1282.0	1282.0	1282.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	550.0	550.0	550.0	550.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	9.84	9.79	9.78	9.78	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	7.11	7.10	7.09	7.09	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	164.2	164.2	164.2	164.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.386	1.359	1.334	1.313	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	2.466	2.494	2.524	2.556	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	56.51	56.92	57.41	57.91	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	19.852	19.987	20.150	20.321	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	4.214	4.154	4.112	4.077	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.372	2.368	2.366	2.365	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - 0/0	-28.89	-27.78	-26.92	-26.17	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	35000	35000	35000	35000	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	159874	155115	151434	148170	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
34 TAKEOFF DST(1)	7707	7642	7622	7607	0	0	0	0	0	0	0	0	0	0	0	0
35 CLIMB GRAD(1)	0.1494	0.1531	0.1562	0.1588	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
36 TAKEOFF DST(2)	7584	7505	7476	7455	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(2)	0.0646	0.0684	0.0716	0.0743	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 CTOL LNDG D(1)	6884	6879	6880	6882	0	0	0	0	0	0	0	0	0	0	0	0
39 AP SPEED-KT(1)	149.8	149.6	149.6	149.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

TABLE 7.0-2 (cont.) ASSET parametric analysis for modified weight equation

DASH 800, L 1011-3, [illegible] FAN, M = 0.83, RANGE=4770[illegible].INTERIM

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	476531.	0.	0.	0.	0.	0.0	0.0	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	476531.	923.	923.	0.	0.	1.0	1.0	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.373	475401.	2993.	3923.	18.	18.	4.0	5.0	0.	140201.	0.	0.663	17.51	0.558
ACCEL	10000.	0.425	472711.	853.	4673.	8.	26.	1.3	6.3	0.	140201.	0.	0.468	18.67	0.608
CLIMB	10000.	0.625	471857.	6923.	11522.	101.	127.	12.8	19.1	0.	140201.	0.	0.363	17.43	0.680
CRUISE	31000.	0.839	464939.	0.	11522.	0.	127.	0.0	19.1	0.	-140101.	0.	0.474	18.09	0.660
ACCEL	31000.	0.839	464939.	0.	11522.	0.	127.	0.0	19.1	0.	140201.	0.	0.474	18.09	0.680
ACCEL	31000.	0.839	464939.	0.	11522.	0.	127.	0.0	19.1	0.	140201.	0.	0.474	18.09	0.680
CRUISE	31000.	0.839	464939.	22974.	34455.	641.	768.	78.9	98.1	0.	-140101.	0.	0.463	18.03	0.663
CLIMB	31000.	0.830	442075.	1434.	35889.	29.	797.	3.6	101.7	0.	140201.	0.	0.500	17.95	0.677
CRUISE	35000.	0.830	440641.	78937.	114826.	2477.	3273.	310.5	412.1	0.	-140101.	0.	0.495	17.84	0.649
CLIMB	35000.	0.830	361654.	1261.	116087.	30.	3303.	3.8	415.9	0.	140201.	0.	0.501	17.73	0.676
CRUISE	39000.	0.830	360393.	36191.	152329.	1297.	4600.	163.4	579.3	0.	-140101.	0.	0.517	17.63	0.647
DESCENT	39000.	0.839	324202.	1574.	153902.	103.	4703.	13.8	593.1	0.	140301.	0.	0.329	15.71	-16.435
DECEL	10000.	0.638	322628.	171.	154073.	15.	4718.	1.8	594.9	0.	140301.	0.	0.335	16.89	-5.331
DESCENT	10000.	0.450	322457.	761.	154834.	32.	4750.	7.1	601.9	0.	140301.	0.	0.459	18.81	3.368
CRUISE	39000.	0.830	321696.	734.	155568.	23.	4773.	3.5	605.4	0.	-140101.	0.	0.487	17.68	0.682
LOITER	1500.	0.350	320962.	733.	156299.	0.	4773.	3.0	608.4	0.	-140101.	0.	0.562	18.76	0.805
CRUISE	1500.	0.378	320231.	502.	156800.	0.	4773.	2.0	610.4	0.	-140101.	0.	0.461	18.91	0.841
[illegible]	0.	0.0	339730.	0.	156800.	-4773.	0.	0.0	610.4	0.	0.	0.	0.0	0.0	0.0
CRUISE	39000.	0.830	339730.	12501.	169322.	0.	0.	61.0	671.5	0.	-140101.	0.	0.476	17.64	0.654
TAKEOFF POWER 1	0.	0.0	327143.	0.	169302.	0.	0.	0.0	671.5	0.	999501.	0.	0.0	0.0	0.822
POWER 2	0.	0.0	327148.	923.	170312.	0.	0.	1.0	672.5	0.	140402.	0.	0.0	0.0	0.392
CLIMB	0.	0.370	326218.	1274.	171815.	10.	10.	2.2	674.6	0.	140201.	0.	0.431	18.53	0.558
ACCEL	10000.	0.456	314542.	853.	172291.	2.	12.	0.3	675.0	0.	140201.	0.	0.361	17.53	0.586
CLIMB	10000.	0.457	323583.	3183.	172272.	42.	55.	6.1	681.1	0.	140201.	0.	0.308	16.34	0.621
CRUISE	30000.	0.760	321253.	1342.	176443.	47.	102.	6.3	687.4	0.	-140101.	0.	0.357	17.17	0.698
DESCENT	30000.	0.753	319313.	747.	177365.	63.	163.	9.4	696.8	0.	140301.	0.	0.303	16.23	13.483
DECEL	10000.	0.547	319143.	91.	177478.	5.	168.	0.9	697.7	0.	140301.	0.	0.359	17.49	5.990
DESCENT	10000.	0.453	319052.	721.	178212.	32.	200.	7.0	704.7	0.	140301.	0.	0.428	18.52	3.368
CRUISE	10000.	0.753	318293.	1.	178212.	0.	200.	0.0	704.7	0.	-140101.	0.	0.355	17.12	0.699
CRUISE	1500.	0.378	318297.	7372.	185165.	0.	200.	52.0	756.7	0.	-140101.	0.	0.445	18.64	0.876

