NASA Contractor Report 4298

Conceptual Design Optimization Study

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Conceptual Design Optimization Study

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FOREWORD

This final report documents research performed under contract NAS1-18015 from August 1987 to November 1988 to apply multilevel functional decomposition and optimization techniques to conceptual design of advanced fighter aircraft. The authors wish to thank the various functional discipline specialists who reviewed this work and provided helpful comments and suggestions. A special thanks is extended to Errie Norris and Sandra Kelly for their word processing and production support in the preparation of this report.

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Section I

INTRODUCTION

A method has been developed for improving the design process for large complex systems by decomposition of the process into a subset of activities which can be solved concurrently (references 1 and 2). Validation testing of the method has been accomplished in several single-disciplinary and multi-disciplinary applications with encouraging results (references 3, 4 and 5). However, application within an aircraft industrial setting has not been demonstrated.

PURPOSE

The purpose of this contract was to study the feasibility of applying multi-level functional decomposition and optimization techniques to conceptual design of a fighter aircraft. This report documents the results of that study.

SCOPE

The study was divided into four tasks. Task 1 defined the conceptual design process currently used at Rockwell International, NAA to develop fighter aircraft. Task 2 defines a modified design process which incorporates multi-level functional decomposition and optimization techniques. The scope of this task was limited to major wing design variables. Task 3 developed an implementation plan for the modified design process defined in Task 2. This plan included level of effort (engineering hours), computer resources and a schedule. Task 4 was to document the results of Tasks 1 through 3 and consists of this final report and a final briefing.

ORGANIZATION

This report is divided into three sections, documenting the results of Tasks 1 through 3, respectively. Each section includes a scope which defines in more detail what is covered in that section. In addition, Appendix A presents some supplemental information on the Parametric Synthesis Module computer program. This information embellishes the Task 2 results. Finally, Appendix B presents a printout of a HyperCard stack which was used to organize the large amount of data evaluated in Task 1.

Section II

TASK 1 RESULTS

INTRODUCTION

PURPOSE

The purpose of Task 1 was to define the conceptual design process currently used at Rockwell International, NAA to develop an advanced fighter type aircraft.

SCOPE

In defining the conceptual design process, the individual contributing functional disciplines (or processes) have been identified, along with tasks they perform, inputs and outputs, as well as connectivity between processes. The format chosen to present this information is an N^2 diagram. This report provides a brief explanation of design levels at Rockwell, then focuses on conceptual design level III for the details of the N^2 diagram. Following that is a discussion of how Rockwell currently optimizes conceptual designs.

The section concludes with a discussion of the nature of the interprocess connectivity, the amenability of the conceptual design process to further functional decomposition and developing analytical expressions for interprocess relationships of real world engineering problems. The information for the diagrams has been organized on a computer program for the Apple Macintosh called HyperCard. This program is well suited to hierarchical decomposition, as well as establishing complicated connectivities between individual pieces of information. The HyperCard database is known as a stack. Appendix B is a printout of the stack, and a disk with this particular stack is available from the author.

DESIGN PROCESS

A design consists of the drawings, specifications, analyses, processes, etc. required to produce a product.

The design process is an iterative procedure consisting of a definition task, followed by an analysis task and then an evaluation task. That is: (1) define a product, (2) analyze the characteristics of the product, and (3) evaluate the suitability of the product. This cycle is repeated until a product is defined which satisfies an original set of requirements in the best possible way. Figure 2-1 illustrates a generic design process which can be used for design of any product.

Within the aerospace industry it has become common practice to apply this process to the various phases of the life cycle of a product. Table 2 - I illustrates these phases, and further breaks the process down into design levels. Early phases examine many possible concepts with several aircraft configurations per concept. Later phases refine the design of a selected configuration. Manufacturing design and product support in the field complete the life cycle.

This contract is concerned with the early phases of the process which culminate with the selection of the final configuration or baseline design. The procedures used during these phases must be simple enough to allow examination of many configurations within a short period of time, yet detailed enough to distinguish between concepts and allow optimization of fundamental configuration design variables. The basic cost of the product is established during this time. Conversely, it becomes increasingly difficult to reduce product cost the further the product moves through the design cycle.