MTO = 476530.0 FUEL A=169187.1 FUEL B=185105.5

TABLE 7.0-3 Asset mission summary for AR 9, T/C 10 and modified weight equation

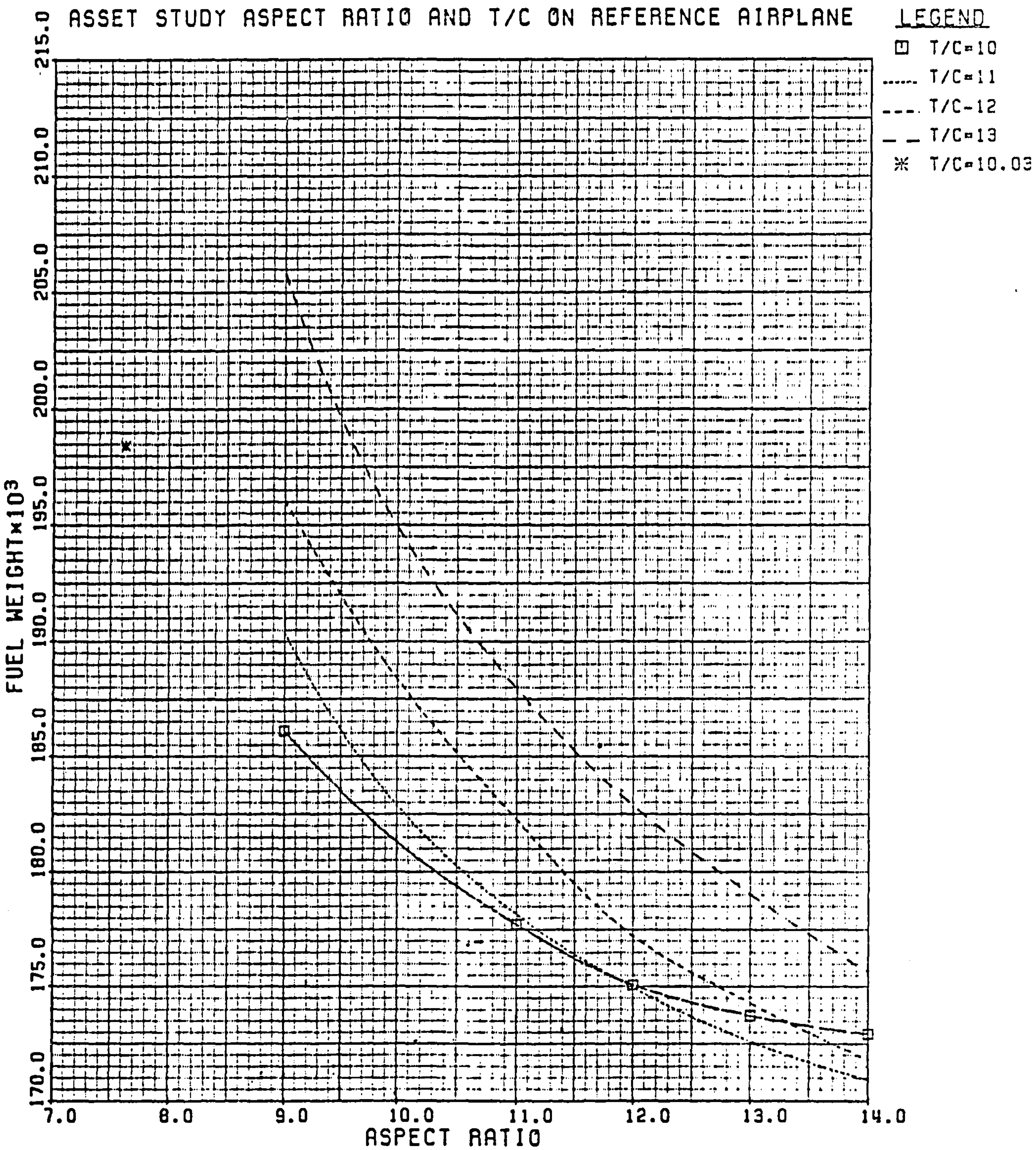


FIGURE 7.0-3 BLOCK FUEL WEIGHT (LB) VS ASPECT RATIO FOR MODIFIED WEIGHT EQUATION

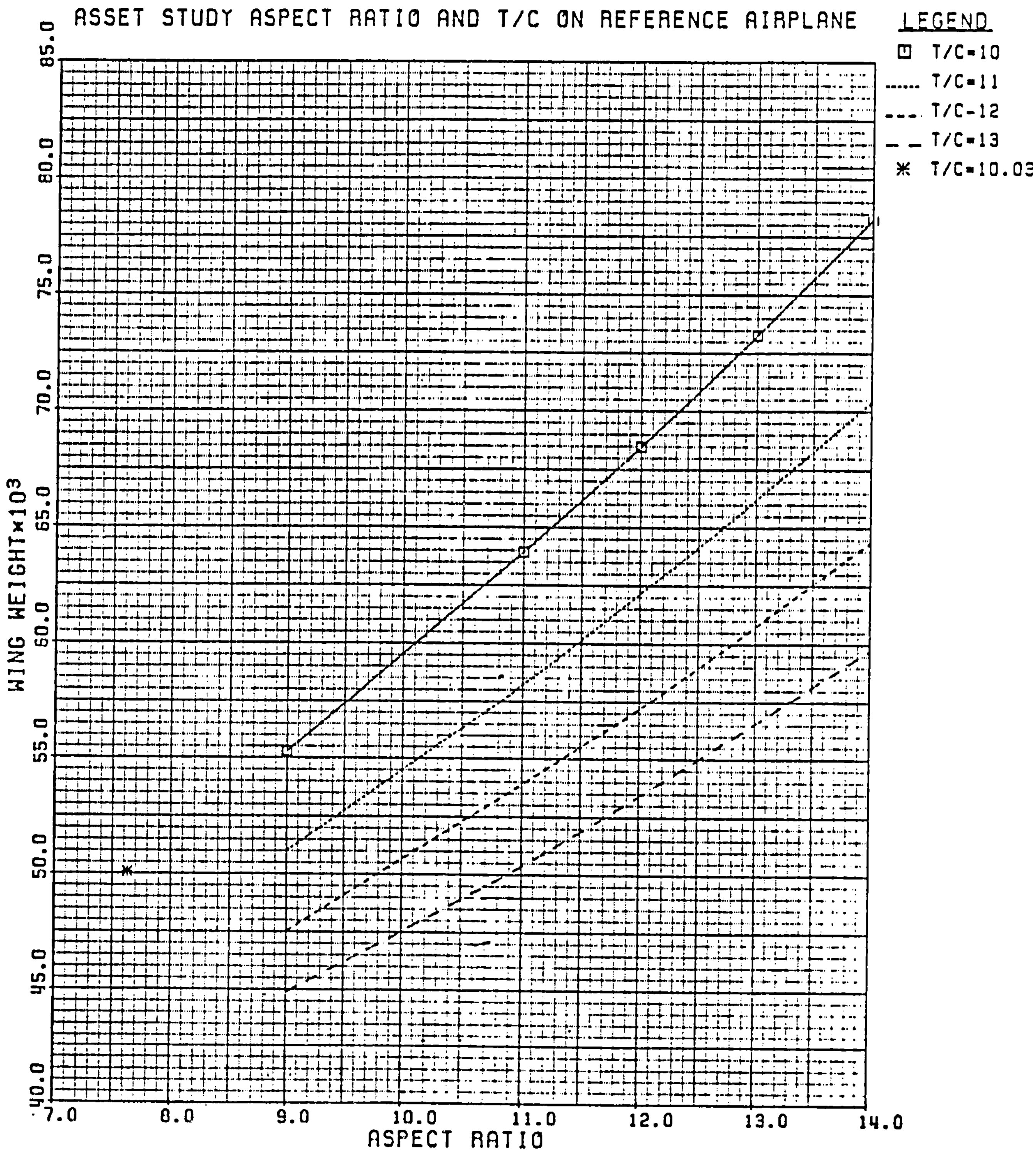


FIGURE 7.0-4 WING WEIGHT (LB) VS ASPECT RATIO FOR MODIFIED WEIGHT EQUATION

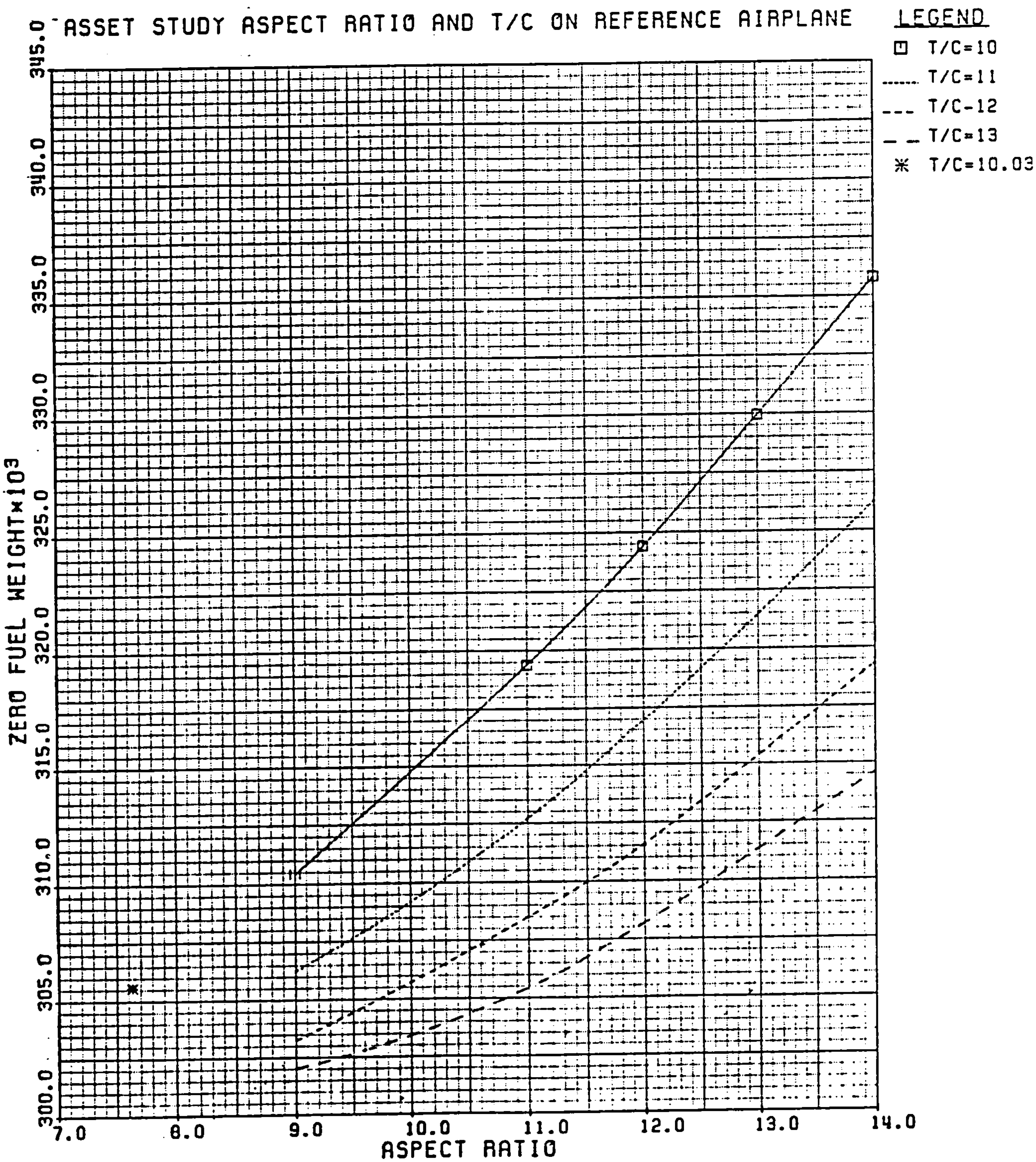


FIGURE 7.0-5 ZERO FUEL WEIGHT (LB) VS ASPECT RATIO FOR MODIFIED WEIGHT EQUATION

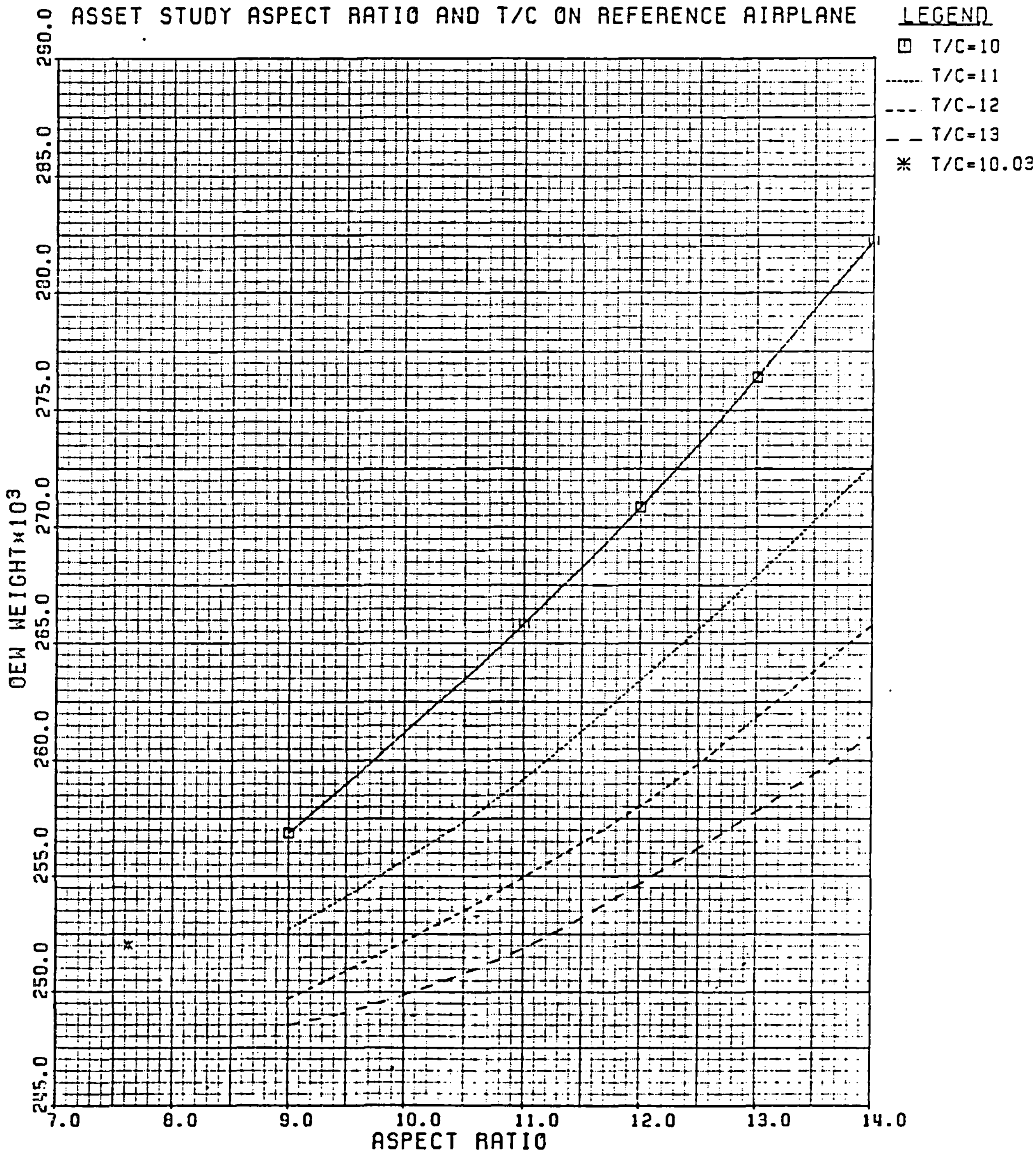


FIGURE 7.0-6 OEW WEIGHT VS ASPECT RATIO FOR MODIFIED WEIGHT EQUATION

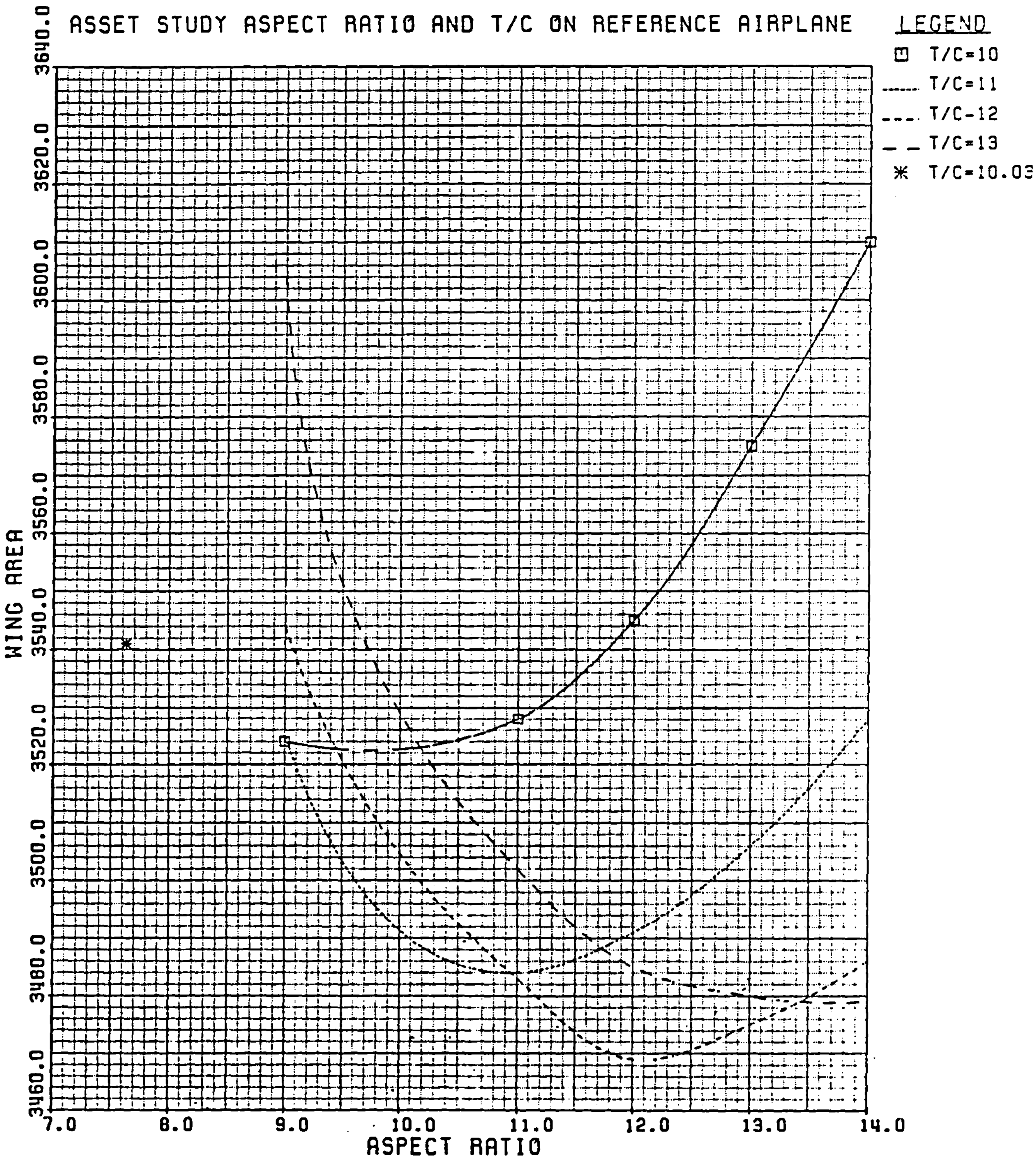


FIGURE 7.0-7 WING AREA (FT**2) VS ASPECT RATIO
FOR MODIFIED WEIGHT EQUATION

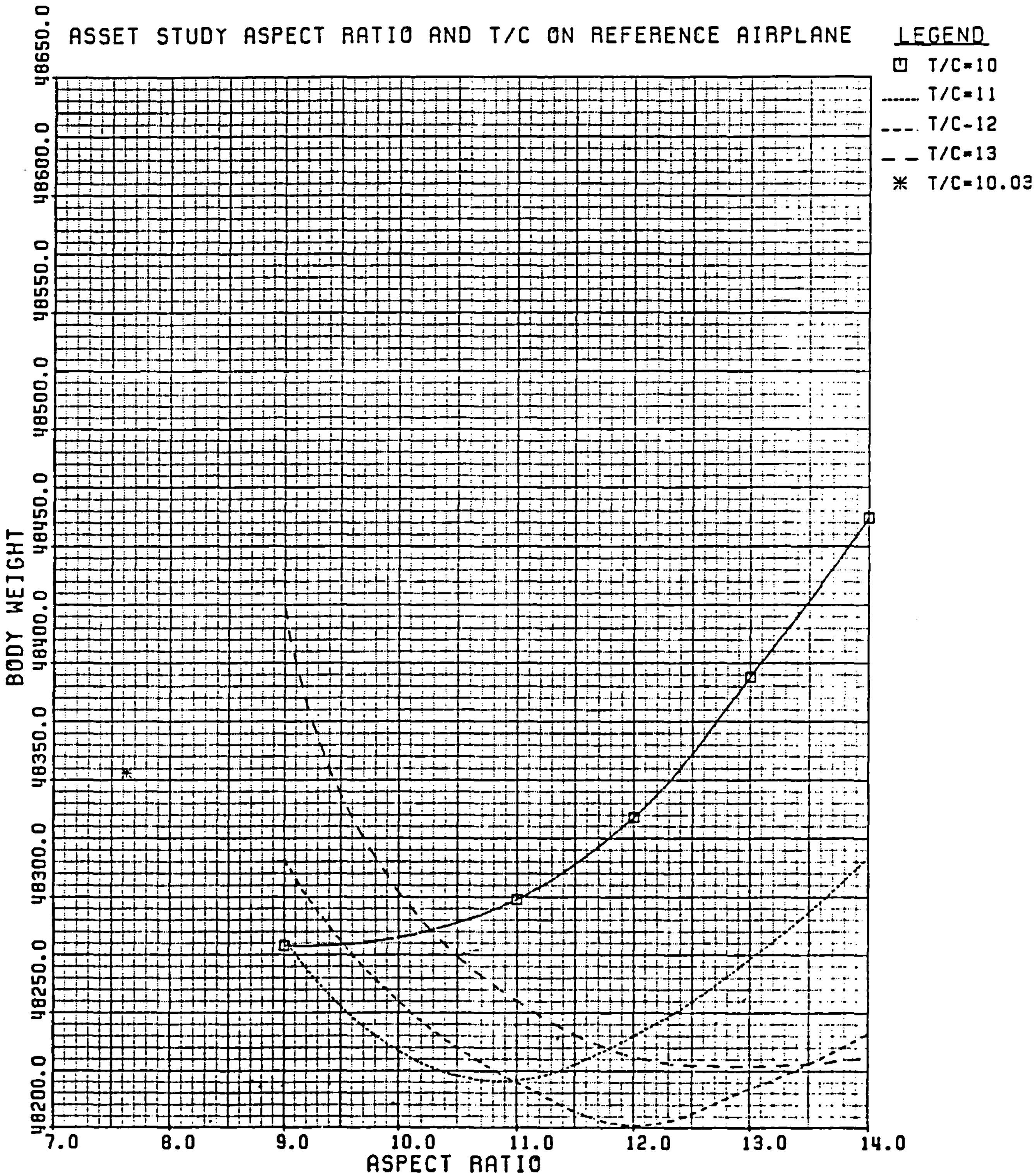


FIGURE 7.0-8 BODY WEIGHT (LB) VS ASPECT RATIO FOR MODIFIED WEIGHT EQUATION

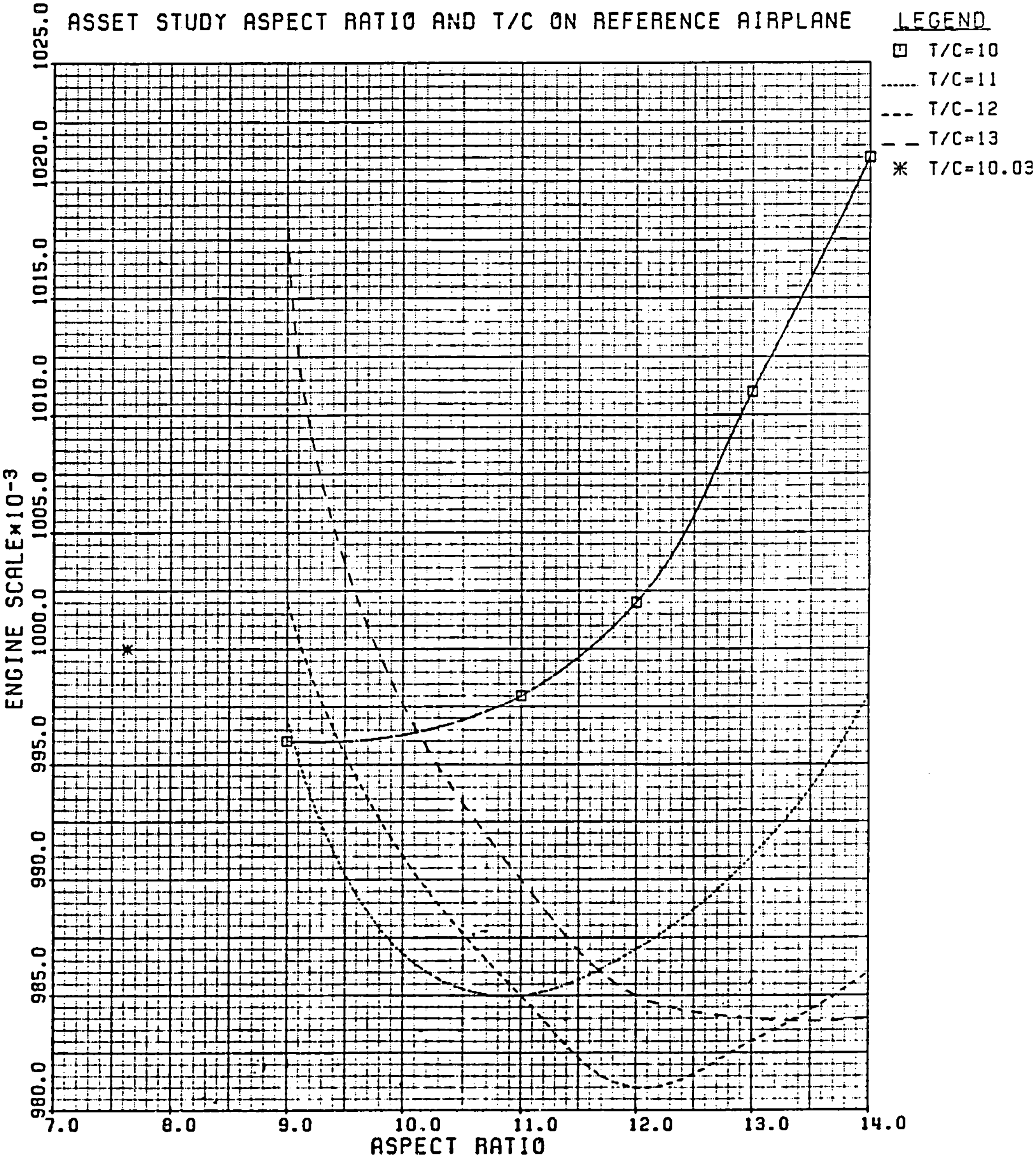


FIGURE 7.0-9 ENGINE SCALE VS ASPECT RATIO
FOR MODIFIED WEIGHT EQUATION

TABLE 7.0-4 PADS/ASSET DESIGN SUMMARY SHEET

DESIGN PARAMETER (1)	POINT DESIGN						ASSET OPTIMAL DESIGN	
	BASELINE ASSET	BASELINE PADS	AR12S35 ASSET	AR12S35 PADS	AR12S25 ASSET	AR12S25 PADS	INITIAL WEIGHT	WITH PADS WT. UPDATE
ASPECT RATIO	7.64	7.64	12.0	12.0	12.0	12.0	12.0 (2)	14.0 (3)
SWEEP (DEGREES)	35	35	35	35	25	25	35.0	35.0
TAPER RATIO	0.259	0.259	0.259	0.258	0.259	0.263	0.298	0.301
WING AREA (SQ. FT)	3541	3552	3541	3552	3541	3552	3650	3528
W/S (LB/SQ. FT)	142.3		148.0		146.9		142.3	140.9
T/W	0.298		0.286		0.288		0.298	0.301
T/C	10.03	10.03	10.03	10.03	10.03	10.03	11.0 (2)	11.0
RADIUS NAUTICAL MILES	4778		4749		4786		4780	4778
CRUISE MACH	0.83	0.83	0.83	0.83	0.76 (4)	0.83 (4)	0.83	0.83
GROSS WT. (1000 LB)	504.0		524.4		520.3			497.1
FUEL WT. (1000 LB)	198.4		188.8		188.1		184.5	170.9
OP. WT. EMPTY (1000 LB)	252.0		288.8		278.6			272.7
WING WEIGHT (1000 LB)	50.0	50.0	85.0	69.3	75.6	72.2	77.0	70.4
ADJUSTED FEM WT. (1000 LB)		36.8		56.1		59.0		
WING SECONDARY WT (1000 LB)		13.2		13.2		13.2		
PADS FEM WT. (1000 LB)		30.9 (5)		44.8 (6)		47.3 (7)		

NOTES:
(1) All weights are for both sides of the aircraft.
(2) Closest whole number for t/c was 11.0 where optimum was 10.75. Aspect ratio 12 was interpolated.
(3) At this aspect ratio the flutter speed will be below Vd. There will be a weight penalty (unknown at present) to bring the flutter speed to Vd.
(4) An ASSET analysis showed a severe drag rise for a cruise Mach 0.83 with a sweep of 25 degrees. Mach 0.76 was the optimum Mach number for block fuel and was used in the flight configuration definition. However, strength level loads were computed based on a cruise Mach number 0.83.
(5) From 2nd flex with margins sizing data set. (E750MPRMA8)
(6) From 2nd flex with margins sizing data set. (E873MPRMA3)
(7) From 3rd flex with margins sizing data set. (E907MPRMA7)

8.0 OBSERVATIONS

The development of a preliminary design tool like PADS received direction from many sources since its formal inception back in 1976. The first observation is the great amount of listening the principal designer of the system must do to permit the system to be as opened ended as economically possible. The architecture associated with database organization, configuration control, and design activity flow model are the principle building blocks for the system. If this architecture is inconsistent with the normal flow of design activity, the system will not serve the purpose for which it was created.

The PADS team required a high degree of design process visibility. Traditional outputs from print and plots were the minimum requirements from the automated design activity. In practice, the team received more visibility of the PADS design evolution than was possible by the traditional approach because of the increase in data blocks organization. This made possible operations on those data blocks using boolean and linear algebraic operators to form new gross data descriptors. Since the design process was compressed from months to four weeks, a major effort in reducing reliance on print output was necessary. Comprehensive graphics output is essential. The current practice is to put all print on microfiche with option to convert to hard print if necessary after reviewing the plots and microfiche. Paper print in the volumes required by the PADS team reached 30 percent of the total computer costs.

The key to using PADS is the ability to have planform changes reflected in the finite element model in a cost effective manner. It was found that the concept of a generic model made this possible. PADS without an automated finite element model generator would not be cost effective. In fact, all geometric bounded calculations such as fuel weight calculations must be able to adapt to planform changes cost effectively. This also includes the movement of the wing carry through structure in the fuselage to account for the changes in the aerodynamic center.

The use of a finite element model for preliminary design was based on the advice of the more senior staff members especially in structural methods group. Experience gained from the three PADS designs show that finite element models with detail on the scale of the PADS designs are necessary to discover unexpected trends. If the configuration is well understood, then simpler approaches will work. However, if questions like the effectiveness of active controls as a function of sweep are to be answered, then a detailed model is required. While the PADS synthesis model currently requires substantial resources in manhours and computer support, continued development of pre- and post-processors will reduce the manhours. More closely tailored accuracy requirements of the customer could reduce the computer resource requirements.

Included in the cost is the number of disciplines to be used in the design. It is highly recommended that all disciplines have processes available for inclusion in the aeroelastic design, no matter how expensive they appear to the customer.

However, the window to automated design application is proportional to the schedule and resources required to achieve sizing based on static loads.

One unexpected benefit of the automated design approach used in PADS is the documentation of the design path taken. The design process uses many tools. PADS was designed to be just one of those tools. However, the benefits of having all of the design processes contained within one of the PADS computer run setups soon motivated the users to put everything into PADS, even the most simple process. The benefits are the documentation of what was done and the simplicity of rerunning some part of the design. It is a matter on one hand of submitting a 2000 computer runs without PADS and on the other hand of submitting 20 runs with PADS. It is a matter of waiting to the last day to submit a major design effort to the computer or having to start submitting to the computer weeks before most of the decisions are firm because of the sheer volume of computer runs to be submitted to the computer.

The sizing of structural elements such as wing covers is only as good as the process which properly accounts for the many structural failure modes. It is the proper marriage of the finite element analysis program with an element sizing scheme that is the core of an acceptable and successful procedure. The PADS approach with PSASA satisfied both issues of correctness and reasonable cost.

The overall convergence of sizing elements to their final values was found to be more rapid than at first expected. The designs were less sensitive to the distribution of external loads than the mechanism of aeroelastic relief.

The major connectivity between disciplines, each with its own coordinate system(s), is a formal way to represent those systems and to have a computer program generate the transformations required to move data from one coordinate system to another.

One of the steps that the ASSET team goes through before engaging in design synthesis work is to generate a known design in order to validate the known input data and special computing routines. The same holds true for PADS activity. The PADS team as a whole is strong on this point, especially for any new generic structural finite element model. One of the major outputs of the PADS design is the formation of a process that converts finite element weights into hardware weights. This was made possible by evaluating a known design and extracting the appropriate factors based on distributive finite element weights.

Derivative capabilities in PADS are limited to those required for structural design to satisfy flutter constraints. A significant tool in PADS would have been a general derivative module to provide design sensitivities for a completed new design. This would have provided a powerful database to help understand some of the results that did not fit our expectations.

9.0 CONCLUSIONS AND RECOMMENDATIONS

The Lockheed inhouse study produced a database for structural optimization methodology investigations. The Baseline airplane wing designed in PADS was based on the L-1011-500 ACS design criteria. Since the wing covers designed in PADS came within 5% of the ideal cover weights of the actual airplane, the database and associated design criteria, therefore, should be a good reference for realistic design exercises.

Wing opimization studies indicated close agreement with the inhouse statistical weight equation for aspect ratio 12 at 25 degress of sweep. The studies showed a significant change in wing weight from the statistical methodology for aspect ratio 12 at 35 degrees of sweep.

The PADS analytically derived weights represent a significant departure from statistical weight estimating relationships. The results warrant further investigation of high aspect ratio configurations. Weight increments due to active controls and stiffness considerations are important for determining the weight of such designs. Detail analysis of these increments was not previously practicable in preliminary design.

As expected, the airplane configuration for minimum block fuel went from aspect ratio 12 to more than 14 when the wing weight model was adjusted for the lower weights indicated by PADS with wing loading, payload, and range held constant.

The studies indicated that a detailed finite element model could be used cost effectively in the study of active control effectiveness as a function of planform configuration variables.

Flutter results show that flutter constraints are necessary in high aspect ratio wing designs.

Gust loads for the designs in this study are a second order effect on the design weight.

The study indicated that a derivative module could be effectively used to identify design parameters which are most likely to affect desired design changes. This could be another modules in the PADS system of processes.

It is recommended that the study in wing optimization methodology be continued. The results to date show promise of increased understanding of high aspect ratio wing design characteristics. It is recommended that PADS be used to generate additional point designs to resolve questions concerning the current results.

10.0 REFERENCES

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8 Lynch,F. T.,"Commercial Transports-Aerodynamic Design for Cruise Performance Efficiency", Chapter II of Transonic Aerodynamics, edited by David Nixon, Vol. 81 of Progress in Astronautics and Aeronautics, edited by Martin Summerfield, Published by AIAA, 1982.

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APPENDIX A

PADS RUN SUMMARY

The design process under the PADS operating system consists of executing commands and supercommands to complete desired computing tasks. Commands and supercommands complete processes such as vibration and flutter analyses in a modular way. Commands and supercommands involk further lower level functional blocks to do operations such as data storage and retrieval. Typical PADS submittals may execute 10 or more Commands and/or Supercommands in a single run. The ability to concatenate processes which must run sequentially allows increased productivity from the engineer.

Computer setups to complete a computing task such as the formation of the net external loads matrices or a complete pass through the sizing process are normally a single dataset. Storage of these computer setups allows for easy recreation of data and the ability to make design pertabations with relative ease.

PADS run datasets which were used in the design process are cataloged in PANVALET under a designated naming convention. PANVALET names can have up to 10 characters. The first four characters designate the reference file number (RF) which is associated with the design. The first character of a reference file number is replace with the corresponding ith letter of the alphabet (ie., 1=A, 2=B, ect.) All PADS run datasets stored in PANVALET have 'RUN' for the 4th through 6th character and a three digit run number for the 7th through 10th character. Following is a set of tables which summarize the PADS runs for the Baseline (RF 's 5580, 5699, and 5750), AR12, 35 Degree Sweep (RF 5873), and AR12, 25 Degree Sweep (RF 5907) designs. Tables A1 through A4 are for the Baseline design, tables A5 through A10 are for the AR12, 35 Degree Sweep design, and tables A11 through A14 are for the AR12, 25 Degree Sweep design. Tables A1 through A14 contain descriptions of the modules run for each PADS submittal, the loads created, and the property and structural sizing allowable decks that were created during the PADS design exercise.

Table A15 provides a definition and function description of the modules used in tables A1 through A14.

Table A16 is a cost accounting summary for the major modules in the PADS design cycle. Most of the modules which are run for a single design cycle are included. The appoximate CPU time is provided consistent with Lockheed's computer accounting proceedures.

RUN	SECT	MDM	VRX CONF MTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	FSD	GUST	FLUT1	FLUT4	FLUT5	FLUT6
015	100	X		X													
016	100,101														X	X	X
017	9001														X	X	
018	9001																X
019	301														X	X	X
021	9002				X												
022	9002														X	X	X
023	9003														X	X	X
024	9002																X
025	100				X												
026	100			X													
27A	100	X	X	X													
029	100						X	X	X								
030	100								X	X							
031	100,200				X	X	X										
032	200							X	X	X	X						
033	200									X	X						
034	200,300				X	X	X	X	X								
035	300											X					
036	9300								X	X	X						
037	9300,9400				X	X	X	X	X								

TABLE A1 - BASELINE DESIGN PADS RUNS 015 - 037 UNDER RF5699 AND RF5750

RUN	SECT	MDM	VRX CONF WTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	FSD	GUST	FLUT1	FLUT4	FLUT5	FLUT6
038	9400								X	X	X						
039	320									X	X						
040	9410									X	X						
041	330											X					
042			PAN STORE FOR RUN040														
043			PAN STORE FOR RUN039														
047	100										X						
048	100									X	X						
049	100									X	X						
050	100									X							
052	100										X						
053	101										X						
054	400				X (K)												
055	400				X (WT)												
056	101			X													
057	400												X				
058	400												X				
059	9001				X (WT)												

TABLE A2 - BASELINE DESIGN PADS RUNS 038 - 059 UNDER RF5750

PADS RUN SUMMARY

RUN #	OUTPUT SECT #	DESCRIPTION	SECT # OF SIC	ACT ON OR OFF	SECT # OF LOADS MODIFIED
029	100	RIGID	-	ON	
032	200	1ST FLEX	100	ON	
034	300	2ND FLEX	200	ON	
036	9300	1ST ACT OFF		OFF	300
037	9400	ACT ON FROM SIC OFF	9300	ON	
038	9410	2ND ACT OFF		OFF	9400

TABLE A3 - LOADS CREATED FOR BASELINE DESIGN UNDER RF5750

RUN #	OUTPUT PROPS	OUTPUT PIPS	DESCRIPTION	INPUT PROPS	INPUT PIPS	LOAD SECT	MODULE
030	MPRWA1	MPIWA1	RIGID	MPRPW1	MPIPW2	100	RC
032	MPRWA0	MPIWA0	2ND PASS RD LDS	MPRWA1	MPIWA1	100	RC
033	MPRWA2	MPIWA2	1ST FLEX RC	MPRWA1	MPIWA1	200	RC
035	MPRWB1	MPIWB1	2ND FLEX FSD	MPRWA2	MPIWA2	300	FSD
036	MPRWA3	MPIWA3	1ST ACT OFF	MPRWA2	MPIWA2	9300	RC
038	MPRWA4	MPIWA4	2ND ACT OFF	MPRWA3	MPIWA3	9410	RC
039	MPRWA5	MPIWA5	2ND FLEX RC	MPRWA2	MPIWA2	300	RC
040	MPRWA6	MPIWA6	3RD ACT OFF	MPRWA4	MPIWA4	9410	RC
041	MPRWB2	MPIWB2	2ND-2ND FLEX FSD	MPRWA5	MPIWA5	300	FSD
047	MPRWA7	MPIWA7	1ST MARGIN PASS	MPRWA5	MPIWA5	300	RC
048	MPRWA8	MPIWA8	2ND MARGIN PASS	MPRWA7	MPIWA7	300	RC
049	MPRWA9	MPIWA9	PROD LOADS	MPRWA8	MPIWA8	6	RC
052	GPRWA1	GPIWA1	GUST ACS ON	MPRWA8	MPIWA8	100	RC
053	GPRWA2	GPIWA2	GUST ACS OFF	MPRWA8	MPIWA8	100	RC

TABLE A4 - PROPERTY AND ALLOWABLE DECKS CREATED FOR BASELINE DESIGN

RUN	SECT	MDL GEN	MDM	TNK WTS	VRX	CONF	LDS WTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	GUST	FLUT 1	FLUT 4	FLUT 5	FLUT 6
001	100	X																		
01A	100	X																		
002	100		X		X															
003	100							X												
004	100							X												
006	100			X																
007	100			X																
008	100			X																
009	100			X																
010	100			X																
011	100			X																
012	100		X																	
013	100					X	X													
014	100										X		X							
015	100													X						
17A	100											X	X							
19B	100														X					
020	101													X	X					

TABLE A5 - AR12 35 DEG DESIGN PADS RUNS 001 - 020 UNDER RF5873

PADS RUN SUMMARY

RUN	SECT	MDL GEN	MDM	TNK HTS	VRX	CONF	LDS HTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	GUST	FLUT 1	FLUT 4	FLUT 5	FLUT 6
022	100	X																		
023	100			X																
024	100			X																
025	101		X			X	X													
026	101										X	X	X							
26C	101												X							
027	99	X																		
028	101													X	X					
029	101														X					
030	200					X	X			X	X	X	X							
30A	200										X	X	X							
031	200													X	X					
032	100	X																		
32A	100	X																		
033	101								X											
034	100							X												
035	100		X																	
036	200,300								X	X	X	X	X							
36A	300											X	X							
037	300													X	X					
38A	300,400								X	X	X	X	X							
38B	400											X	X							
39A	400													X	X					

TABLE A6 - AR12 35 DEG DESIGN PADS RUNS 022 - 39A UNDER RF5873

RUN	SECT	MDL GEN	MDM	TNK MTS	VRX	CONF	LDS MTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	GUST	FLUT 1	FLUT 4	FLUT 5	FLUT 6
39B	400													X	X					
040	401												X							
041	401														X					
042	402												X							
043	402														X					
044	403									X	X	X	X							
045	403														X					
046	404									X	X	X	X							
047	404														X					
048	100																X			
049	100																X			
050	100																X			
051	100																X			
052	100							X												
053	100							X												
054	100							X												
055	300																	X	X	X
056	400															X				
057	401															X				
058	301																	X	X	X

TABLE A7 - AR12 35 DEG DESIGN PADS RUNS 39B - 058 UNDER RF5873

PADS RUN SUMMARY

RUN	SECT	MDL GEN	MDM	TNK WTS	VRX	CONF	LDS WTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA	GUST	FLUT 1	FLUT 4	FLUT 5	FLUT 6
059	402															X				
060	300														X					
061	301														X					
062	302														X					
065	300								X (WT)											
066	500								X (K)									X	X	X
067	401								X (DEL K FOR BFOP)											
068	999								X (DEL K FOR BFOP)											
069	401																	X	X	X

TABLE A8 - AR12 35 DEG DESIGN PADS RUNS 059 - 069 UNDER RF5873

PADS RUN SUMMARY

RUN #	OUTPUT SECT #	DESCRIPTION	SECT # OF SIC	ACS ON OR OFF	ACS GAIN	SECT # OF LOADS MODIFIED
26C	101	RIGID	-	ON	11.33	
30A	200	1ST FLEX	100	ON	11.33	
36A	300	2ND FLEX	200	ON	11.33	
38B	400	15. DEG/G	300	ON	15.0	
040	401	2G ON .4G OFF	-	ON	15.0	400
042	402	2.5 G ACS OFF		OFF		400
044	403	20. DEG/G	300	ON	20.0	
046	404	30. DEG/G	300	ON	30.0	
056	400*	0.0 DEG/G GUST LOADS	300	ON	0.0	
057	401*	11.3 DEG/G GUST LOADS	300	ON	11.3	
059	403*	15.0 DEG/G GUST LOADS	300	ON	15.0	

* GUST LOAD MATRICES HAVE DIFFERENT MATRIX NUMBERS THEN STATIC LOADS MATRICES. COMMON SECTION NUMBERS DON'T IMPLY SAME CONFIGURATION.

TABLE A9 - LOADS CREATED FOR AR12 35 DEG DESIGN UNDER RF5873

PADS RUN SUMMARY

RUN #	OUTPUT PROPS @	OUTPUT PIPS @	DESCRIPTION	INPUT PROPS@	INPUT PIPS @	LOAD SECT	MODULE
029	MPRWA1	MPIWA1	RIGID	MPRPW1	MPIPW1	101	PSASA
031	MPRWA2	MPIWA2	1ST FLEX	MPRWA1	MPIWA1	200	PSASA
037	MPRWA3	MPIWA3	2ND FLEX	MPRWA2	MPIWA2	300	PSASA
039	MPRWA4	MPIWA4	15. DEG/G	MPRWA3	MPIWA3	400	PSASA
041	MPRWA5	MPIWA45	2G ACS ON	MPRWA3	MPIWA3	401	PSASA
043	MPRWA6	MPIWA6	2.5 G OFF	MPRWA3	MPIWA3	402	PSASA
045	MPRWA7	MPIWA7	20. DEG/G	MPRWA3	MPIWA3	403	PSASA
047	MPRWA8	MPIWA8	30. DEG/G	MPRWA3	MPIWA3	404	PSASA
060	GPRWA1	GPIWA1	0.0 DEG/G GUST	MPRWA3	MPRWA3	400*	PSASA
061	GPRWA2	GPIWA2	11.3 DEG/G GUST	MPRWA3	MPRWA3	401*	PSASA
062	GPRWA3	GPIWA3	15.0 DEG/G GUST	MPRWA3	MPRWA3	403*	PSASA

* GUST LOADS WERE STACKED WITH SECT 300 STATIC LOADS FOR SIZING

@ - LAST 6 DIGITS OF PANVALET NAME WITH PREFIX E907.

TABLE A10 - PROPERTY AND ALLOWABLE DECKS CREATED FOR AR12 35 DEG DESIGN

PADS RUN SUMMARY

RUN	SECT	MDL GEN	MDM	TNK HTS	VRX	CONF	LDS HTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA
001	100	X													
01A	100	X													
002	100	X													
003	100		X												
005	100			X											
006	100			X											
007	100			X											
008	100			X											
009	100			X											
010	100			X											
011	100				X										
012	100	X													
013	100							X							
014	100		X												
015	100					X	X								
016	100										X				
16A	100											X			
16B	100												X		
017	100													X	X
018	100,200								X	X	X				
18A	200											X			
18B	200												X		

TABLE A11 - AR12 25 DEG DESIGN PADS RUNS 001 - 18B UNDER RF5907

PADS RUN SUMMARY

RUN	SECT	MDL GEN	MDM	MIS	VRX	CONF	LDS MTS	GTP	RF 150	JIG	PSRL	GRDH	STK	ST. SOLN	PSASA
19A	200													X	X
020	200,300								X	X	X	X			
20A	300												X		
021	300													X	
21A	300														X
022	400								X	X	X	X			
22A	400												X		
023	400													X	
23A	400														X
024	401												X		
025	401														X
026	500								X	X	X	X	X		
027	500														X
028	600								X	X	X	X	X		
029	600														X
030	600								X						
031	100							X							

TABLE A12 - AR12 25 DEG DESIGN PADS RUNS 19A - 031 UNDER RF5907

PADS RUN SUMMARY

RUN #	OUTPUT SECT #	DESCRIPTION	SECT # OF SIC	ACS ON OR OFF	ACS GAIN	SECT # OF LOADS MODIFIED
16B	100	RIGID	-	ON	11.33	
18B	200	1ST FLEX	100	ON	11.33	
20A	300	2ND FLEX	200	ON	11.33	
22A	400	9. DEG/G	300	ON	9.0	
024	401	2.5G ACS OFF	-	OFF	-	400
026	500	18. DEG/G	300	ON	18.0	
028	600	3RD FLEX	300	ON	11.33	

TABLE A13 - LOADS CREATED FOR AR12 25 DEG DESIGN UNDER RF5907

PROPERTY AND PIP DECKS CREATED
FOR JOB 5907
(ALL HAVE STRESS MARGINS OF SAFETY INCLUDED)

RUN #	OUTPUT PROPS*	OUTPUT PIPS*	DESCRIPTION	INPUT PROPS*	INPUT PIPS*	LOAD SECT	MODULE
017	MPRHA1	MPIHA1	RIGID	MPRPH1	MPIPH1	100	PSASA
19A	MPRHA2	MPIHA2	1ST FLEX	MPRHA1	MPIHA1	200	PSASA
21A	MPRHA3	MPIHA3	2ND FLEX	MPRHA2	MPIHA2	300	PSASA
23A	MPRHA4	MPIHA4	9. DEG/G	MPRHA3	MPIHA3	400	PSASA
025	MPRHA5	MPIHA5	2.5 G OFF	MPRHA3	MPIHA3	401	PSASA
027	MPRHA6	MPIHA6	18. DEG/G	MPRHA3	MPIHA3	500	PSASA
029	MPRHA7	MPIHA7	3RD FLEX	MPRHA3	MPIHA3	600	PSASA

* - LAST 6 DIGITS OF PANVALET NAME WITH PREFIX E907.

TABLE A14 - PROPERTY AND ALLOWABLE DECKS CREATED FOR AR12 25 DEG DESIGN

TABLE A15 - PROGRAM NAME ABBREVIATIONS

ABBREV.	PROGRAM NAME	FUNCTION
MDL GEN	FEM MODEL GENERATOR	Creates finite element model
MDM	MASS DISTRIBUTION MODULE	Forms weight data
TNK WTS	TANK WEIGHTS MODULE	Forms tank weight distributions
VRX	VORLAX	Creates Static Loads aero
CONF	CONFIGURATION MODULE	Creates Static Loads configuration data
LDS WTS	STATIC LOADS WEIGHT MODULE	Processes weight for production of Static External Loads
GTP	GRID TRANSFORMATION MODULE	Produces grid transforms
RF150	NASTRAN RIGID FORMAT 150	SIC and stiffness generation
JIG	JIG SHAPE MODULE	Used to account for flexibility for Static Load generation
PSRL	PSRL MODULE	Computes balanced net external static loads
GDHL	GROUND HANDLING MODULE	Computes ground handling loads
STK	STACKING MODULE	Stacks net external load conditions
ST SOLN	NASTRAN STATIC SOLUTION	Computes FEM internal loads
PSASA	PANEL SIZING AND STESS ALLOWABLE	Computes wing panel sizing and allowables
GUST	GLP5K GUST MODULE	Compute gust internal loads for sizing
FLUT1	FLUT1/DOUBLET LATTICE	Gust & Flutter unsteady aero
FLUT4	FLUT4 MODULE	Transform FLUT1 aero
FLUT5	FLUT5/VIBRATION MODULE	Vibration analysis
FLUT6	FLUT6/FLUTTER MODULE	Flutter analysis

TABLE A16 - MODULE COST ACCOUNTING

MODULE	IBM 3081 CPU TIME
MDL GEN	10 MIN.
MDM (4 COND.)	3 MIN.
TNK WTS (4 COND.)	14 MIN.
VRX	5 MIN.
CONF AND LDS WTS	
GTP FOR:	
STATIC LOADS	6 MIN. 37 SEC.
FLUTTER	6 MIN. 30 SEC.
GUST	5 MIN. 30 SEC.
FUEL TANK WEIGHT	7 MIN. 30 SEC.
RF150 FOR:	
SIC/STIFFNESS	11 MIN.
FEM COVER WEIGHT	2 MIN. 25 SEC.
DEL K FOR WING CTR. SECT. AND ROOT TRIANGLE	8 MIN. 12 SEC.
JIG,PSRL,GDHL,STK	12 MIN. 30 SEC.
ST SOLN	6 MIN. 30 SEC.
PSASA	11 MIN. 27 SEC.
GUST (1 FLT. COND.)	11 MIN.
FLUT1 (9 RED. FREQ.)	81 MIN.
FLUT4	5 MIN. 30 SEC.
FLUT5 FOR:	
22 MODES	2 MIN. 10 SEC.
50 MODES	8 MIN. 40 SEC.
FLUT6 FOR:	
22 MODES	3 MIN. 38 SEC.
50 MODES	9 MIN. 45 SEC.

APPENDIX B

SIZING STRESS MARGINS OF SAFETY

Stress margins of safety were included in the sizing process for selected designs on each of the PADS models. These margins were derived from data taken from LR 28029 and LR 28030. The initial margin of safety data consisted of margins for both compression and tension loads at selected locations on the wing for a known aircraft similiar to the PADS baseline design. These margins were modified to reflect the fact that the ultimate allowable load used in the PADS design process was different than that of the initial data. The formula MS(PADS) = [(1 + MS(INITIAL))(0.845) - 1] was used in order to correct for this discrepancy. The minimum of the compression and tension margins for each location was selected. These minimum margins were then applied to wing panels of the PADS model by a transformation process. Areas where margins of safety data did not exist or where minimum skin thickness was the designing factor have zero for the margin of safety. The following margins of safety were used for the PADS sizing procedure:

TABLE B-1 SPECIFIED STRESS MARGINS OF SAFETY

Panel ID	Margin	Panel ID	Margin	Panel ID	Margin	Panel ID	Margin
10103	0.000	10104	0.000	10105	0.000	10106	0.000
10107	0.000	10108	0.000	10109	0.000	10110	0.000
10203	0.000	10204	0.000	10205	0.000	10206	0.000
10207	0.000	10208	0.000	10209	0.000	10210	0.000
10303	0.000	10304	0.000	10305	0.000	10306	0.000
10307	0.000	10308	0.000	10309	0.000	10310	0.000
10403	0.000	10404	0.000	10405	0.000	10406	0.000
10407	0.000	10408	0.000	10409	0.000	10410	0.000
10503	0.625	10504	0.000	10505	1.009	10506	0.000
10507	1.356	10508	0.000	10509	0.926	10510	0.000
10603	0.171	10604	0.483	10605	0.388	10606	0.366
10607	0.396	10608	0.391	10609	0.386	10610	0.774
10703	0.140	10704	0.229	10705	0.057	10706	0.240

SIZING STRESS MARGINS OF SAFETY

10707	0.126	10708	0.211	10709	0.126	10710	0.338
10803	0.203	10804	0.137	10805	0.120	10806	0.257
10807	0.065	10808	0.230	10809	0.148	10810	0.213
10903	0.220	10904	0.155	10905	0.112	10906	0.193
10907	0.279	10908	0.252	10909	0.214	10910	0.320
11003	0.230	11004	0.118	11005	0.109	11006	0.162
11007	0.093	11008	0.277	11009	0.047	11010	0.400
11103	0.211	11104	0.054	11105	0.076	11106	0.138
11107	0.073	11108	0.283	11109	0.024	11110	0.442
11203	0.246	11204	0.063	11205	0.159	11206	0.109
11207	0.123	11208	0.258	11209	0.098	11210	0.470
11303	0.123	11304	0.103	11305	0.060	11306	0.121
11307	0.055	11308	0.233	11309	0.072	11310	0.434
11403	0.151	11404	0.121	11405	0.089	11406	0.128
11407	0.095	11408	0.218	11409	0.135	11410	0.370
11503	0.117	11504	0.030	11505	0.046	11506	0.069
11507	0.077	11508	0.040	11509	0.168	11510	0.057
11603	0.156	11604	0.512	11605	0.079	11606	0.289
11607	0.094	11608	0.119	11609	0.155	11610	0.091
11703	0.158	11704	0.595	11705	0.089	11706	0.286
11707	0.084	11708	0.173	11709	0.113	11710	0.187
11803	0.134	11804	0.164	11805	0.127	11806	0.074
11807	0.069	11808	0.055	11809	0.062	11810	0.060
11903	0.166	11904	0.260	11905	0.142	11906	0.234
11907	0.093	11908	0.121	11909	0.098	11910	0.036
12003	0.168	12004	0.327	12005	0.126	12006	0.141
12007	0.166	12008	0.157	12009	0.369	12010	0.374
12103	0.077	12104	0.372	12105	0.041	12106	0.088
12107	0.085	12108	0.036	12109	0.000	12110	0.001
12203	0.122	12204	0.484	12205	0.084	12206	0.230
12207	0.056	12208	0.085	12209	0.010	12210	0.070
12303	0.134	12304	0.628	12305	0.061	12306	0.267
12307	0.059	12308	0.120	12309	0.013	12310	0.071
12403	0.243	12404	1.103	12405	0.035	12406	0.423
12407	0.084	12408	0.074	12409	0.089	12410	0.039
12503	0.134	12504	0.000	12505	0.069	12506	0.000
12507	0.063	12508	0.000	12509	0.155	12510	0.000
12603	0.120	12604	0.000	12605	0.090	12606	0.000
12607	0.070	12608	0.000	12609	0.146	12610	0.000
12703	0.104	12704	0.000	12705	0.094	12706	0.000
12707	0.069	12708	0.000	12709	0.160	12710	0.000

APPENDIX C

MASS DISTRIBUTION MODULE OUTPUT

Output from the Mass Distribution Module consists of printed output to provide component and total aircraft weights as well as the full inertia matrix for the weight panel points.