Table 2 - I. DESIGN LEVELS

PROGRAM PHASE	ACTIVITY	DESIGN LEVEL
	EXPLORATORY DEVELOPMENT	I. CONTINUING RESEARCH
CONCEPT EXPLORATION	CONCEPTUAL DESIGN	II. CONCEPT FORMULATION III. CONFIGURATION SELECTION
DEMONSTRATION/ VALIDATION	PRELIMINARY DESIGN	IV. CONFIGURATION REFINEMENT V. CONFIGURATION
FULL SCALE DEVELOPMENT	DETAIL DESIGN	VI. DETAIL DESIGN/DEVELOPMENT
PRODUCTION	SUSTAINING ENGINEERING	VII. PRODUCT MANUFACTURE
DEPLOYMENT	SUSTAINING ENGINEERING	VIII. PRODUCT VERIFICATION
OPERATIONAL	SUSTAINING ENGINEERING	IX. PRODUCT SUPPORT
		X. PRODUCT IMPROVEMENT

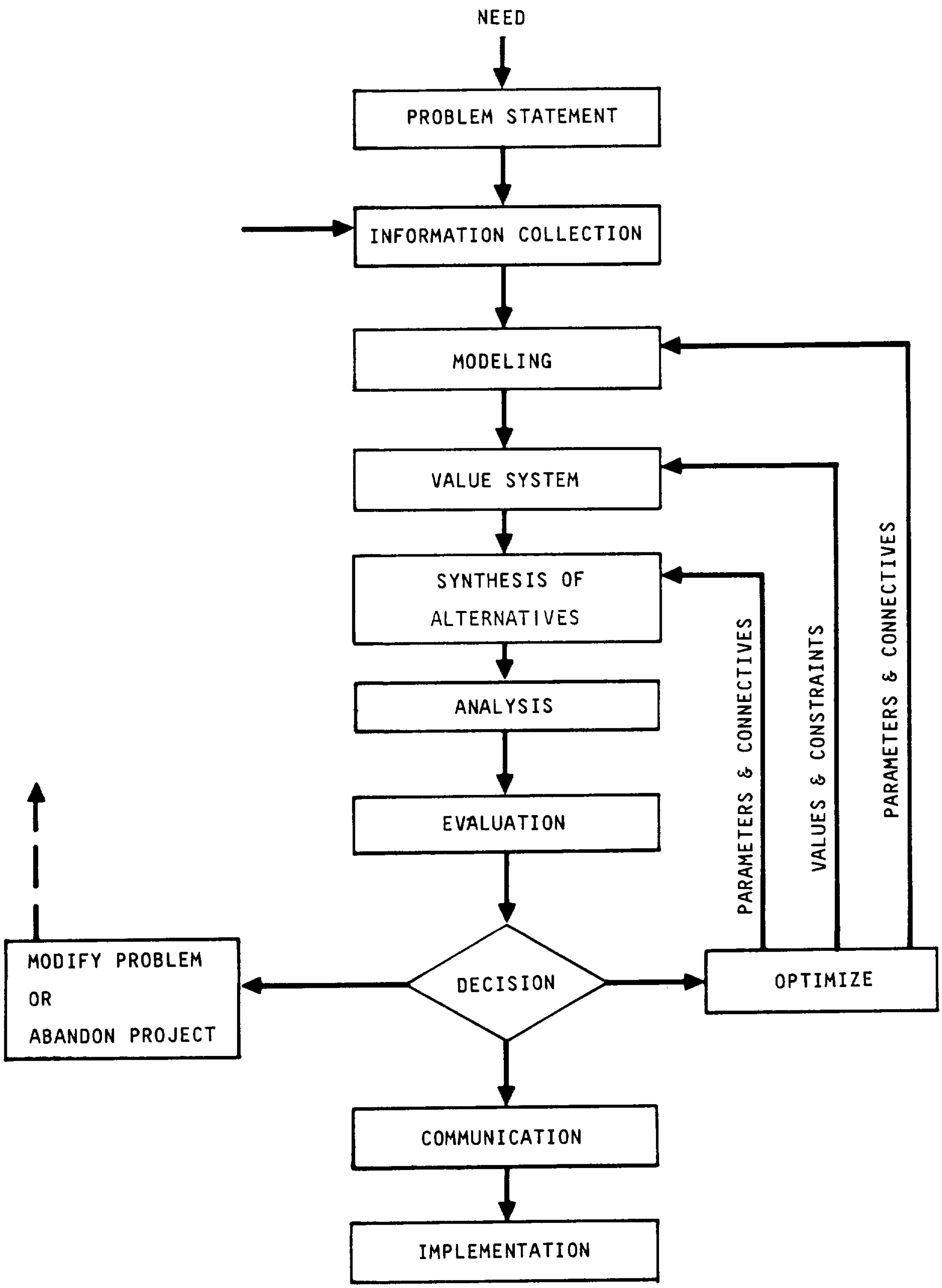

Figure 2-1. ***Design Process***

DESIGN LEVELS

Rockwell, like many aerospace companies, breaks the aircraft design process down into conceptual, preliminary and detailed design phases. The conceptual design phase is further broken down into Design Level I (continuing research), Design Level II (concept formulation) and Design Level III (configuration selection). Figures 2 - 2 through 2 - 4 show flow diagrams for each of these design levels (reference 6). As Design Level III is the first attempt to pick the best configuration (optimize), the rest of this report will focus on that level. The following paragraphs provide a word description of the first five design levels (reference 7).

DESIGN LEVEL I - CONTINUING RESEARCH

The first design level, continuing research of the aircraft system development cycle, is primarily concerned with the problems of developing the technology and methodology required to do the total design and manufacturing functions. In this design cycle activities such as computer-aided design development, technical and functional group research, experimental verification, and other technology advancements are brought to useful application. The types of projects which support Design Level I activities may be found in Rockwell's Independent Research and Development (IR&D) Technical Programs documents.

The IR&D activities are categorized as research and technology programs which encompass the following: design and evaluation, flight science, propulsion, structures and materials, crew systems, subsystems, manufacturing, and Q&RA. These projects are part of the annual planning task and are aligned with the business areas of the particular division. During this level, the continuing research activities are monitored and assimilated so that applicable results that are important to the designer will be available in the computer-aided design environment.

It is important to note that, as technology advancements are made, the methodology must be developed that will enable the new technical element to enter the design process. This entry requires adequate lead-time and resources in the evolutionary process involved with implementing a given technology into an aircraft system. Results of this design level are the methodologies, data base, and basic ideas for various aircraft systems.