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

BODY

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	1	2997.9	278.0	58.7	210.6			
20.0110	2	5023.5	390.6	58.7	210.6			
20.0110	3	2355.3	460.6	58.7	210.6			
20.0110	4	2522.7	526.0	58.7	210.6			
20.0110	5	1600.3	577.0	58.7	210.6			
20.0110	6	1954.3	637.1	58.7	210.6			
20.0110	7	2135.2	712.0	58.7	210.6			
20.0110	8	2112.6	785.0	58.7	210.6			
20.0110	9	1942.4	845.7	58.7	210.6			
20.0110	10	2946.5	911.7	58.7	210.6			
20.0110	11	2751.4	995.2	58.7	210.6			
20.0110	12	1759.9	1077.4	58.7	210.6			
20.0110	13	1819.2	1156.0	58.7	210.6			
20.0110	14	2247.7	1225.9	58.7	210.6			
20.0110	15	3543.2	1307.1	58.7	210.6			
20.0110	16	2535.8	1390.4	58.7	210.6			
20.0110	17	1322.8	1451.1	58.7	210.6			
20.0110	18	2212.2	1513.6	58.7	210.6			
20.0110	19	1583.8	1577.6	58.7	210.6			
20.0110	20	2226.9	1634.5	58.7	210.6			
20.0110	21	3335.1	1695.0	58.7	210.6			
20.0110	22	5224.6	1765.5	58.7	210.6			
20.0110	23	2511.1	1859.1	58.7	210.6			
20.0110	24	2321.6	1945.5	58.7	210.6			
20.0110	25	71.5	2022.1	58.7	210.6			
	TOTAL	122257.2	1115.4	58.7	210.6	0.35551510 10	0.95658500 11	0.92947300 11

CENTER BOX STRUCTURE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	11	771.1	995.2	58.7	210.6			
20.0110	12	1176.9	1077.4	58.7	210.6			
20.0110	13	973.4	1156.0	58.7	210.6			
20.0110	14	349.8	1225.9	58.7	210.6			
	TOTAL	6540.2	1097.5	58.7	210.6	0.15632400 09	0.41013390 10	0.39675890 10

CENTER BOX SYSTEMS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	12	93.4	1077.4	58.7	210.6			
20.0110	13	95.4	1156.0	58.7	210.6			
20.0110	14	121.7	1225.9	58.7	210.6			
	TOTAL	620.9	1157.8	58.7	210.6	0.14841490 08	0.43254370 09	0.41984550 09

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

WING BOX STRUCTURE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	26	78.3	985.3	150.5	164.1			
10.0110	27	460.6	1029.7	150.5	164.1			
10.0110	28	460.6	1074.0	150.5	164.1			
10.0110	29	465.6	1118.4	150.5	164.1			
10.0110	30	467.7	1162.7	150.5	164.1			
10.0110	31	417.5	1207.1	150.5	164.1			
10.0110	32	28.1	1251.4	150.5	164.1			
10.0110	36	91.1	1035.4	217.5	169.8			
10.0110	37	400.7	1075.5	217.5	169.8			
10.0110	38	400.7	1115.5	217.5	169.8			
10.0110	39	404.7	1155.6	217.5	169.8			
10.0110	40	402.7	1195.6	217.5	169.8			
10.0110	41	402.7	1235.7	217.5	169.8			
10.0110	42	80.6	1275.7	217.5	169.8			
10.0110	47	393.9	1121.3	284.5	175.6			
10.0110	48	350.8	1157.0	284.5	175.6			
10.0110	49	350.8	1192.8	284.5	175.6			
10.0110	50	353.6	1228.5	284.5	175.6			
10.0110	51	355.6	1264.3	284.5	175.6			
10.0110	52	118.9	1300.0	284.5	175.6			
10.0110	57	354.7	1167.1	351.5	181.3			
10.0110	58	293.7	1198.5	351.5	101.3			
10.0110	59	293.7	1230.0	351.5	181.3			
10.0110	60	293.5	1261.4	351.5	181.3			
10.0110	61	278.1	1292.9	351.5	181.3			
10.0110	62	99.3	1324.3	351.5	181.3			
10.0120	67	208.3	1209.9	412.3	188.4			
10.0120	68	243.5	1238.3	412.3	188.4			
10.0120	69	243.5	1266.8	412.3	188.4			
10.0120	70	245.5	1295.2	412.3	188.4			
10.0120	71	244.6	1323.7	412.3	188.4			
10.0120	72	183.6	1352.2	412.3	188.4			
10.0120	77	203.2	1249.7	466.7	196.9			
10.0120	78	252.6	1276.5	466.7	196.9			
10.0120	79	252.6	1303.3	466.7	196.9			
10.0120	80	254.3	1330.0	466.7	196.9			
10.0120	81	254.3	1356.8	466.7	196.9			
10.0120	82	122.9	1383.6	466.7	196.9			
10.0120	87	105.1	1289.6	521.3	205.3			
10.0120	88	168.0	1314.7	521.3	205.3			
10.0120	89	168.9	1339.8	521.3	205.3			
10.0120	90	170.0	1364.8	521.3	205.3			
10.0120	91	170.0	1389.9	521.3	205.3			
10.0120	92	45.1	1415.0	521.3	205.3			
10.0120	97	99.3	1329.5	575.7	213.8			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

WING BOX STRUCTURE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0120	98	160.1	1352.9	575.7	213.8			
10.0120	99	159.2	1376.2	575.7	213.8			
10.0120	100	159.2	1399.6	575.7	213.8			
10.0120	101	158.1	1423.0	575.7	213.8			
10.0120	102	31.5	1446.4	575.7	213.8			
10.0120	107	64.7	1369.3	630.3	222.3			
10.0120	108	117.8	1391.0	630.3	222.3			
10.0120	109	117.8	1412.7	630.3	222.3			
10.0120	110	118.9	1434.4	630.3	222.3			
10.0120	111	114.1	1456.1	630.3	222.3			
10.0120	112	61.0	1477.8	630.3	222.3			
10.0120	113	25.5	1499.5	630.3	222.3			
10.0120	117	61.0	1409.2	684.8	230.7			
10.0120	118	112.1	1429.2	684.8	230.7			
10.0120	119	112.1	1449.2	684.8	230.7			
10.0120	120	111.0	1469.2	684.8	230.7			
10.0120	121	111.0	1489.2	684.8	230.7			
10.0120	122	23.6	1509.2	694.8	230.7			
10.0120	127	6.0	1449.1	739.3	239.2			
10.0120	128	92.2	1467.4	739.3	239.2			
10.0120	129	79.5	1485.7	739.3	239.2			
10.0120	130	79.5	1504.0	737.3	239.2			
10.0120	131	78.6	1522.3	739.3	239.2			
10.0120	132	13.6	1540.6	739.3	239.2			
10.0120	138	28.5	1505.6	793.8	247.6			
10.0120	139	69.8	1522.2	793.8	247.6			
10.0120	140	69.8	1538.8	793.8	247.6			
10.0120	141	70.7	1555.4	793.8	247.6			
10.0120	142	49.1	1572.1	793.8	247.6			
10.0120	148	71.8	1543.7	848.3	256.1			
10.0120	149	55.9	1558.7	848.3	256.1			
10.0120	150	55.9	1573.6	848.3	256.1			
10.0120	151	53.9	1588.5	848.3	256.1			
10.0120	152	31.8	1603.5	848.3	256.1			
10.0120	160	30.4	1608.4	902.8	264.6			
10.0120	161	38.3	1621.6	902.8	264.6			
10.0120	162	41.1	1634.9	902.8	264.6			
10.0120	172	45.1	1666.1	957.0	273.5			
	TOTAL	29364.9	1263.5	396.3	189.9	0.3286062D 10	0.2425244D 11	0.2646036D 11

WING BOX SYSTEMS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	26	147.3	985.3	150.5	164.1			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

WING BOX SYSTEMS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	27	29.5	1029.7	150.5	164.1			
10.0110	28	0.9	1074.0	150.5	164.1			
10.0110	30	15.6	1162.7	150.5	164.1			
10.0110	31	15.6	1207.1	150.5	164.1			
10.0110	32	53.9	1251.4	150.5	164.1			
10.0110	33	18.7	1295.8	150.5	164.1			
10.0110	34	53.9	1340.1	150.5	164.1			
10.0110	36	23.6	1035.4	217.5	169.8			
10.0110	37	46.3	1075.5	217.5	169.8			
10.0110	39	0.9	1155.6	217.5	169.8			
10.0110	40	9.9	1195.6	217.5	169.8			
10.0110	41	7.9	1235.7	217.5	169.8			
10.0110	42	458.6	1275.7	217.5	169.8			
10.0110	43	212.3	1315.8	217.5	169.8			
10.0110	44	29.5	1355.8	217.5	169.8			
10.0110	46	23.6	1085.5	284.5	175.6			
10.0110	47	37.5	1121.3	284.5	175.6			
10.0110	49	0.9	1192.8	284.5	175.6			
10.0110	50	4.0	1228.5	284.5	175.6			
10.0110	51	6.0	1264.3	284.5	175.6			
10.0110	52	437.0	1300.0	284.5	175.6			
10.0110	53	210.3	1335.8	284.5	175.6			
10.0110	54	29.5	1371.5	284.5	175.6			
10.0110	56	20.7	1135.6	351.5	181.3			
10.0110	57	47.1	1167.1	351.5	181.3			
10.0110	58	0.9	1198.5	351.5	181.3			
10.0110	59	0.9	1230.0	351.5	181.3			
10.0110	60	16.7	1261.4	351.5	181.3			
10.0110	61	16.7	1292.9	351.5	181.3			
10.0110	62	22.7	1324.3	351.5	181.3			
10.0110	63	12.8	1355.8	351.5	181.3			
10.0110	64	2.8	1387.2	351.5	181.3			
10.0120	66	55.1	1181.4	412.3	188.4			
10.0120	67	35.5	1209.9	412.3	188.4			
10.0120	68	2.8	1238.3	412.3	188.4			
10.0120	69	2.0	1266.8	412.3	188.4			
10.0120	70	6.0	1295.2	412.3	188.4			
10.0120	71	6.0	1323.7	412.3	188.4			
10.0120	72	25.5	1352.2	412.3	188.4			
10.0120	73	36.3	1380.6	412.3	188.4			
10.0120	74	9.9	1409.1	412.3	188.4			
10.0120	76	0.9	1223.0	466.7	196.9			
10.0120	77	27.5	1249.7	466.7	196.9			
10.0120	80	4.0	1330.0	466.7	196.9			
10.0120	81	4.8	1356.8	466.7	196.9			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

WING BOX SYSTEMS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0120	82	53.1	1383.6	466.7	196.9			
10.0120	83	123.7	1410.3	466.7	196.9			
10.0120	84	2.8	1437.1	466.7	196.9			
10.0120	86	6.8	1264.5	521.3	205.3			
10.0120	87	16.7	1289.6	521.3	205.3			
10.0120	88	4.0	1314.7	521.3	205.3			
10.0120	89	2.0	1339.8	521.3	205.3			
10.0120	90	4.0	1364.8	521.3	205.3			
10.0120	91	4.8	1389.9	521.3	205.3			
10.0120	92	26.4	1415.0	521.3	205.3			
10.0120	93	7.7	1440.1	521.3	205.3			
10.0120	94	4.0	1465.1	521.3	205.3			
10.0120	97	26.4	1329.5	575.7	213.8			
10.0120	100	4.8	1399.6	575.7	213.8			
10.0120	101	4.8	1423.0	575.7	213.8			
10.0120	102	25.5	1446.4	575.7	213.8			
10.0120	103	6.8	1469.8	575.7	213.8			
10.0120	104	2.8	1493.2	575.7	213.8			
10.0120	107	9.9	1369.3	630.3	222.3			
10.0120	110	6.0	1434.4	630.3	222.3			
10.0120	111	4.8	1456.1	630.3	222.3			
10.0120	112	53.1	1477.8	630.3	222.3			
10.0120	113	21.6	1499.5	630.3	222.3			
10.0120	114	2.8	1521.2	630.3	222.3			
10.0120	117	6.0	1409.2	684.8	230.7			
10.0120	120	4.8	1469.2	684.8	230.7			
10.0120	121	4.0	1489.2	684.8	230.7			
10.0120	122	49.1	1509.2	684.8	230.7			
10.0120	123	7.9	1529.2	684.8	230.7			
10.0120	124	4.0	1549.2	684.8	230.7			
10.0120	127	18.7	1449.1	739.3	239.2			
10.0120	128	6.8	1467.4	739.3	239.2			
10.0120	130	4.8	1504.0	739.3	239.2			
10.0120	131	4.0	1522.3	739.3	239.2			
10.0120	132	40.3	1540.6	739.3	239.2			
10.0120	133	4.8	1559.0	739.3	239.2			
10.0120	137	2.0	1488.9	793.8	247.6			
10.0120	138	6.8	1505.6	793.8	247.6			
10.0120	140	4.0	1538.8	793.8	247.6			
10.0120	141	2.8	1555.4	793.8	247.6			
10.0120	142	28.4	1572.1	793.8	247.6			
10.0120	143	28.4	1588.7	793.8	247.6			
10.0120	147	0.9	1528.8	848.3	256.1			
10.0120	148	23.6	1543.7	848.3	256.1			
10.0120	149	0.9	1558.7	848.3	256.1			

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

WING BOX SYSTEMS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0120	150	2.8	1573.6	848.3	256.1			
10.0120	151	9.9	1588.5	848.3	256.1			
10.0120	152	39.3	1603.5	848.3	256.1			
10.0120	153	8.8	1618.4	848.3	256.1			
10.0120	157	4.0	1568.7	902.8	264.6			
10.0120	158	22.7	1581.9	902.8	264.6			
10.0120	160	2.8	1608.4	902.8	264.6			
10.0120	161	10.8	1621.6	902.8	264.6			
10.0120	163	7.9	1648.1	902.8	264.6			
10.0120	165	7.9	1674.6	902.8	264.6			
	TOTAL	6049.8	1313.3	361.5	187.2	0.62373150D 09	0.53840080D 10	0.57914730D 10

LEADING EDGE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	26	92.5	985.3	150.5	164.1			
10.0110	36	98.2	1035.4	217.5	169.8			
10.0110	46	128.6	1085.5	284.5	175.6			
10.0110	56	83.2	1135.6	351.5	181.3			
10.0120	66	39.3	1181.4	412.3	188.4			
10.0120	67	68.7	1209.9	412.3	188.4			
10.0120	77	78.6	1249.7	466.7	196.9			
10.0120	87	78.6	1289.6	521.3	205.3			
10.0120	97	73.8	1329.5	575.7	213.8			
10.0120	107	68.7	1369.3	630.3	222.3			
10.0120	117	63.9	1409.2	684.8	230.7			
10.0120	127	72.7	1449.1	739.3	239.2			
10.0120	137	57.9	1488.9	793.8	247.6			
10.0120	147	37.5	1528.8	848.3	256.1			
10.0120	157	6.5	1568.7	902.8	264.6			
10.0120	158	55.1	1591.9	902.8	264.6			
10.0120	159	41.1	1595.2	902.8	264.6			
10.0120	160	9.9	1608.4	902.8	264.6			
10.0120	168	33.5	1619.9	957.0	273.5			
10.0120	169	35.3	1631.4	957.0	273.5			
10.0120	170	32.4	1643.0	957.0	273.5			
10.0120	171	31.5	1654.5	957.0	273.5			
	TOTAL	2574.5	1307.7	554.2	213.4	0.54352740D 09	0.23185040D 10	0.27412350D 10

TRAILING EDGE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	31	9.9	1207.1	150.5	164.1			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

TRAILING EDGE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	32	180.2	1251.4	150.5	164.1			
10.0110	33	54.5	1295.8	150.5	164.1			
10.0110	42	203.2	1275.7	217.5	169.8			
10.0110	43	55.9	1315.8	217.5	169.8			
10.0110	52	133.5	1300.0	284.5	175.6			
10.0110	53	39.2	1335.8	284.5	175.6			
10.0110	62	140.5	1324.3	351.5	181.3			
10.0110	63	51.9	1355.8	351.5	101.3			
10.0120	72	31.5	1352.2	412.3	188.4			
10.0120	73	84.6	1380.6	412.3	188.4			
10.0120	74	27.5	1409.1	412.3	188.4			
10.0120	82	36.3	1383.6	466.7	196.9			
10.0120	83	45.1	1410.3	466.7	196.9			
10.0120	84	17.0	1437.1	466.7	196.9			
10.0120	93	40.3	1440.1	521.3	205.3			
10.0120	103	35.5	1469.8	575.7	213.8			
10.0120	113	61.9	1499.5	630.3	222.3			
10.0120	123	33.4	1529.2	684.8	230.7			
10.0120	132	2.0	1540.6	739.3	239.2			
10.0120	133	50.2	1559.0	739.3	239.2			
10.0120	134	27.5	1577.3	739.3	239.2			
10.0120	142	6.8	1572.1	793.8	247.6			
10.0120	143	10.7	1538.7	793.8	247.6			
10.0120	152	10.5	1603.5	848.3	256.1			
10.0120	163	9.6	1648.1	902.8	264.6			
10.0120	173	12.8	1677.6	957.0	273.5			
	TOTAL	2844.1	1360.7	370.4	188.0	0.30133330 09	0.26993400 10	0.28982350 10

INBOARD AILERON

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0310	223	14.8	1380.6	412.3	188.4			
10.0310	224	14.8	1409.1	412.3	188.4			
10.0310	225	39.2	1437.5	412.3	188.4			
10.0310	233	14.8	1410.3	466.7	196.9			
10.0310	234	14.8	1437.1	466.7	196.9			
10.0310	235	35.5	1463.9	466.7	196.9			
	TOTAL	267.3	1432.0	438.7	192.5	0.30787400 08	0.27916180 09	0.30003370 09

OUTBOARD AILERON

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0320	443	14.8	1538.7	793.8	247.6			

MASS DISTRIBUTION MODULE OUTPUT

OUTBOARD AILERON

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0320	444	18.7	1605.3	793.8	247.6			
10.0320	445	18.7	1621.9	793.8	247.6			
10.0320	452	9.9	1603.5	848.3	256.1			
10.0320	453	20.7	1618.4	848.3	256.1			
10.0320	454	16.7	1633.3	848.3	256.1			
10.0320	455	16.7	1648.3	848.3	256.1			
10.0320	463	13.9	1648.1	902.8	264.6			
10.0320	464	18.7	1661.4	902.8	264.6			
10.0320	465	9.9	1674.6	902.8	264.6			
10.0320	473	2.8	1677.6	957.0	273.5			
10.0320	474	8.8	1689.2	957.0	273.5			
10.0320	475	15.6	1700.7	957.0	273.5			
	TOTAL	372.3	1639.0	861.3	258.2	0.1511174D 09	0.5127361D 09	0.6389982D 09

OUTBOARD FLAP

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0510	534	23.3	1437.1	466.7	196.9			
10.0510	535	25.5	1463.9	466.7	196.9			
10.0510	542	89.4	1415.0	521.3	205.3			
10.0510	543	83.5	1440.1	521.3	205.3			
10.0510	544	61.9	1465.1	521.3	205.3			
10.0510	545	51.1	1490.2	521.3	205.3			
10.0510	552	69.8	1446.4	575.7	213.8			
10.0510	553	78.6	1469.8	575.7	213.8			
10.0510	554	51.1	1493.2	575.7	213.8			
10.0510	555	46.3	1516.6	575.7	213.8			
10.0510	562	53.9	1477.8	630.3	222.3			
10.0510	563	55.1	1499.5	630.3	222.3			
10.0510	564	95.4	1521.2	630.3	222.3			
10.0510	565	85.6	1542.9	630.3	222.3			
10.0510	572	51.1	1509.2	684.8	230.7			
10.0510	573	65.8	1529.2	684.8	230.7			
10.0510	574	45.2	1549.2	684.8	230.7			
10.0510	575	43.1	1569.2	684.8	230.7			
10.0510	582	49.1	1540.6	739.3	239.2			
10.0510	583	31.5	1559.0	739.3	239.2			
10.0510	584	24.7	1577.3	739.3	239.2			
10.0510	585	49.1	1595.6	739.3	239.2			
	TOTAL	2478.1	1499.3	610.2	219.2	0.5284019D 09	0.2847583D 10	0.3256613D 10

INBOARD FLAP

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

10.0520	806	38.6	1295.8	150.5	164.1			
10.0520	807	185.6	1340.1	150.5	164.1			
10.0520	808	139.5	1384.5	150.5	164.1			
10.0520	816	68.7	1315.8	217.5	169.8			
10.0520	817	227.0	1355.8	217.5	169.8			
10.0520	818	186.7	1395.9	217.5	169.8			
10.0520	826	35.3	1335.8	284.5	175.6			
10.0520	827	168.0	1371.5	284.5	175.6			
10.0520	828	95.4	1407.3	284.5	175.6			
10.0520	836	78.6	1355.8	351.5	181.3			
10.0520	837	115.8	1387.2	351.5	181.3			
10.0520	838	78.6	1418.7	351.5	181.3			
10.0520	847	39.2	1409.1	412.3	188.4			
10.0520	848	39.2	1437.5	412.3	188.4			
	TOTAL	2992.3	1371.9	249.3	172.7	0.14700860 09	0.28621370 10	0.29198000 10

INBOARD SLAT

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0530	949	111.2	935.3	150.5	164.1			
10.0530	959	115.2	1035.4	217.5	169.8			
10.0530	969	115.2	1085.5	284.5	175.6			
10.0530	979	110.4	1135.6	351.5	181.3			
	TOTAL	904.2	1060.3	250.8	172.7	0.44434770 08	0.52314500 09	0.54058140 09

OUTBOARD SLAT

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0540	1139	81.4	1181.4	412.3	188.4			
10.0540	1149	114.9	1223.0	466.7	196.9			
10.0540	1159	106.1	1264.5	521.3	205.3			
10.0540	1169	99.3	1306.1	575.7	213.8			
10.0540	1179	84.6	1347.6	630.3	222.3			
10.0540	1189	77.5	1389.2	684.8	230.7			
10.0540	1199	63.9	1430.8	739.3	239.2			
10.0540	1209	57.9	1472.3	793.8	247.6			
10.0540	1219	51.1	1513.9	848.3	256.1			
10.0540	1220	13.6	1528.8	848.3	256.1			
10.0540	1229	47.1	1555.4	902.8	264.6			
10.0540	1230	40.6	1568.7	902.8	264.6			
10.0540	1239	30.4	1596.8	957.0	273.5			
10.0540	1240	40.3	1608.3	957.0	273.5			
	TOTAL	1817.4	1372.6	661.3	227.1	0.47221130 09	0.17760050 10	0.21531360 10

HORIZONTAL TAIL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

30.0110	629	82.3	1805.6	26.8	226.2
30.0110	630	114.8	1836.6	26.8	226.2
30.0110	631	112.6	1867.5	26.8	226.2
30.0110	632	145.1	1898.5	26.8	226.2
30.0110	633	17.3	1929.4	26.8	226.2
30.0110	636	24.9	1755.8	80.5	228.2
30.0110	637	66.0	1784.0	80.5	228.2
30.0110	638	129.9	1812.3	80.5	228.2
30.0110	639	94.2	1840.5	80.5	228.2
30.0110	640	109.4	1868.8	80.5	228.2
30.0110	641	123.4	1897.0	80.5	228.2
30.0110	642	95.3	1925.3	80.5	228.2
30.0110	643	62.3	1953.5	80.5	228.2
30.0110	644	40.0	1981.8	80.5	228.2
30.0110	645	24.9	2010.0	80.5	228.2
30.0110	646	24.9	1798.8	134.2	230.3
30.0110	647	55.3	1824.3	134.2	230.3
30.0110	648	87.7	1849.9	134.2	230.3
30.0110	649	90.9	1875.4	134.2	230.3
30.0110	650	85.5	1901.0	134.2	230.3
30.0110	651	102.9	1926.5	134.2	230.3
30.0110	652	39.0	1952.1	134.2	230.3
30.0110	653	57.4	1977.6	134.2	230.3
30.0110	654	28.1	2003.2	134.2	230.3
30.0110	655	18.4	2028.7	134.2	230.3
30.0110	656	17.3	1841.8	187.9	232.3
30.0110	657	52.0	1864.6	187.9	232.3
30.0110	658	60.6	1887.5	187.9	232.3
30.0110	659	59.5	1910.3	187.9	232.3
30.0110	660	54.1	1933.2	187.9	232.3
30.0110	661	46.5	1956.0	187.9	232.3
30.0110	662	67.1	1978.9	187.9	232.3
30.0110	663	55.2	2001.7	187.9	232.3
30.0110	664	22.7	2024.6	187.9	232.3
30.0110	665	15.2	2047.4	187.9	232.3
30.0110	666	23.8	1884.8	241.6	234.3
30.0110	667	42.2	1905.0	241.6	234.3
30.0110	668	39.0	1925.1	241.6	234.3
30.0110	669	33.6	1945.3	241.6	234.3
30.0110	670	50.9	1965.4	241.6	234.3
30.0110	671	77.9	1985.6	241.6	234.3
30.0110	672	108.3	2005.7	241.6	234.3
30.0110	673	48.7	2025.9	241.6	234.3
30.0110	674	30.3	2046.0	241.6	234.3
30.0110	675	11.9	2066.2	241.6	234.3
30.0110	676	14.1	1927.8	295.3	236.3
30.0110	677	31.4	1945.3	295.3	236.3
30.0110	678	41.1	1962.7	295.3	236.3
30.0110	679	39.0	1980.2	295.3	236.3

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

HORIZONTAL TAIL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
30.0110	680	37.9	1997.6	295.3	236.3			
30.0110	681	30.3	2015.1	295.3	236.3			
30.0110	682	75.8	2032.5	295.3	236.3			
30.0110	683	58.5	2050.0	295.3	236.3			
30.0110	684	28.1	2067.4	295.3	236.3			
30.0110	685	9.7	2084.9	295.3	236.3			
30.0110	686	9.7	1970.8	349.0	238.4			
30.0110	687	22.7	1985.6	349.0	238.4			
30.0110	688	28.1	2000.3	349.0	238.4			
30.0110	689	28.1	2015.1	349.0	238.4			
30.0110	690	28.1	2029.8	349.0	238.4			
30.0110	691	28.1	2044.6	349.0	238.4			
30.0110	692	56.3	2059.3	349.0	238.4			
30.0110	693	68.2	2074.1	349.0	238.4			
30.0110	694	14.1	2089.8	349.0	238.4			
30.0110	695	7.6	2103.6	349.0	238.4			
30.0110	696	3.8	2013.8	402.7	240.4			
30.0110	697	3.8	2025.9	402.7	240.4			
30.0110	698	29.2	2037.9	402.7	240.4			
30.0110	699	29.2	2050.0	402.7	240.4			
30.0110	700	27.6	2062.0	402.7	240.4			
30.0110	701	27.6	2074.1	402.7	240.4			
30.0110	702	33.0	2086.1	402.7	240.4			
30.0110	703	33.0	2098.2	402.7	240.4			
30.0110	704	7.6	2110.2	402.7	240.4			
30.0110	705	7.6	2122.3	402.7	240.4			
	TOTAL	7260.8	1936.7	177.5	231.9	0.35481630 09	0.13837810 11	0.13802050 11

VERTICAL TAIL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
40.0110	707	1.0	1801.6	0.0	347.3			
40.0110	710	1.0	1896.5	0.0	347.3			
40.0110	711	9.5	1928.1	0.0	347.3			
40.0110	712	10.5	1959.8	0.0	347.3			
40.0110	720	1.0	1919.6	0.0	391.8			
40.0110	721	5.3	1948.9	0.0	391.8			
40.0110	722	51.5	1978.3	0.0	391.8			
40.0110	723	20.0	2007.7	0.0	391.8			
40.0110	724	13.7	2037.1	0.0	391.8			
40.0110	725	9.5	2066.5	0.0	391.8			
40.0110	727	9.5	1861.2	0.0	436.3			
40.0110	728	14.7	1888.3	0.0	436.3			
40.0110	729	17.9	1915.5	0.0	436.3			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

VERTICAL TAIL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
40.0110	730	27.3	1942.6	0.0	436.3			
40.0110	731	26.3	1969.7	0.0	436.3			
40.0110	732	128.2	1996.9	0.0	436.3			
40.0110	733	15.8	2024.0	0.0	436.3			
40.0110	734	10.5	2051.2	0.0	436.3			
40.0110	735	7.4	2078.3	0.0	436.3			
40.0110	736	4.2	1866.1	0.0	480.8			
40.0110	737	21.0	1891.0	0.0	480.8			
40.0110	738	14.7	1915.9	0.0	480.8			
40.0110	739	14.7	1940.8	0.0	480.8			
40.0110	740	13.7	1965.7	0.0	480.8			
40.0110	741	16.8	1990.6	0.0	480.8			
40.0110	742	27.3	2015.4	0.0	480.8			
40.0110	743	14.7	2040.3	0.0	480.8			
40.0110	744	9.5	2065.2	0.0	480.8			
40.0110	745	6.3	2090.1	0.0	480.8			
40.0110	746	8.4	1893.2	0.0	525.3			
40.0110	747	20.0	1920.8	0.0	525.3			
40.0110	748	12.6	1943.4	0.0	525.3			
40.0110	749	12.6	1966.1	0.0	525.3			
40.0110	750	12.6	1988.7	0.0	525.3			
40.0110	751	12.6	2011.4	0.0	525.3			
40.0110	752	32.6	2034.0	0.0	525.3			
40.0110	753	25.2	2056.6	0.0	525.3			
40.0110	754	8.4	2079.3	0.0	525.3			
40.0110	755	4.2	2101.9	0.0	525.3			
40.0110	756	7.4	1930.2	0.0	569.8			
40.0110	757	16.8	1950.6	0.0	569.8			
40.0110	758	10.5	1971.0	0.0	569.8			
40.0110	759	10.5	1991.4	0.0	569.8			
40.0110	760	10.5	2011.8	0.0	569.8			
40.0110	761	10.5	2032.2	0.0	569.8			
40.0110	762	16.8	2052.6	0.0	569.8			
40.0110	763	11.6	2072.9	0.0	569.8			
40.0110	764	8.4	2093.3	0.0	569.8			
40.0110	765	2.1	2113.7	0.0	569.8			
40.0110	766	5.3	1962.3	0.0	614.3			
40.0110	767	13.7	1980.4	0.0	614.3			
40.0110	768	8.4	1998.6	0.0	614.3			
40.0110	769	8.4	2016.7	0.0	614.3			
40.0110	770	8.4	2034.8	0.0	614.3			
40.0110	771	9.5	2053.0	0.0	614.3			
40.0110	772	17.9	2071.1	0.0	614.3			
40.0110	773	9.5	2089.2	0.0	614.3			
40.0110	774	6.3	2107.4	0.0	614.3			

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

VERTICAL TAIL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
40.0110	775	2.1	2125.5	0.0	616.3			
40.0110	776	4.2	1994.3	0.0	653.8			
40.0110	777	13.7	2010.2	0.0	658.8			
40.0110	778	10.5	2026.1	0.0	658.8			
40.0110	779	10.5	2042.0	0.0	653.8			
40.0110	780	23.4	2057.9	0.0	653.8			
40.0110	781	23.4	2073.8	0.0	658.8			
40.0110	782	30.5	2089.7	0.0	658.8			
40.0110	783	8.4	2105.5	0.0	653.8			
40.0110	784	5.3	2121.4	0.0	658.8			
40.0110	785	2.1	2137.3	0.0	658.8			
	TOTAL	2017.1	2003.4	0.0	510.4	0.27074550 09	0.43219890 10	0.40512430 10

S-DUCT

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
	TOTAL	0.0	0.0	0.0	0.0	0.0	0.0	0.0

NOSE GEAR UP

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0520	792	0.1	495.2	0.0	132.8			
	TOTAL	0.2	495.2	0.0	132.8	0.99999990-04	0.99999990-04	0.99999990-04

MAIN GEAR UP

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0820	794	0.1	1277.0	112.2	150.2			
	TOTAL	0.2	1277.0	112.2	150.2	0.99999990-04	0.99999990-04	0.99999990-04

NOSE GEAR DOWN

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0510	793	906.0	529.1	0.0	72.0			
	TOTAL	1812.0	529.1	0.0	72.0	0.54770000 07	0.25910000 09	0.25370000 09

MAIN GEAR DOWN

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0810	795	7862.0	1280.6	213.0	68.0			

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MAIN GEAR DOWN

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
	TOTAL	15724.0	1280.6	213.0	68.0	0.40789990 09	0.12900000 11	0.13300000 11

ENGINES

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0610	797	14212.9	1038.4	415.1	120.6			
20.0330	795	6627.1	2029.0	0.0	264.4			
	TOTAL	41680.0	1387.5	283.1	166.3	0.31378300 10	0.44833560 11	0.46610180 11

PYLON

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0910	796	2230.0	1184.3	418.0	170.0			
	TOTAL	4460.0	1184.3	418.0	170.0	0.45399990 09	0.32000000 10	0.35300000 10

OPERATING EMPTY MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
252037.5	1234.9	182.7	192.2	0.19665330 11	0.52321310 11	0.68792340 11

PASSENGERS

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	3	1485.4	460.6	58.7	210.6			
20.0110	4	1334.7	526.0	58.7	210.6			
20.0110	5	861.0	577.0	58.7	210.6			
20.0110	6	1722.1	637.1	58.7	210.6			
20.0110	7	1506.8	712.0	58.7	210.6			
20.0110	8	1308.7	785.0	58.7	210.6			
20.0110	9	522.3	845.7	58.7	210.6			
20.0110	10	993.0	911.7	58.7	210.6			
20.0110	11	947.1	996.2	58.7	210.6			
20.0110	12	912.7	1077.4	58.7	210.6			
20.0110	13	895.5	1156.0	58.7	210.6			
20.0110	14	711.9	1225.9	58.7	210.6			
20.0110	15	1148.1	1307.1	58.7	210.6			
20.0110	16	769.2	1390.4	58.7	210.6			
20.0110	17	620.1	1451.1	58.7	210.6			
20.0110	18	1528.3	1513.6	58.7	210.6			
20.0110	19	1227.0	1577.6	58.7	210.6			
20.0110	20	1227.0	1634.5	58.7	210.6			
20.0110	21	1448.8	1695.0	58.7	210.6			
	TOTAL	42499.6	1059.1	0.0	230.0	0.10158220 10	0.28446720 11	0.27577590 11

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

CARGO AND BAGGAGE

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	5	2110.8	577.0	58.7	210.6			
20.0110	6	2541.9	637.1	58.7	210.6			
20.0110	7	2734.9	712.0	58.7	210.6			
20.0110	8	2517.4	785.0	58.7	210.6			
20.0110	9	2335.7	845.7	58.7	210.6			
20.0110	10	2233.1	911.7	58.7	210.6			
20.0110	16	1406.9	1390.4	58.7	210.6			
20.0110	17	1777.6	1451.1	58.7	210.6			
20.0110	18	1825.4	1513.6	58.7	210.6			
20.0110	19	2147.1	1577.6	58.7	210.6			
	TOTAL	43461.5	991.0	0.0	160.0	0.1038814D 10	0.2521226D 11	0.2432346D 11

ZERO FUEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
	337998.6	1181.5	136.2	192.8	0.2051944D 11	0.6896355D 11	0.8517387D 11

FUEL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
20.0110	12	3236.2	1077.4	58.7	210.6			
20.0110	13	2947.3	1156.0	53.7	210.6			
20.0110	14	576.8	1225.9	53.7	210.6			
10.0110	27	4036.1	1029.7	150.5	164.1			
10.0110	28	3352.9	1074.0	150.5	164.1			
10.0110	29	4135.9	1118.4	150.5	164.1			
10.0110	30	3907.6	1162.7	150.5	164.1			
10.0110	31	4160.0	1207.1	150.5	164.1			
10.0110	37	3082.6	1075.5	217.5	169.8			
10.0110	33	2563.0	1115.5	217.5	169.8			
10.0110	39	2932.2	1155.6	217.5	169.8			
10.0110	40	2593.8	1195.6	217.5	169.8			
10.0110	41	2651.8	1235.7	217.5	169.8			
10.0110	42	829.2	1275.7	217.5	169.8			
10.0110	47	2193.7	1121.3	284.5	175.6			
10.0110	48	2011.1	1157.0	284.5	175.6			
10.0110	49	2145.5	1192.8	284.5	175.6			
10.0110	50	2354.7	1228.5	284.5	175.6			
10.0110	51	1972.9	1264.3	284.5	175.6			
10.0110	52	705.5	1300.0	284.5	175.6			
10.0110	57	979.4	1167.1	351.5	181.3			
10.0110	58	1431.5	1198.5	351.5	181.3			
10.0110	59	1253.3	1230.0	351.5	181.3			
10.0110	60	1496.5	1261.4	351.5	181.3			
10.0110	61	1353.7	1292.9	351.5	181.3			

MASS DISTRIBUTION MODULE OUTPUT

MASS DISTRIBUTION PROGRAM   PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

FUEL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0110	62	666.5	1324.3	351.5	181.3			
10.0120	67	606.7	1209.9	412.3	188.4			
10.0120	68	869.0	1238.3	412.3	188.4			
10.0120	69	1161.2	1266.8	412.3	188.4			
10.0120	70	1136.3	1295.2	412.3	188.4			
10.0120	71	937.1	1323.7	412.3	188.4			
10.0120	72	490.5	1352.2	412.3	188.4			
10.0120	77	490.5	1249.7	466.7	196.9			
10.0120	78	883.1	1276.5	466.7	196.9			
10.0120	79	869.0	1303.3	466.7	196.9			
10.0120	80	951.2	1330.0	466.7	196.9			
10.0120	81	846.6	1356.8	466.7	196.9			
10.0120	82	370.2	1383.6	466.7	196.9			
10.0120	87	435.7	1289.6	521.3	205.3			
10.0120	88	635.8	1314.7	521.3	205.3			
10.0120	89	703.8	1339.8	521.3	205.3			
10.0120	90	547.0	1364.8	521.3	205.3			
10.0120	91	643.2	1389.9	521.3	205.3			
10.0120	92	233.2	1415.0	521.3	205.3			
10.0120	97	111.2	1329.5	575.7	213.8			
10.0120	98	336.1	1352.9	575.7	213.8			
10.0120	99	357.7	1376.2	575.7	213.8			
10.0120	100	355.2	1399.6	575.7	213.8			
10.0120	101	342.8	1423.0	575.7	213.8			
10.0120	102	237.4	1446.4	575.7	213.8			
10.0120	107	132.0	1369.3	630.3	222.3			
10.0120	108	410.8	1391.0	630.3	222.3			
10.0120	109	468.1	1412.7	630.3	222.3			
10.0120	110	548.6	1434.4	630.3	222.3			
10.0120	111	604.2	1456.1	630.3	222.3			
10.0120	112	112.0	1477.8	630.3	222.3			
10.0120	117	132.8	1409.2	684.8	230.7			
10.0120	118	493.0	1429.2	684.8	230.7			
10.0120	119	541.2	1449.2	684.8	230.7			
10.0120	120	434.1	1469.2	684.8	230.7			
10.0120	121	404.2	1489.2	684.8	230.7			
10.0120	122	236.5	1509.2	684.8	230.7			
10.0120	128	502.1	1467.4	739.3	239.2			
10.0120	129	390.9	1485.7	739.3	239.2			
10.0120	130	308.8	1504.0	739.3	239.2			
10.0120	131	444.9	1522.3	739.3	239.2			
10.0120	132	143.6	1540.6	739.3	239.2			
10.0120	138	183.4	1505.6	793.8	247.6			
10.0120	139	439.1	1522.2	793.8	247.6			
10.0120	140	200.9	1538.8	793.8	247.6			

MASS DISTRIBUTION PROGRAM PADS BASELINE AIRCRAFT, FULL FUEL 11/19/82

FUEL

GROUP	PANEL	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
10.0120	141	268.9	1555.4	793.8	247.6			
10.0120	142	85.5	1572.1	793.8	247.6			
10.0120	148	227.4	1543.7	848.3	256.1			
10.0120	149	118.7	1558.7	848.3	256.1			
10.0120	150	337.6	1573.6	848.3	256.1			
	TOTAL	166000.0	1219.0	302.8	104.9	0.1336783D 11	0.1275079D 12	0.1351159D 12

TAKEOFF	MASS (LBS)	XCG (IN)	YCG (IN)	ZCG (IN)	IXCG (LB-IN2)	IYCG (LB-IN2)	IZCG (LB-IN2)
	503998.6	1193.8	191.1	190.2	0.4158749D 11	0.7179249D 11	0.1098864D 12

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REAL DIAG DIAG SORT INDICATORS 12