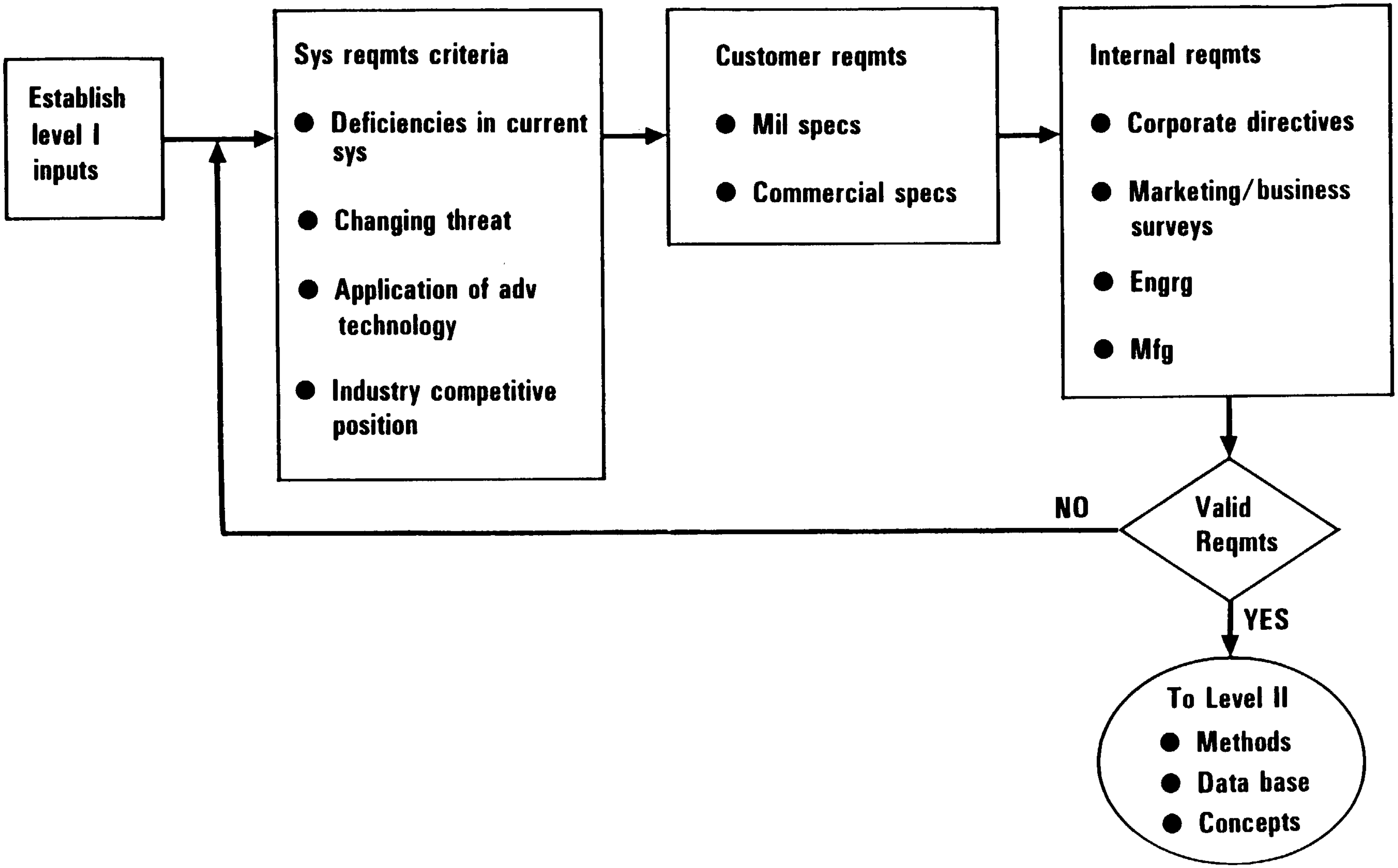

Figure 2-2. ***Flow Network – Design Level I – Continuing Research***

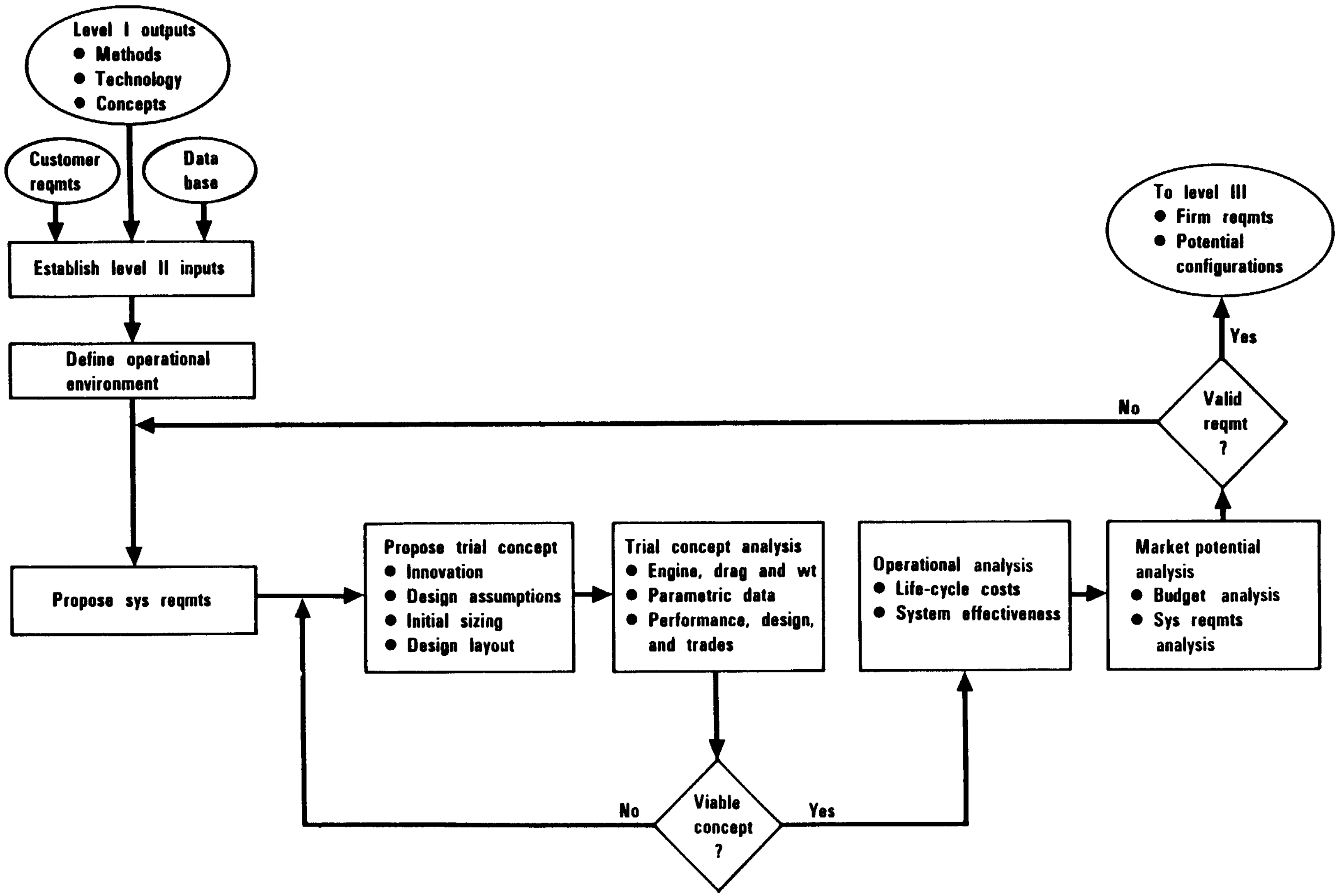

Figure 2-3. ***Flow Network – Design Level II – Concept Formulation***

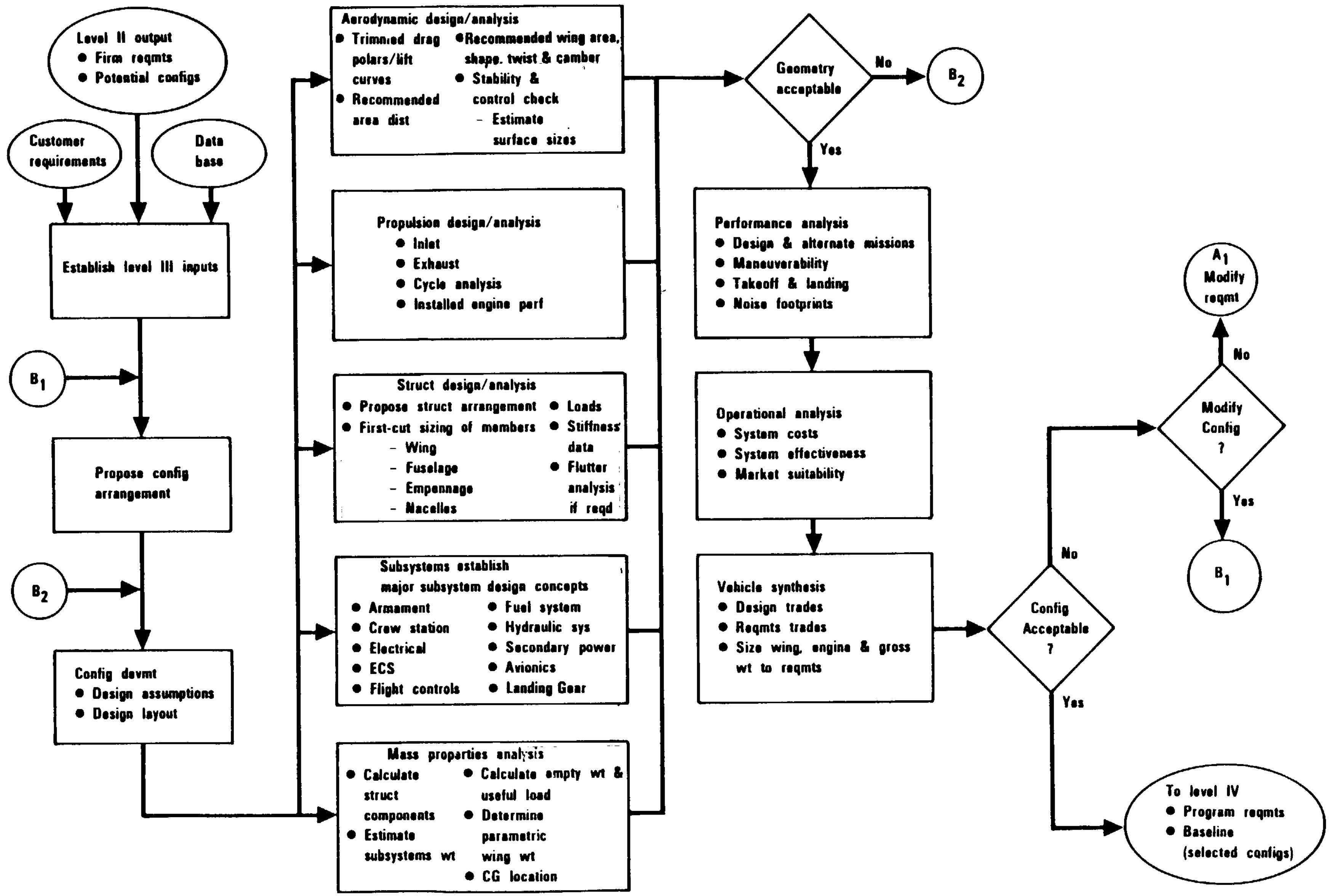

Figure 2.4 ***Flow Network – Design Level III – Configuration Selection***

DESIGN LEVEL II - CONCEPT FORMULATION

Concept formulation is the design phase where the viable configuration arrangements, representative technologies, preliminary data (aerodynamics, propulsion, subsystems, mass properties, etc), preliminary concept layout, and preliminary sizing are established. The purpose of this design level is to establish system requirements and air vehicle concepts that meet a given set of design criteria. The major technical and functional group involvement is indicated in Figure 2-3. Each group activity enters with the appropriate requirement applicable to that group, proceeds with a competitive system evaluation, applies innovative and advanced technologies, and completes an initial sizing activity. These are combined in a candidate concept arrangement.