DIAG	DIAG		DIAG		DIAG		DIAG		DIAG		DIAG	
1	1	0.298794D 04	2	0.298794D 04	3	0.298794D 04	4	0.502350D 04	5	0.502350D 04	6	0.502350D 04
	7	0.384064E 04	8	0.384064E 04	9	0.384064E 04	10	0.391737E 04	11	0.391737E 04	12	0.391737E 04
	13	0.457215E 04	14	0.457215E 04	15	0.457215E 04	16	0.621828E 04	17	0.621828E 04	18	0.621828E 04
	19	0.642691E 04	20	0.642691E 04	21	0.642691E 04	22	0.601863E 04	23	0.601863E 04	24	0.601863E 04
	25	0.475039E 04	26	0.475039E 04	27	0.475039E 04	28	0.619267E 04	29	0.619267E 04	30	0.619267E 04
	31	0.446954E 04	32	0.446954E 04	33	0.446954E 04	34	0.717897E 04	35	0.717897E 04	36	0.717897E 04
	37	0.673073E 04	38	0.673073E 04	39	0.673073E 04	40	0.400692E 04	41	0.400692E 04	42	0.400692E 04
	43	0.479135E 04	44	0.479135E 04	45	0.479135E 04	46	0.471295E 04	47	0.471295E 04	48	0.471295E 04
	49	0.372051E 04	50	0.372051E 04	51	0.372051E 04	52	0.556595E 04	53	0.556596E 04	54	0.556596E 04
	55	0.495786E 04	56	0.495786E 04	57	0.495786E 04	58	0.345387E 04	59	0.345387E 04	60	0.345387E 04
	61	0.478395E 04	62	0.473395E 04	63	0.478395E 04	64	0.522461E 04	65	0.522461E 04	66	0.522461E 04
	67	0.251114E 04	68	0.251114E 04	69	0.251114E 04	70	0.232164E 04	71	0.232164E 04	72	0.232164E 04
	73	0.715197E 02	74	0.715197E 02	75	0.715197E 02	76	0.318128E 03	77	0.318128E 03	78	0.318128E 03
	79	0.457619E 04	80	0.457619E 04	81	0.457619E 04	82	0.431430E 04	83	0.431430E 04	84	0.431430E 04
	85	0.460244E 04	86	0.460244E 04	87	0.460244E 04	88	0.439093E 04	89	0.439093E 04	90	0.439093E 04
	91	0.460295E 04	92	0.460295E 04	93	0.460295E 04	94	0.262222E 03	95	0.262222E 03	96	0.262222E 03
	97	0.732178E 02	98	0.732178E 02	99	0.732178E 02	100	0.539201E 02	101	0.539201E 02	102	0.539201E 02
	106	0.212842E 03	107	0.212842E 03	108	0.212842E 03	109	0.352959E 04	110	0.352959E 04	111	0.352959E 04
	112	0.296373E 04	113	0.296375E 04	114	0.296375E 04	115	0.338773E 04	116	0.338773E 04	117	0.338773E 04
	118	0.300630E 04	119	0.300630E 04	120	0.300638E 04	121	0.306249E 04	122	0.306249E 04	123	0.306249E 04
	124	0.157156E 04	125	0.157156E 04	126	0.157156E 04	127	0.260181E 03	128	0.268181E 03	129	0.268181E 03
	130	0.295141E 02	131	0.295141E 02	132	0.295141E 02	136	0.152111E 03	137	0.152111E 03	138	0.152111E 03
	139	0.262505E 04	140	0.262505E 04	141	0.262505E 04	142	0.236185E 04	143	0.236185E 04	144	0.236185E 04
	145	0.249717E 04	146	0.249717E 04	147	0.249717E 04	148	0.271228E 04	149	0.271228E 04	150	0.271228E 04
	151	0.233446E 04	152	0.233446E 04	153	0.233446E 04	154	0.139993E 04	155	0.139993E 04	156	0.139993E 04
	157	0.249451E 03	158	0.249451E 03	159	0.249451E 03	160	0.295141E 02	161	0.295141E 02	162	0.295141E 02
	166	0.103957E 03	167	0.103957E 03	168	0.103957E 03	169	0.138125E 04	170	0.138125E 04	171	0.138125E 04
	172	0.177612E 04	173	0.177612E 04	174	0.177612E 04	175	0.154787E 04	176	0.154787E 04	177	0.154787E 04
	178	0.181178E 04	179	0.181178E 04	180	0.181178E 04	181	0.164859E 04	182	0.164859E 04	183	0.164859E 04
	184	0.928775E 03	185	0.928775E 03	186	0.928775E 03	187	0.647041E 02	188	0.647041E 02	189	0.647041E 02
	190	0.283790E 01	191	0.283790E 01	192	0.283790E 01	196	0.933669E 02	197	0.933669E 02	198	0.933669E 02
	199	0.919183E 03	200	0.919183E 03	201	0.919183E 03	202	0.111534E 04	203	0.111534E 04	204	0.111534E 04
	205	0.140665E 04	206	0.140665E 04	207	0.140665E 04	208	0.139771E 04	209	0.139771E 04	210	0.139771E 04
	211	0.118766E 04	212	0.118766E 04	213	0.118766E 04	214	0.731184E 03	215	0.731184E 03	216	0.731184E 03
	217	0.120895E 03	218	0.120895E 03	219	0.120895E 03	220	0.374603E 02	221	0.374603E 02	222	0.374603E 02
	226	0.851370E 00	227	0.851370E 00	228	0.851370E 00	229	0.799261E 03	230	0.799261E 03	231	0.799261E 03
	232	0.113567E 04	233	0.113567E 04	234	0.113567E 04	235	0.112153E 04	236	0.112158E 04	237	0.112158E 04
	238	0.120943E 04	239	0.120943E 04	240	0.120943E 04	241	0.110570E 04	242	0.110570E 04	243	0.110570E 04
	244	0.582455E 03	245	0.582455E 03	246	0.582455E 03	247	0.168855E 03	248	0.168855E 03	249	0.168855E 03
	250	0.193653E 02	251	0.193653E 02	252	0.193653E 02	256	0.681096E 01	257	0.681096E 01	258	0.681096E 01
	259	0.639227E 03	260	0.639227E 03	261	0.639227E 03	262	0.807756E 03	263	0.807756E 03	264	0.807756E 03
	265	0.874601E 03	266	0.874601E 03	267	0.874681E 03	268	0.720933E 03	269	0.720933E 03	270	0.720933E 03
	271	0.810064E 03	272	0.810064E 03	273	0.810064E 03	274	0.309725E 03	275	0.309725E 03	276	0.309725E 03
	277	0.479605E 02	278	0.479605E 02	279	0.479605E 02	280	0.397306E 01	281	0.397306E 01	282	0.397306E 01
	289	0.310724E 03	290	0.310724E 03	291	0.310724E 03	292	0.496207E 03	293	0.496207E 03	294	0.496207E 03
	295	0.516936E 03	296	0.516936E 03	297	0.516936E 03	298	0.519270E 03	299	0.519270E 03	300	0.519270E 03
	301	0.505585E 03	302	0.505585E 03	303	0.505585E 03	304	0.294422E 03	305	0.294422E 03	306	0.294422E 03
	307	0.422847E 02	308	0.422847E 02	309	0.422847E 02	310	0.283790E 01	311	0.283790E 01	312	0.283790E 01

MASS DISTRIBUTION MODULE OUTPUT

MDM FULL INERTIA MATRIX   3765 BY 3765   JOB 5443   SECT 104   DATE 11/22/82   TIME 1124   PAGE 2

REAL   DIAG   DIAG SORT   INDICATORS 12

DIAG	DIAG		DIAG		DIAG		DIAG		DIAG		DIAG	
1	319	0.275284D 03	320	0.275284D 03	321	0.275284D 03	322	0.528623D 03	323	0.528623D 03	324	0.528623D 03
	325	0.585892E 03	326	0.585892E 03	327	0.585892E 03	328	0.673497E 03	329	0.673497E 03	330	0.673497E 03
	331	0.723148E 03	332	0.723148E 03	333	0.723148E 03	334	0.226134E 03	335	0.226134E 03	336	0.226134E 03
	337	0.108975E 03	338	0.109975E 03	339	0.108975E 03	340	0.283790E 01	341	0.283790E 01	342	0.283790E 01
	349	0.263627E 03	350	0.263627E 03	351	0.263627E 03	352	0.610097E 03	353	0.610097E 03	354	0.610097E 03
	355	0.653257E 03	356	0.653257E 03	357	0.653257E 03	358	0.549876E 03	359	0.549876E 03	360	0.549876E 03
	361	0.519145E 03	362	0.519145E 03	363	0.519145E 03	364	0.309200E 03	365	0.309200E 03	366	0.309200E 03
	367	0.383116E 02	368	0.383116E 02	369	0.393116E 02	370	0.397306E 01	371	0.397306E 01	372	0.397306E 01
	379	0.973400E 02	380	0.973400E 02	381	0.973400E 02	382	0.601192E 03	383	0.601192E 03	384	0.601192E 03
	385	0.470391E 03	386	0.470391E 03	387	0.470391E 03	388	0.393045E 03	389	0.393045E 03	390	0.393045E 03
	391	0.527463E 03	392	0.527463E 03	393	0.527463E 03	394	0.199497E 03	395	0.199497E 03	396	0.199497E 03
	397	0.550552E 02	398	0.550552E 02	399	0.550552E 02	400	0.275276E 02	401	0.275276E 02	402	0.275276E 02
	409	0.593797E 02	410	0.598797E 02	411	0.598797E 02	412	0.278783E 03	413	0.278783E 03	414	0.278783E 03
	415	0.508882E 03	416	0.508882E 03	417	0.508882E 03	418	0.274645E 03	419	0.274645E 03	420	0.274645E 03
	421	0.342421E 03	422	0.342421E 03	423	0.342421E 03	424	0.169776E 03	425	0.169776E 03	426	0.169776E 03
	427	0.471091E 02	428	0.471091E 02	429	0.471091E 02	439	0.383116E 02	440	0.383116E 02	441	0.393116E 02
	442	0.322773E 03	443	0.322773E 03	444	0.322773E 03	445	0.175448E 03	446	0.175448E 03	447	0.175448E 03
	448	0.446354E 03	449	0.446354E 03	450	0.446354E 03	451	0.638527E 02	452	0.638527E 02	453	0.638527E 02
	454	0.805963E 02	455	0.805963E 02	456	0.805963E 02	457	0.879748E 01	458	0.879748E 01	459	0.879740E 01
	469	0.105002E 02	470	0.105002E 02	471	0.105002E 02	472	0.777594E 02	473	0.777584E 02	474	0.777584E 02
	475	0.411495E 02	476	0.411495E 02	477	0.411495E 02	478	0.431361E 02	479	0.431361E 02	480	0.431361E 02
	481	0.490956E 02	482	0.490956E 02	483	0.490956E 02	484	0.411495E 02	485	0.411495E 02	486	0.411495E 02
	487	0.175950E 02	488	0.175950E 02	489	0.175950E 02	493	0.794611E 01	494	0.794611E 01	495	0.794611E 01
	502	0.334872E 02	503	0.334872E 02	504	0.334872E 02	505	0.363251E 02	506	0.363251E 02	507	0.363251E 02
	508	0.323521E 02	509	0.323521E 02	510	0.323521E 02	511	0.315007E 02	512	0.315007E 02	513	0.315007E 02
	514	0.451226E 02	515	0.451226E 02	516	0.451226E 02	517	0.127705E 02	518	0.127705E 02	519	0.127705E 02
	667	0.147571E 02	668	0.147571E 02	669	0.147571E 02	670	0.147571E 02	671	0.147571E 02	672	0.147571E 02
	673	0.391630E 02	674	0.391630E 02	675	0.391630E 02	697	0.147571E 02	698	0.147571E 02	699	0.147571E 02
	700	0.147571E 02	701	0.147571E 02	702	0.147571E 02	703	0.354737E 02	704	0.354737E 02	705	0.354737E 02
	1327	0.147571E 02	1328	0.147571E 02	1329	0.147571E 02	1330	0.187301E 02	1331	0.187301E 02	1332	0.187301E 02
	1333	0.187301E 02	1334	0.187301E 02	1335	0.187301E 02	1354	0.993264E 01	1355	0.993264E 01	1356	0.993264E 01
	1357	0.207167E 02	1358	0.207167E 02	1359	0.207167E 02	1360	0.167436E 02	1361	0.167436E 02	1362	0.167436E 02
	1363	0.167436E 02	1364	0.167436E 02	1365	0.167436E 02	1387	0.139057E 02	1388	0.139057E 02	1389	0.139057E 02
	1390	0.187301E 02	1391	0.187301E 02	1392	0.187301E 02	1393	0.993264E 01	1394	0.993264E 01	1395	0.993264E 01
	1417	0.283790E 01	1418	0.283790E 01	1419	0.283790E 01	1420	0.879748E 01	1421	0.879748E 01	1422	0.879748E 01
	1423	0.156084E 02	1424	0.156084E 02	1425	0.156084E 02	1600	0.232709E 02	1601	0.232708E 02	1602	0.232708E 02
	1603	0.255411E 02	1604	0.255411E 02	1605	0.255411E 02	1624	0.893938E 02	1625	0.893938E 02	1626	0.893939E 02
	1627	0.885424E 02	1628	0.885424E 02	1629	0.885424E 02	1630	0.618662E 02	1631	0.618662E 02	1632	0.618662E 02
	1633	0.510822E 02	1634	0.510822E 02	1635	0.510822E 02	1654	0.698123E 02	1655	0.698123E 02	1656	0.698123E 02
	1657	0.786098E 02	1658	0.786098E 02	1659	0.786098E 02	1660	0.510822E 02	1661	0.510822E 02	1662	0.510822E 02
	1663	0.462578E 02	1664	0.462578E 02	1665	0.462578E 02	1684	0.539201E 02	1685	0.539201E 02	1686	0.539201E 02
	1687	0.550552E 02	1688	0.550552E 02	1689	0.550552E 02	1690	0.953534E 02	1691	0.953534E 02	1692	0.953534E 02
	1693	0.865559E 02	1694	0.865559E 02	1695	0.865559E 02	1714	0.510822E 02	1715	0.510822E 02	1716	0.510822E 02
	1717	0.658393E 02	1718	0.653393E 02	1719	0.658393E 02	1720	0.482443E 02	1721	0.482443E 02	1722	0.482443E 02
	1723	0.431351E 02	1724	0.431361E 02	1725	0.431361E 02	1744	0.490957E 02	1745	0.490957E 02	1746	0.490957E 02
	1747	0.315007E 02	1748	0.315007E 02	1749	0.315007E 02	1750	0.246897E 02	1751	0.246897E 02	1752	0.246897E 02
	1753	0.490957E 02	1754	0.490957E 02	1755	0.490957E 02	1897	0.335954E 02	1898	0.385954E 02	1899	0.335954E 02
	1900	0.185599E 03	1901	0.185599E 03	1902	0.105599E 03	1903	0.138489E 03	1904	0.138489E 03	1905	0.138489E 03

MDM FULL INERTIA MATRIX 3765 BY 3765 JOB 5443 SECT 104 DATE 11/22/82 TIME 1124 PAGE 3

REAL DIAG DIAG SORT INDICATORS 12

DIAG	DIAG		DIAG		DIAG		DIAG		DIAG		DIAG	
1	1927	0.686772D 02	1928	0.686772D 02	1929	0.686772D 02	1930	0.227032D 03	1931	0.227032D 03	1932	0.227032D 03
	1933	0.186734E 03	1934	0.186734E 03	1935	0.186734E 03	1957	0.363251E 02	1958	0.363251E 02	1959	0.363251E 02
	1960	0.168004E 03	1961	0.168004E 03	1962	0.168004E 03	1963	0.953534E 02	1964	0.953534E 02	1965	0.953534E 02
	1987	0.786098E 02	1988	0.786098E 02	1989	0.786098E 02	1990	0.115786E 03	1991	0.115786E 03	1992	0.115786E 03
	1993	0.786098E 02	1994	0.786098E 02	1995	0.786098E 02	2020	0.391630E 02	2021	0.391630E 02	2022	0.391630E 02
	2023	0.391630E 02	2024	0.391630E 02	2025	0.391630E 02	2326	0.111246E 03	2327	0.111246E 03	2328	0.111246E 03
	2356	0.115219E 03	2357	0.115219E 03	2358	0.115219E 03	2386	0.115219E 03	2387	0.115219E 03	2388	0.115219E 03
	2416	0.110394E 03	2417	0.110394E 03	2418	0.110394E 03	2896	0.814477E 02	2897	0.814477E 02	2898	0.814477E 02
	2926	0.114935E 03	2927	0.114935E 03	2928	0.114935E 03	2956	0.106137E 03	2957	0.106137E 03	2958	0.106137E 03
	2986	0.993265E 02	2987	0.993265E 02	2988	0.993265E 02	3016	0.845694E 02	3017	0.845694E 02	3018	0.845694E 02
	3046	0.774746E 02	3047	0.774746E 02	3048	0.774746E 02	3076	0.639527E 02	3077	0.639527E 02	3078	0.639527E 02
	3106	0.578931E 02	3107	0.578931E 02	3108	0.578931E 02	3136	0.510822E 02	3137	0.510822E 02	3138	0.510822E 02
	3139	0.136219E 02	3140	0.136219E 02	3141	0.136219E 02	3166	0.471091E 02	3167	0.471091E 02	3168	0.471091E 02
	3169	0.405820E 02	3170	0.405820E 02	3171	0.405820E 02	3196	0.303655E 02	3197	0.303655E 02	3198	0.303655E 02
	3199	0.402982E 02	3200	0.402982E 02	3201	0.402982E 02	3235	0.823034E 02	3236	0.823034E 02	3237	0.823034E 02
	3238	0.114760E 03	3239	0.114760E 03	3240	0.114760E 03	3241	0.112618E 03	3242	0.112618E 03	3243	0.112618E 03
	3244	0.145075E 03	3245	0.145075E 03	3246	0.145075E 03	3247	0.173175E 02	3248	0.173175E 02	3249	0.173175E 02
	3256	0.249052E 02	3257	0.249052E 02	3258	0.249052E 02	3259	0.660307E 02	3260	0.660307E 02	3261	0.660307E 02
	3262	0.129936E 03	3263	0.129936E 03	3264	0.129936E 03	3265	0.942114E 02	3266	0.942114E 02	3267	0.942114E 02
	3268	0.109351E 03	3269	0.109351E 03	3270	0.109351E 03	3271	0.123437E 03	3272	0.123437E 03	3273	0.123437E 03
	3274	0.953006E 02	3275	0.953006E 02	3276	0.953006E 02	3277	0.823034E 02	3278	0.823034E 02	3279	0.823034E 02
	3280	0.400444E 02	3281	0.400444E 02	3282	0.400444E 02	3283	0.249052E 02	3284	0.249052E 02	3285	0.249052E 02
	3286	0.249052E 02	3287	0.249052E 02	3288	0.249052E 02	3289	0.563090E 02	3290	0.563090E 02	3291	0.563090E 02
	3292	0.877128E 02	3293	0.877128E 02	3294	0.877128E 02	3295	0.909440E 02	3296	0.909440E 02	3297	0.909440E 02
	3298	0.855345E 02	3299	0.855345E 02	3300	0.855345E 02	3301	0.102852E 03	3302	0.102852E 03	3303	0.102852E 03
	3304	0.389915E 02	3305	0.389915E 02	3306	0.389915E 02	3307	0.573982E 02	3308	0.573982E 02	3309	0.573982E 02
	3310	0.281363E 02	3311	0.281363E 02	3312	0.281363E 02	3313	0.184066E 02	3314	0.184066E 02	3315	0.184066E 02
	3316	0.173175E 02	3317	0.173175E 02	3318	0.173175E 02	3319	0.519887E 02	3320	0.519887E 02	3321	0.519887E 02
	3322	0.606293E 02	3323	0.606293E 02	3324	0.606293E 02	3325	0.595402E 02	3326	0.595402E 02	3327	0.595402E 02
	3328	0.541307E 02	3329	0.541307E 02	3330	0.541307E 02	3331	0.465430E 02	3332	0.465430E 02	3333	0.465430E 02
	3334	0.671279E 02	3335	0.671279E 02	3336	0.671279E 02	3337	0.552199E 02	3338	0.552199E 02	3339	0.552199E 02
	3340	0.227269E 02	3341	0.227269E 02	3342	0.227269E 02	3343	0.151755E 02	3344	0.151755E 02	3345	0.151755E 02
	3346	0.238161E 02	3347	0.238161E 02	3348	0.238161E 02	3349	0.422227E 02	3350	0.422227E 02	3351	0.422227E 02
	3352	0.389915E 02	3353	0.389915E 02	3354	0.389915E 02	3355	0.335821E 02	3356	0.335821E 02	3357	0.335821E 02
	3358	0.508996E 02	3359	0.508996E 02	3360	0.508996E 02	3361	0.779468E 02	3362	0.779468E 02	3363	0.779468E 02
	3364	0.108261E 03	3365	0.108261E 03	3366	0.108261E 03	3367	0.487213E 02	3368	0.487213E 02	3369	0.487213E 02
	3370	0.303147E 02	3371	0.303147E 02	3372	0.303147E 02	3373	0.119080E 02	3374	0.119080E 02	3375	0.119080E 02
	3376	0.140863E 02	3377	0.140863E 02	3378	0.140863E 02	3379	0.314038E 02	3380	0.314038E 02	3381	0.314038E 02
	3382	0.411335E 02	3383	0.411335E 02	3384	0.411335E 02	3385	0.389915E 02	3386	0.389915E 02	3387	0.389915E 02
	3388	0.379024E 02	3389	0.379024E 02	3390	0.379024E 02	3391	0.303147E 02	3392	0.303147E 02	3393	0.303147E 02
	3394	0.758048E 02	3395	0.758048E 02	3396	0.758048E 02	3397	0.584510E 02	3398	0.584510E 02	3399	0.584510E 02
	3400	0.281363E 02	3401	0.281363E 02	3402	0.281363E 02	3403	0.972973E 01	3404	0.972973E 01	3405	0.972973E 01
	3406	0.972973E 01	3407	0.972973E 01	3408	0.972973E 01	3409	0.227269E 02	3410	0.227269E 02	3411	0.227269E 02
	3412	0.281363E 02	3413	0.281363E 02	3414	0.281363E 02	3415	0.281363E 02	3416	0.281363E 02	3417	0.281363E 02
	3418	0.281363E 02	3419	0.281363E 02	3420	0.281363E 02	3421	0.281363E 02	3422	0.281363E 02	3423	0.281363E 02
	3424	0.563090E 02	3425	0.563090E 02	3426	0.563090E 02	3427	0.682171E 02	3428	0.682171E 02	3429	0.682171E 02
	3430	0.140863E 02	3431	0.140863E 02	3432	0.140863E 02	3433	0.753774E 01	3434	0.753774E 01	3435	0.753774E 01
	3436	0.377572E 01	3437	0.377572E 01	3438	0.377572E 01	3439	0.381202E 01	3440	0.381202E 01	3441	0.381202E 01

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REAL DIAG DIAG SORT INDICATORS 12

DIAG	DIAG		DIAG		DIAG		DIAG		DIAG		DIAG	
1	3442	0.292255D 02	3443	0.292255D 02	3444	0.292255D 02	3445	0.292255D 02	3446	0.292255D 02	3447	0.292255D 02
	3448	0.275918E 02	3449	0.275918E 02	3450	0.275918E 02	3451	0.276201E 02	3452	0.276201E 02	3453	0.276201E 02
	3454	0.330375E 02	3455	0.330375E 02	3456	0.330375E 02	3457	0.330012E 02	3458	0.330012E 02	3459	0.330012E 02
	3460	0.758774E 01	3461	0.758774E 01	3462	0.758774E 01	3463	0.758774E 01	3464	0.758774E 01	3465	0.758774E 01
	3469	0.104885E 01	3470	0.104885E 01	3471	0.104885E 01	3478	0.104885E 01	3479	0.104885E 01	3480	0.104885E 01
	3491	0.945986E 01	3482	0.945986E 01	3483	0.945986E 01	3484	0.105087E 02	3485	0.105087E 02	3486	0.105087E 02
	3508	0.104885E 01	3509	0.104885E 01	3510	0.104885E 01	3511	0.525435E 01	3512	0.525435E 01	3513	0.525435E 01
	3514	0.514745E 02	3515	0.514745E 02	3516	0.514745E 02	3517	0.199585E 02	3518	0.199585E 02	3519	0.199585E 02
	3520	0.136553E 02	3521	0.136553E 02	3522	0.136553E 02	3523	0.945986E 01	3524	0.945986E 01	3525	0.945986E 01
	3529	0.945986E 01	3530	0.945986E 01	3531	0.945986E 01	3532	0.147041E 02	3533	0.147041E 02	3534	0.147041E 02
	3535	0.178608E 02	3536	0.178608E 02	3537	0.178608E 02	3538	0.273105E 02	3539	0.273105E 02	3540	0.273105E 02
	3541	0.262617E 02	3542	0.262617E 02	3543	0.262617E 02	3544	0.128162E 03	3545	0.128162E 03	3546	0.128162E 03
	3547	0.157631E 02	3548	0.157631E 02	3549	0.157631E 02	3550	0.105087E 02	3551	0.105087E 02	3552	0.105087E 02
	3553	0.735206E 01	3554	0.735206E 01	3555	0.735206E 01	3556	0.420550E 01	3557	0.420550E 01	3558	0.420550E 01
	3559	0.210073E 02	3560	0.210073E 02	3561	0.210073E 02	3562	0.147041E 02	3563	0.147041E 02	3564	0.147041E 02
	3565	0.147041E 02	3566	0.147041E 02	3567	0.147041E 02	3568	0.136553E 02	3569	0.136553E 02	3570	0.136553E 02
	3571	0.168119E 02	3572	0.168119E 02	3573	0.168119E 02	3574	0.273105E 02	3575	0.273105E 02	3576	0.273105E 02
	3577	0.147041E 02	3578	0.147041E 02	3579	0.147041E 02	3580	0.945986E 01	3581	0.945986E 01	3582	0.945986E 01
	3583	0.630321E 01	3584	0.630321E 01	3585	0.630321E 01	3586	0.840092E 01	3587	0.840092E 01	3588	0.840092E 01
	3589	0.199585E 02	3590	0.199585E 02	3591	0.199585E 02	3592	0.126064E 02	3593	0.126064E 02	3594	0.126064E 02
	3595	0.126064E 02	3596	0.126064E 02	3597	0.126064E 02	3598	0.126064E 02	3599	0.126064E 02	3600	0.126064E 02
	3601	0.126064E 02	3602	0.126064E 02	3603	0.126064E 02	3604	0.325649E 02	3605	0.325649E 02	3606	0.325649E 02
	3607	0.252128E 02	3608	0.252128E 02	3609	0.252128E 02	3610	0.840092E 01	3611	0.840092E 01	3612	0.840092E 01
	3613	0.420550E 01	3614	0.420550E 01	3615	0.420550E 01	3616	0.735206E 01	3617	0.735206E 01	3618	0.735206E 01
	3619	0.168119E 02	3620	0.168119E 02	3621	0.168119E 02	3622	0.105087E 02	3623	0.105087E 02	3624	0.105087E 02
	3625	0.105087E 02	3626	0.105087E 02	3627	0.105087E 02	3628	0.105087E 02	3629	0.105087E 02	3630	0.105087E 02
	3631	0.105087E 02	3632	0.105087E 02	3633	0.105087E 02	3634	0.168119E 02	3635	0.168119E 02	3636	0.168119E 02
	3637	0.115576E 02	3638	0.115576E 02	3639	0.115576E 02	3640	0.840092E 01	3641	0.840092E 01	3642	0.840092E 01
	3643	0.209771E 01	3644	0.209771E 01	3645	0.209771E 01	3646	0.525435E 01	3647	0.525435E 01	3648	0.525435E 01
	3649	0.136553E 02	3650	0.136553E 02	3651	0.136553E 02	3652	0.840092E 01	3653	0.840092E 01	3654	0.840092E 01
	3655	0.840092E 01	3656	0.840092E 01	3657	0.840092E 01	3658	0.840092E 01	3659	0.840092E 01	3660	0.840092E 01
	3661	0.945986E 01	3662	0.945986E 01	3663	0.945986E 01	3664	0.178608E 02	3665	0.178608E 02	3666	0.178608E 02
	3667	0.945986E 01	3668	0.945986E 01	3669	0.945986E 01	3670	0.630321E 01	3671	0.630321E 01	3672	0.630321E 01
	3673	0.209771E 01	3674	0.209771E 01	3675	0.209771E 01	3676	0.420550E 01	3677	0.420550E 01	3678	0.420550E 01
	3679	0.136553E 02	3680	0.136553E 02	3681	0.136553E 02	3682	0.105087E 02	3683	0.105087E 02	3684	0.105087E 02
	3685	0.105087E 02	3686	0.105087E 02	3687	0.105087E 02	3688	0.283695E 02	3689	0.283695E 02	3690	0.283695E 02
	3691	0.283695E 02	3692	0.283695E 02	3693	0.283695E 02	3694	0.304672E 02	3695	0.304672E 02	3696	0.304672E 02
	3697	0.840092E 01	3698	0.840092E 01	3699	0.840092E 01	3700	0.525435E 01	3701	0.525435E 01	3702	0.525435E 01
	3703	0.209771E 01	3704	0.209771E 01	3705	0.209771E 01	3724	0.100000E 00	3725	0.100000E 00	3726	0.100000E 00
	3727	0.100000E-03	3728	0.100000E-03	3729	0.100000E-03	3730	0.100000E 00	3731	0.100000E 00	3732	0.100000E 00
	3733	0.100000E-03	3734	0.100000E-03	3735	0.100000E-03	3736	0.906000E 03	3737	0.906000E 03	3738	0.906000E 03
	3739	0.780000E 06	3740	0.780000E 06	3741	0.100000E 06	3742	0.786200E 04	3743	0.786200E 04	3744	0.786200E 04
	3745	0.148600E 08	3746	0.148600E 08	3747	0.354000E 07	3748	0.142129E 05	3749	0.142129E 05	3750	0.142129E 05
	3751	0.132355E 08	3752	0.310252E 08	3753	0.294761E 08	3754	0.662712E 04	3755	0.662712E 04	3756	0.662712E 04
	3757	0.560218E 07	3758	0.129615E 08	3759	0.121389E 08	3760	0.223000E 04	3761	0.223000E 04	3762	0.223000E 04
	3763	0.268800D 06	3764	0.120800D 08	3765	0.119500D 08						

APPENDIX D

SYMMETRIC NASTRAN MASTER TABLE FOR BASELINE AIRPLANE

The symmetric master table for the baseline FEM follows. The master table defines the DOF, locations, and grid identification of points where stiffness and flexibility properties are defined. Cards labeled SICSYM apply to the symmetric model. The columns labeled SIC-COL and KAA-COL identify the row and column numbers of the SIC and stiffness matrices where the information applies. The column labeled GRIDID defines the grid point identification number of the finite element model. Integers under the heading "CP" define the coordinate system for which the DOF is defined in. The NASTRAN Master table is disscused in section 3.5.

SYMMETRIC NASTRAN MASTER TABLE FOR BASELINE AIRPLANE

APPENDIX D - SYMMETRIC NASTRAN MASTER TABLE FOR BASELINE AIRPLANE

SICSYA	TABLE	PADS	BASELINE						
SICSYC	SIC-COL	KAA-COL	PART#	DOF	GRIDID	X	Y	Z	CP
SICSYM	1	1	10.0120	3	153	740.759	954.452	-7.475	12
SICSYM	2	2	10.0120	3	155	746.571	954.452	-6.676	12
SICSYM	3	3	10.0120	3	157	762.327	954.452	-5.479	12
SICSYM	4	4	10.0120	3	159	788.599	954.452	-3.635	12
SICSYM	5	5	10.0120	3	161	814.882	954.452	-1.953	12
SICSYM	6	6	10.0320	3	163	834.171	954.452	-0.918	12
SICSYM	7	7	10.0320	3	165	848.792	954.452	-0.266	12
SICSYM	8	8	10.0120	3	253	692.606	891.884	-6.642	12
SICSYM	9	9	10.0120	3	255	699.387	891.884	-5.816	12
SICSYM	10	10	10.0120	3	257	717.758	891.884	-4.704	12
SICSYM	11	11	10.0120	3	259	748.384	891.884	-3.028	12
SICSYM	12	12	10.0120	3	261	779.021	891.884	-1.540	12
SICSYM	13	13	10.0320	3	263	801.500	891.884	-0.681	12
SICSYM	14	14	10.0320	3	265	818.539	891.884	-0.184	12
SICSYM	15	15	10.0120	3	353	648.404	834.451	-5.877	12
SICSYM	16	16	10.0120	3	355	656.076	834.451	-5.027	12
SICSYM	17	17	10.0120	3	357	676.845	834.451	-3.993	12
SICSYM	18	18	10.0120	3	359	711.469	834.451	-2.470	12
SICSYM	19	19	10.0120	3	361	746.102	834.451	-1.161	12
SICSYM	20	20	10.0320	3	363	771.511	834.451	-0.463	12
SICSYM	21	21	10.0320	3	365	790.768	834.451	-0.109	12
SICSYM	22	22	10.0120	3	453	604.203	777.018	-5.121	12
SICSYM	23	23	10.0120	3	455	612.765	777.018	-4.245	12
SICSYM	24	24	10.0120	3	457	635.933	777.018	-3.289	12
SICSYM	25	25	10.0120	3	459	674.555	777.018	-1.920	12
SICSYM	26	26	10.0120	3	461	713.184	777.018	-0.786	12
SICSYM	27	27	10.0320	3	463	741.522	777.018	-0.247	12
SICSYM	28	28	10.0320	3	465	762.998	777.018	-0.034	12
SICSYM	29	29	10.0120	3	553	555.526	713.770	-4.303	12
SICSYM	30	30	10.0120	3	555	565.069	713.770	-3.398	12
SICSYM	31	31	10.0120	3	557	590.879	713.770	-2.526	12
SICSYM	32	32	10.0120	3	559	633.902	713.770	-1.320	12
SICSYM	33	33	10.0120	3	561	676.932	713.770	-0.378	12
SICSYM	34	34	10.0510	3	563	708.497	713.770	-0.012	12
SICSYM	35	35	10.0510	3	565	732.415	713.770	.047	12
SICSYM	36	36	10.0120	3	653	502.377	644.712	-3.402	12
SICSYM	37	37	10.0120	3	655	512.992	644.712	-2.466	12
SICSYM	38	38	10.0120	3	657	541.686	644.712	-1.685	12
SICSYM	39	39	10.0120	3	659	589.515	644.712	-0.658	12
SICSYM	40	40	10.0120	3	661	637.351	644.712	.073	12
SICSYM	41	41	10.0510	3	663	672.438	644.712	.247	12
SICSYM	42	42	10.0510	3	665	699.024	644.712	.137	12
SICSYM	43	43	10.0321	0	730	.0	.0	.0	
SICSYM	44	44	10.0120	3	753	449.229	575.654	-2.471	12
SICSYM	45	45	10.0120	3	755	460.914	575.654	-1.506	12
SICSYM	46	46	10.0120	3	757	492.493	575.654	-0.821	12
SICSYM	47	47	10.0120	3	759	545.129	575.654	.019	12

SYMMETRIC NASTRAN MASTER TABLE FOR BASELINE AIRPLANE

SICSYM	48	48	10.0120	3	761	597.769	575.654	.535	12
SICSYM	49	49	10.0510	3	763	636.379	575.654	.513	12
SICSYM	50	50	10.0510	3	765	665.633	575.654	.229	12
SICSYM	51	51	10.0120	3	853	396.081	506.596	-1.505	12
SICSYM	52	52	10.0120	3	855	408.837	506.596	-0.513	12
SICSYM	53	53	10.0120	3	857	443.301	506.596	.071	12
SICSYM	54	54	10.0120	3	859	500.743	506.596	.714	12
SICSYM	55	55	10.0120	3	861	558.188	506.596	1.001	12
SICSYM	56	56	10.0510	3	863	600.320	506.596	.780	12
SICSYM	57	57	10.0510	3	865	632.241	506.596	.321	12
SICSYM	58	58	10.0120	3	953	352.564	450.048	-0.296	12
SICSYM	59	59	10.0120	3	955	366.199	450.048	.680	12
SICSYM	60	60	10.0120	3	957	403.024	450.048	1.118	12
SICSYM	61	61	10.0120	3	959	464.402	450.048	1.509	12
SICSYM	62	62	10.0120	3	961	525.782	450.048	1.523	12
SICSYM	63	63	10.0310	3	963	570.796	450.048	1.074	12
SICSYM	64	64	10.0310	3	965	604.900	450.048	.419	12
SICSYM	65	65	10.0120	3	1053	319.746	407.402	.906	12
SICSYM	66	66	10.0120	3	1055	334.044	407.402	1.842	12
SICSYM	67	67	10.0120	3	1057	372.651	407.402	2.128	12
SICSYM	68	68	10.0120	3	1059	436.997	407.402	2.267	12
SICSYM	69	69	10.0120	3	1061	501.344	407.402	2.016	12
SICSYM	70	70	10.0310	3	1063	548.532	407.402	1.348	12
SICSYM	71	71	10.0310	3	1065	584.281	407.402	.510	12
SICSYM	72	72	10.0110	3	1153	252.063	319.017	3.669	12
SICSYM	73	73	10.0110	3	1155	268.855	319.017	4.583	12
SICSYM	74	74	10.0110	3	1157	314.175	319.017	4.368	12
SICSYM	75	75	10.0110	3	1159	389.708	319.017	3.997	12
SICSYM	76	76	10.0110	3	1161	465.238	319.017	3.224	12
SICSYM	77	77	10.0110	3	1163	520.622	319.017	2.193	12
SICSYM	78	78	10.0110	3	1165	562.576	319.017	.840	12
SICSYM	79	79	10.0110	3	1253	148.727	183.872	14.792	12
SICSYM	80	80	10.0110	3	1255	169.849	183.872	14.537	12
SICSYM	81	81	10.0110	3	1257	226.787	183.872	11.212	12
SICSYM	82	82	10.0110	3	1259	321.669	183.872	5.222	12
SICSYM	83	83	10.0110	3	1261	416.673	183.872	2.794	12
SICSYM	84	84	10.0110	3	1263	486.360	183.872	1.560	12
SICSYM	85	85	10.0110	3	1265	539.155	183.872	.637	12
SICSYM	86	86	10.0311	0	1730	.0	.0	.0	
SICSYM	87	87	10.0610	1	1890	1095.3	419.5	116.1	
SICSYM	88	88	10.0610	2	1890	1095.3	419.5	116.1	
SICSYM	89	89	10.0610	3	1890	1095.3	419.5	116.1	
SICSYM	90	90	10.0610	4	1890	1095.3	419.5	116.1	
SICSYM	91	91	10.0610	5	1890	1095.3	419.5	116.1	
SICSYM	92	92	10.0610	6	1890	1095.3	419.5	116.1	
SICSYM	93	93	10.0910	1	1892	364.197	421.286	-7.292	12
SICSYM	94	94	10.0910	2	1892	364.197	421.286	-7.292	12
SICSYM	95	95	10.0910	3	1892	364.197	421.286	-7.292	12
SICSYM	96	96	10.0910	4	1892	364.197	421.286	-7.292	12
SICSYM	97	97	10.0910	5	1892	364.197	421.286	-7.292	12
SICSYM	98	98	10.0910	6	1892	364.197	421.286	-7.292	12
SICSYM	99	99	10.0810	1	2190	1280.0	217.0	17.1	
SICSYM	100	100	10.0810	2	2190	1280.0	217.0	17.1	
SICSYM	101	101	10.0810	3	2190	1280.0	217.0	17.1	
SICSYM	102	102	10.0810	4	2190	1280.0	217.0	17.1	

SYMMETRIC NASTRAN MASTER TABLE FOR BASELINE AIRPLANE

SICSYM	103	103	10.0810	5	2190	1280.0	217.0	17.1	
SICSYM	104	104	10.0810	6	2190	1280.0	217.0	17.1	
SICSYM	105	105	10.0820	1	2195	1277.0	112.2	150.2	
SICSYM	106	106	10.0820	2	2195	1277.0	112.2	150.2	
SICSYM	107	107	10.0820	3	2195	1277.0	112.2	150.2	
SICSYM	108	108	10.0820	4	2195	1277.0	112.2	150.2	
SICSYM	109	109	10.0820	5	2195	1277.0	112.2	150.2	
SICSYM	110	110	10.0820	6	2195	1277.0	112.2	150.2	
SICSYM	111	111	50.0100	1	3000	251.239	.000	62.000	11
SICSYM	112		50.0100	2	3000	251.239	.000	62.000	11
SICSYM	113	112	50.0100	3	3000	251.239	.000	62.000	11
SICSYM	114		50.0100	4	3000	251.239	.000	62.000	11
SICSYM	115	113	50.0100	5	3000	251.239	.000	62.000	11
SICSYM	116		50.0100	6	3000	251.239	.000	62.000	11
SICSYM	117	114	20.0110	3	5500	420.059	.00	62.0	11
SICSYM	118		20.0110	2	5500	420.059	.00	62.0	11
SICSYM	119		20.0110	4	5500	420.059	.00	62.0	11
SICSYM	120	115	30.0110	3	6001	2061.74	411.70	227.00	61
SICSYM	121	116	30.0110	3	6002	2102.26	411.70	227.00	61
SICSYM	122	117	30.0110	3	6003	2051.96	396.05	227.00	61
SICSYM	123	118	30.0110	3	6004	2085.77	374.94	227.00	61
SICSYM	124	119	30.0110	3	6005	2030.38	361.48	227.00	61
SICSYM	125	120	30.0110	3	6006	2068.92	337.41	227.00	61
SICSYM	126	121	30.0110	3	6007	1998.50	310.43	227.00	61
SICSYM	127	122	30.0110	3	6008	2044.05	281.99	227.00	61
SICSYM	128	123	30.0110	3	6009	1966.63	259.39	227.00	61
SICSYM	129	124	30.0110	3	6010	2019.18	226.57	227.00	61
SICSYM	130	125	30.0110	3	6011	1934.76	208.34	227.00	61
SICSYM	131	126	30.0110	3	6012	1994.31	171.15	227.00	61
SICSYM	132	127	30.0110	3	6013	1902.88	157.29	227.00	61
SICSYM	133	128	30.0110	3	6014	1969.44	115.73	227.00	61
SICSYM	134	129	30.0110	3	6015	1875.53	113.48	227.00	61
SICSYM	135	130	30.0110	3	6016	1952.86	78.78	227.00	61
SICSYM	136	131	30.0110	3	6017	1854.93	80.50	227.00	61
SICSYM	137	132	30.0111	0	6030	.0	.0	.0	
SICSYM	138		40.0110	2	7001	2074.10	.0	658.80	71
SICSYM	139		40.0110	2	7002	2108.57	.0	658.80	71
SICSYM	140		40.0110	2	7003	2045.31	.0	614.30	71
SICSYM	141		40.0110	2	7004	2081.91	.0	590.62	71
SICSYM	142		40.0110	2	7005	2016.52	.0	569.80	71
SICSYM	143		40.0110	2	7006	2062.22	.0	540.23	71
SICSYM	144		40.0110	2	7007	1987.73	.0	525.30	71
SICSYM	145		40.0110	2	7008	2042.53	.0	489.85	71
SICSYM	146		40.0110	2	7009	1958.94	.0	480.80	71
SICSYM	147		40.0110	2	7010	2022.83	.0	439.46	71
SICSYM	148		40.0110	2	7011	1930.15	.0	436.30	71
SICSYM	149		40.0110	2	7012	2009.48	.0	405.29	71
SICSYM	150		40.0110	2	7013	1901.35	.0	391.80	71
SICSYM	151	133	20.0510	1	8000	546.0	.0	18.700	
SICSYM	152		20.0510	2	8000	546.0	.0	18.700	
SICSYM	153	134	20.0510	3	8000	546.0	.0	18.700	
SICSYM	154		20.0510	4	8000	546.0	.0	18.700	
SICSYM	155	135	20.0510	5	8000	546.0	.0	18.700	
SICSYM	156		20.0510	6	8000	546.0	.0	18.700	
SICSYM	157	136	20.0110	3	8001	277.	.0	219.0	

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SICSYM	158		20.0110 2	8001	277.	.0	219.0
SICSYM	159		20.0110 4	8001	277.	.0	219.0
SICSYM	160	137	20.0110 3	8011	359.	.0	219.0
SICSYM	161		20.0110 2	8011	359.	.0	219.0
SICSYM	162		20.0110 4	8011	359.	.0	219.0
SICSYM	163	138	20.0110 3	8021	430.	.0	219.0
SICSYM	164		20.0110 2	8021	430.	.0	219.0
SICSYM	165		20.0110 4	8021	430.	.0	219.0
SICSYM	166	139	20.0110 3	8031	497.	.0	219.0
SICSYM	167		20.0110 2	8031	497.	.0	219.0
SICSYM	168		20.0110 4	8031	497.	.0	219.0
SICSYM	169	140	20.0520 1	8035	495.2	.0	132.8
SICSYM	170		20.0520 2	8035	495.2	.0	132.8
SICSYM	171	141	20.0520 3	8035	495.2	.0	132.8
SICSYM	172		20.0520 4	8035	495.2	.0	132.8
SICSYM	173	142	20.0520 5	8035	495.2	.0	132.8
SICSYM	174		20.0520 6	8035	495.2	.0	132.8
SICSYM	175	143	20.0110 3	8051	567.	.0	219.0
SICSYM	176		20.0110 2	8051	567.	.0	219.0
SICSYM	177		20.0110 4	8051	567.	.0	219.0
SICSYM	178	144	20.0110 3	8061	639.	.0	219.0
SICSYM	179		20.0110 2	8061	639.	.0	219.0
SICSYM	180		20.0110 4	8061	639.	.0	219.0
SICSYM	181	145	20.0110 3	8071	713.	.0	219.0
SICSYM	182		20.0110 2	8071	713.	.0	219.0
SICSYM	183		20.0110 4	8071	713.	.0	219.0
SICSYM	184	146	20.0110 3	8081	791.	.0	219.0
SICSYM	185		20.0110 2	8081	791.	.0	219.0
SICSYM	186		20.0110 4	8081	791.	.0	219.0
SICSYM	187	147	20.0110 3	8091	867.	.0	219.0
SICSYM	188		20.0110 2	8091	867.	.0	219.0
SICSYM	189		20.0110 4	8091	867.	.0	219.0
SICSYM	190	148	20.0110 3	8101	947.920	.0	219.0
SICSYM	191		20.0110 2	8101	947.920	.0	219.0
SICSYM	192		20.0110 4	8101	947.920	.0	219.0
SICSYM	193	149	20.0110 3	8141	1373.004.0		219.0
SICSYM	194		20.0110 2	8141	1373.004.0		219.0
SICSYM	195		20.0110 4	8141	1373.004.0		219.0
SICSYM	196	150	20.0110 3	8151	1423.	.0	219.0
SICSYM	197		20.0110 2	8151	1423.	.0	219.0
SICSYM	198		20.0110 4	8151	1423.	.0	219.0
SICSYM	199	151	20.0110 3	8161	1503.75	.0	219.0
SICSYM	200		20.0110 2	8161	1503.75	.0	219.0
SICSYM	201		20.0110 4	8161	1503.75	.0	219.0
SICSYM	202	152	20.0110 3	8171	1579.25	.0	219.0
SICSYM	203		20.0110 2	8171	1579.25	.0	219.0
SICSYM	204		20.0110 4	8171	1579.25	.0	219.0
SICSYM	205	153	20.0110 3	8181	1658.	.0	219.0
SICSYM	206		20.0110 2	8181	1658.	.0	219.0
SICSYM	207		20.0110 4	8181	1658.	.0	219.0
SICSYM	208	154	20.0110 3	8191	1753.5	.0	219.0
SICSYM	209		20.0110 2	8191	1753.5	.0	219.0
SICSYM	210		20.0110 4	8191	1753.5	.0	219.0
SICSYM	211	155	20.0110 3	8201	1799.21	.0	227.219
SICSYM	212		20.0110 2	8201	1799.21	.0	227.219

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SICSYM	213		20.0110	4	8201	1799.21	.0	227.219
SICSYM	214	156	20.0110	3	8211	1854.93	.0	237.243
SICSYM	215		20.0110	2	8211	1854.93	.0	237.243
SICSYM	216		20.0110	4	8211	1854.93	.0	237.243
SICSYM	217	157	20.0110	3	8221	1916.0	.0	248.227
SICSYM	218		20.0110	2	8221	1916.0	.0	248.227
SICSYM	219		20.0110	4	8221	1916.0	.0	248.227
SICSYM	220	158	20.0110	3	8231	1930.0	.0	250.745
SICSYM	221		20.0110	2	8231	1930.0	.0	250.745
SICSYM	222		20.0110	4	8231	1930.0	.0	250.745
SICSYM	223	159	20.0300	1	8241	2031.5	.0	269.0
SICSYM	224		20.0300	2	8241	2031.5	.0	269.0
SICSYM	225	160	20.0300	3	8241	2031.5	.0	269.0
SICSYM	226		20.0300	4	8241	2031.5	.0	269.0
SICSYM	227	161	20.0300	5	8241	2031.5	.0	269.0
SICSYM	228		20.0300	6	8241	2031.5	.0	269.0

APPENDIX E

SUPPORTING PANVALET DATA

E.0.1 Weights

Formation of the weight matrices used in the PADS design process is accomplished by use of FAMAS Program 172 (P172). Execution of P172 requires a MADOL deck and a Matdata set. Following is a list and description of the PANVALET data sets used to form the weights:

PAN Name	Run	Description
E750VFULL1	27A	MADOL to form 166,000 lb full wing fuel, 504,000 TOGW weight matrices.
E750VLOWW1	27A	MADOL to form 12,300 lb fuel, 350,300 lb TOGW weight matrices.
E750VLP631	27A	MADOL to form 30,000 lb fuel, forward c.g. weight matrices. (ref. WDA963)
E750VLP651	27A	MADOL to form 168,000 lb fuel, aft c.g. weight matrices (ref. WDA965)
E750VENDP1		MADOL to form 168,000 lb fuel, aft c.g., end loaded payload weight matrices. (ref. WDA965)
E750VNOPP1	Y	MADOL to form 168,000 lb fuel, no payload weight matrices. (ref. WDA965 landing gear load condition)
E750VHALF1	Y	MADOL to form half fuel weight matrices. (not used)
E750VTAXI1	Y	MADOL to form taxi weight matrices. (not used)

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E750VTHRD1	Y	MADOL to form 1/3 fuel matrices. (not used)
E750VUUPP1	27A	Matdata for the gear up condition for VFULL1, VHALF1, VLOWW1, VTAXI1, OR VTHRD1 MADOL.
E750VDOWN1	27A	Matdata for the gear down condition for VFULL1, VHALF1, VLOWW1, VTAXI1, or VTHRD1 MADOL.
E750VLM631	27A	Matdata for VLP631. Gear down condition.
E750VLM651	27A	Matdata for VLP651. Gear down condition.
E750VENDM1		Matdata for VENDP1.
E750VNOPM1	Y	Matdata for VNOPP1.

Y - Not Run

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E.0.2 Structures

The creation of the bulk data deck is described in the section titled Finite Element Analysis. A NASTRAN run assembles the bulk data from a dataset referred to as a set deck. A set deck contains the names of all the PANVALET data sets to be used for a particular analysis. This includes the executive and case control decks as well as the bulk data. Following is a list of all the set decks used for structural analyses and a description of what type of analysis they were used for:

PAN Name	Run NO.	Description
E750PZ145A	030	Static solution to compute internal loads for external loads on rigid airplane. Internal loads stored as E750PLDST1.
E750PZ150A	031	Computes K and SIC matrices for property cards E750MPRWA1 formed from internal loads computed in RUN030. K AND SIC stored under J5750.S200 and later changed to J5750.S100
E750PZ145B	033	Static solution to compute internal loads on first flex airplane external loads. Internal loads stored as E750PLDST2.
E750PZ150B	034	Computes K and SIC matrices for property cards E750MPRWA2 formed from internal loads computed in RUN033. K AND SIC stored under J5750.S200.
E750PZ146A	035	Fully Stress Design (FSD) to update wing property deck E750MPRWA2. Property deck stored as E750MPRWB1 (2nd flex FSD properties).
E750PZ145C	036	Static solution to compute internal loads on second flex airplane external loads without active controls (1st ACT off loads). Internal loads stored as E750PLDST3.
E750PZ150C	037	Computes K and SIC matrices for property cards E750MPRWA3 formed from internal loads computed in RUN036. K AND SIC stored under J5750.S9300.
E750PZ145D	038	Static solution to compute internal loads on 2nd ACT off airplane external loads. Internal loads stored as E750PLDST4.

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E750PZ145E	039	Static solution to compute internal loads on second flex airplane external loads. Internal loads stored as E750PLDST5.
E750PZ145F	040	Static solution to compute internal loads on 3rd ACT off airplane external loads. Internal loads stored as E750PLDST6.
E750PZ146B	041	Fully Stress Design (FSD) to update wing property deck E750MPRWA5. Property deck stored as E750MPRWB2 (3rd flex FSD properties).
E750PZ145G	048	Static solution to compute internal loads on second flex airplane external loads with 1st margin properties. Internal loads stored as E750PLDST8.
E750PZ145H	049	Static solution to compute internal loads on 10 production external load conditions with 2nd margin properties. Internal loads stored as E750PLDST9.
E750PZ145I	050	Static solution to compute internal loads for unit external SIC loads. E750MPRWA8 properties were used. Internal loads stored as J5750.S100.M3501.
E750PZ150D	054	Computes K and SIC matrices for property cards E750MPRWA8 formed from internal loads computed in RUN048. K AND SIC stored under J5750.S400.
E750MZ150M	055	Compute weight for FEM with E750MPRWA8 properties. Weight matrices stored under J5750.S400.
E750MZ150N	059	Compute weight for FEM with E580MPRPW2 properties. (discrete production properties) Weight matrices stored under J5750.S9001.

The bulk data deck is formed by assembling selective PANVALET files. Following is a list of the PANVALET files with description:

<u>PAN</u> <u>Name</u> <u>Description</u>

E750MCONW1 - wing connection cards
E750MCONC1 - fuselage barrel connection cards

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E750MCONFI - fuselage beam connection cards
E750MCONH1 - horizontal stabilizer connection cards
E750MCONV1 - vertical tail connection cards
E750MGRDW1 - wing grid cards
E750MGRDC1 - fuselage barrel grid cards
E750MGRDF1 - fuselage beam grid cards
E750MGRDH1 - horizontal stabilizer grid cards
E750MGRDV1 - vertical tail grid cards
E750MMATR1 - airplane material cards
E750MSMPCG - MPC, LDREF, LGROUP, SPC, CELAS4, and assorted other type cards
E750MPRPC1 - fuselage barrel property cards
E750MPRPF1 - fuselage beam property cards
E750MPRPH1 - horizontal stabilizer property cards
E750MPRPV1 - vertical stabilizer property cards
E750MLTRNS - FORCE, MOMENT, LDREF, and LMAT cards
E750MLGRPW - wing LGROUP cards
E750MLREFM - wing LDREF cards
E750MMPSIC - wing MPC cards
E750MTRFPY - DMIG for pylon stiffness
E750MCOW11 - coordinate system 11 definition for inner wing and fuselage barrel
E750MCOW12 - coordinate system 12 definition for outer wing
E750MCOW13 - coordinate system 13 definition
E750MCOW14 - coordinate system 14 definition
E750MCOH61 - coordinate system 61 definition for horizontal tail
E750MCOV71 - coordinate system 71 definition for vertical tail
E750MSUPSY - symmetric support card
E750MASETS - symmetric aset cards
E750MDLKA1 - PARAM cards for computing rigid format 150
E750MDELM1 - DELK cards for computing FEM weights
E750PX145* - executive control decks for static solutions
E750PY145* - case control decks for static solutions
E750PX146* - executive control decks for FSD runs
E750PY146* - case control decks for FSD runs
E750PX150* - executive control decks for rigid format 150 runs
E750PY150* - case control decks for rigid format 150 runs
E580MPRPW2 - discrete production property cards
E580MPIPW2 - production PIP cards with 0.08 minimum skin thickness
E750MPRWA* - property cards from PSASA sizing procedures
E750MPIWA* - PIP cards from PSASA sizing procedure
E750MPRWB* - property cards from FSD sizing procedures

* - signifies any character

E.0.3 Loads

Computation of theoretical maneuver loads in the PADS design process consist of execution of a series of modules. A typical load computation run would exercise the following PADS modules:

Vorlax Aero module
Load Grid Transformation module
Configuration module
Loads Weight module
PSRL module
Ground Handling module
Stacking module

Subsequent runs with inclusion of flexibility effects exercise the Jig Shape module prior to the PSRL module and execution of the first four modules a second time is not required. Further details on the function of these modules appears in section 3.1. Following is a listing of the PANVALET dataset names containing the inputs for computation of loads on the baseline aircraft along with the PADS run number in which they were used and a small description of each dataset:

PAN Name	Run	Description
E750XGRDT1	27A	Case data for Loads Grid transformation.
E750LGDHF0		Ground handling pgm. case data check case
E750LGDHF1	032	Ground handling pgm. case data for 1st flex airplane
E750LGDHF2	034	Ground handling pgm. case data for 2nd flex airplane
E750LGDHF4	037	Ground handling case data for ACS off case
E750LGDHM1	029-037	Ground handling input matdata for rigid and flexible airplane runs
E750LGDHR1	029	Ground handling MADOL section for rigid airplane
E750LJIGL1	031-037	Jig shape MADOL section for flexible airplane load computation
E750LPLCF0		PSRL case data check case
E750LPLCF1	031	PSRL case data to compute loads for 1st flexible airplane
E750LPLCF2	034	PSRL case data to compute loads for 2nd flexible airplane
E750LPLCF4	037	PSRL case data to compute loads for flexible airplane with ACS off
E750LPLCM1	029-037	PSRL input matdata for rigid and flexible airplane runs

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E750LPLCR1	029	PSRL case data to compute loads for rigid airplane
E750LSLML1	028	Loads Configuration MADOL section
E750LSLMM1	028	Loads Configuration input matdata set
E750LSLMU1	029	PADS alter cards used for the rigid airplane loads computation
E750LSLMU2	032	PADS alter cards used for the 1st flex airplane loads computation
E750LSLMU3	034	PADS alter cards used for the 2nd flex airplane loads computation
E750LSLMU4	037	PADS alter cards used for the ACS off loads computation
E750LSTKL1	029-038	Loads Stacking MADOL section for ACS on load conditions
E750LSTKL2	036	Loads Stacking MADOL section for ACS off load conditions
E750LVORL1	027	Vorlax aero computation MADOL section
E750LVORM3	027	Vorlax aero computation input matdata
E750LWTSL1	028	Loads Weight module MADOL section

E.0.4 Gust

The process of sizing the aircraft structure for the effects of gust loads is accomplished by exercising the gust modules developed for PADS. A typical sizing run would execute the following PADS modules:

NASTRAN RF145	Forms internal loads for unit external loads
GLPPR5 Preprocessor	Forms input matrices for GLP5K
GLP5K	Gust analysis
PSASA	Panel sizing routine

Following is a listing of the PANVALET dataset names containing the inputs for computation of gust loads and the associated sizing on the baseline aircraft along with the PADS run number in which they were used and a small description of each dataset:

PAN Name	Run	Description
E750GCS5K1	057	Case data for GLP5K1 pgm
E750GILD01	057	Gust internal load octagon for 350.3 k lb and E750MPRWA8 properties
E750GILD02	057	E750GILD01 internal loads combined with E750MLDST8 internal Static Loads.
E750GILD03	057	Gust internal load octagon for 350.3 k lb and E750MPRWA8 properties with 2g ACS off airplane
E750GILD04	057	E750GILD03 internal loads combined with E750MLDST8 internal Static Loads.
E750GILD05	057	Gust internal load octagon for 350.3 k lb and E750MPRWA8 properties with nacelle aero added.
E750GILD06	057	E750GILD05 internal loads combined with E750MLDST8 internal Static Loads.
E750GLPPIN	057	Input data for GLPPR5 Preprocessor (Fortran pgm)
E750GLPPR5	057	Source code for GLPPR5 Preprocessor
E750GMS5K1	057	Matdata input for GLP5K1 program
E750GPIWA1	057	Allowable update from E750GILD02 loads
E750GPIWA3	057	Allowable update from E750GILD04 loads
E750GPRWA1	057	Property update from E750GILD02 loads
E750GPRWA3	057	Property update from E750GILD04 loads

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E.0.5 Flutter

The Flutter analyses for the baseline aircraft consist of the execution of a series of PADS modules. These modules are the FLUT series which were developed for PADS. The following modules are used for the Flutter survey:

Module Name		Output/Function
FLUT1	-	doublet lattice unsteady aerodynamic matrices
FLUT2	-	grid transformation matrices
FLUT3	-	grid plots
FLUT4	-	transform aero to analysis DOF
FLUT5	-	vibration analysis
FLUT6	-	flutter analysis
FLUT7	-	vibration vector plots

Following is a listing and description of the PANVALET datasets which were used in conjunction with the FLUT series modules to evaluate flutter characteristic of the baseline model.

PAN Name	Run	Description
E580ADLC01		Case data for FLUT1 (Doublet Lattice Pgm.) for 9 k values (sect 101)
E580ADLM01		Matdata for FLUT1 (Doublet Lattice Pgm.) for 9 k values (sect 101)
E699FLUT4A	017	Case data for FLUT4 for unsteady aero transformation to analysis DOF.
E699FLUT4N	017-023	Mat data for FLUT4 (MATRICES 4503 AND 4326)
E699FLUT5A	017	Case data for FLUT5 (Vibration pgm.) Full fuel with 2nd flex stiffness matrix
E699FLUT5B	019	Case data for FLUT5. Min fuel with 2nd flex stiffness
E699FLUT5C	022	Case data for FLUT5. Fuel fuel with discrete production stiffness matrix (M3310.S9002).
E699FLUT5D	023	Case data for FLUT5. Min fuel with discrete production stiffness matrix (M3310.S9002).
E699FLUT6A	018	Case data for FLUT6 (Flutter analysis pgm.) Full fuel with 2nd flex stiffness matrix
E699FLUT6B	019	Case data for FLUT6. Min fuel with 2nd flex stiffness
E699FLUT6C	022	Case data for FLUT6. Fuel fuel with discrete production stiffness matrix (M3310.S9002).

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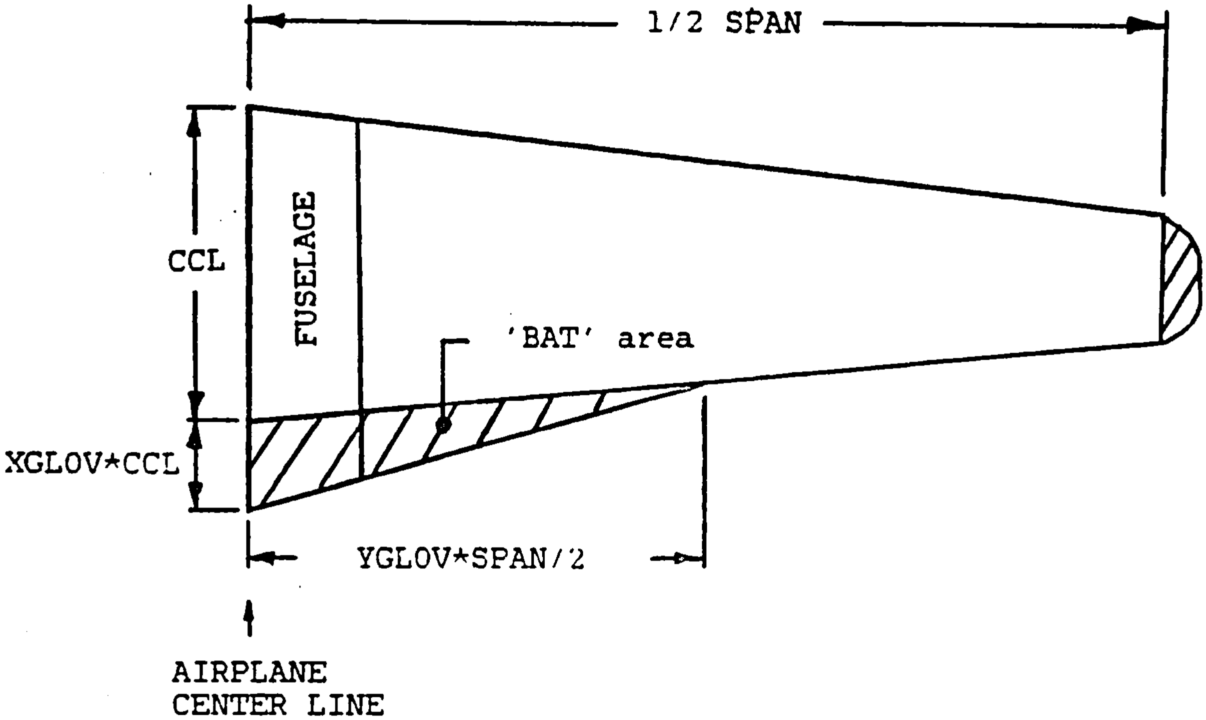
E699FLUT6D	023	Case data for FLUT6. Min fuel with discrete production stiffness matrix (M3310.S9002).
E699FLUT6N	018-024	Mat data for FLUT6. (speeds, MSC, and freqs.)
E699FLUT7A	017	Case data for FLUT7 (Vibration plots)
E699FLUT7B	019	Case data for FLUT7 (Vibration plots)
E699FLUT7C	022	Case data for FLUT7 (Vibration plots)
E699FLUT7D	023	Case data for FLUT7 (Vibration plots)
E699FLUT7N	017-023	Mat data for FLUT7. (Planform)
E750FLUT5A	046	Case data for FLUT5. Fuel fuel with discrete production stiffness matrix (M3310.S9002). Similar to RUN022 but with a new set of transformations.
E750FLUT6A	046	Case data for FLUT6. Fuel fuel with discrete production stiffness matrix (M3310.S9002). Similar to RUN022 but with a new set of transformations.
E750FLUT6N	046	Mat data for FLUT6. (speeds, MSC, and freqs.)
E750FLUT7A	046	Case data for FLUT7 (Vibration plots)
E750FLUT7N	046	Mat data for FLUT7. (Planform)

APPENDIX F

WING PARAMETER DEFINITIONS

Definition of geometric wing parameters as agreed upon by NASA-LaRC and Calac are shown in figure F-1. Figure F-1 contains the definition for wing reference area, aspect ratio, thickness to chord ratio, and sweep angle.

WING PARAMETER DEFINITIONS



PARAMETER	DEFINED AS
AREA	2.0 * wing trapezoid (excluding 'BAT' and wing tip area). Outer wing leading and trailing edges are extended to center line.
ASPECT RATIO	SPAN**2 / AREA
t/c (local)	Thickness / Chord
t/c (wing)	Average thickness / Average chord for wing trapezoid
TAPER RATIO	Tip chord / Centerline chord (CCL) (as defined in ASSET runs)
EXPOSED TAPER RATIO	Tip chord / Root chord for trapezoid (as defined for wing transforms)
SWEEP	Angle between center line and 0.25% chord line of wing trapezoid

FIGURE F-1 WING PARAMETER DEFINITIONS

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15. Supplementary Notes

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16. Abstract

This report is a documentation of work performed at Lockheed-California Company in the design of a transport aircraft wing for maximum fuel efficiency. It forms the basis for transmitting to NASA-LaRC material associated with Lockheed's design criteria, design methodology, and three design configurations. The design database includes complete finite element model description, sizing data, geometry data, loads data, and inertial data. This report illustrates a design process which satisfied the economics and practical aspects of a real design.

The report also discusses the cooperative study relationship between Lockheed and NASA during the course of the contract.

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