The activities associated with this design level are initiated by a set of customer requirements defining the aircraft weapons systems objectives. The customer may be either in or outside the division depending on the nature of the problem, and may also have the task of establishing the proper design criteria. Design criteria, whether specified or to be determined, become the first input for developing vehicle concepts that will meet the customer's requirements. At the point where some set of design requirements are established, vehicle concept formulation begins.

Realism of concept validity comes through conceptual configuration layouts and technical specialist involvement. There is no substitute for the experienced engineer or designer. Qualified technical specialists are required to establish the data bases, and interpret the basic methods. Results are then used in configuration conceptual formulation. The interrelationship of disciplinary activities at this design level are depicted in Figure 2-3. Engineers involved at this level may vary from one to 20 engineers and tasks may require 1 day to 2 weeks turn-around time.

DESIGN LEVEL III - CONCEPT SELECTION

The design level involves selecting the best of many concepts that meet the basic requirements established by the customer. The inputs to this phase are the configurations that were developed during the concept formulation activity. During concept selection, an increase in the scope of involvement of the disciplines shown in Figure 2-4 is required. The disciplines involved and activities will in most cases require first order configuration optimization and refinement before the most cost-effective configuration may be selected. The criteria for selection are based on design sensitivities and risk assessment, prior to proceeding to configuration refinement and verification. Outputs of this task are the aircraft system program requirements and the selected baseline configuration. Engineer involvement may vary from 10 to 50 engineers and require 1 to 4 weeks to elapsed time.

DESIGN LEVEL IV - CONFIGURATION REFINEMENT

The objective of this design level is to refine the selected configuration by applying more advanced analytical methods to the vehicle design problem. The design emphasis at this time involves considerably more details within the major disciplines, such as aerodynamics, structures, subsystems, mass properties, etc, to build confidence in the design. Trade studies are conducted against the Level III baseline to provide design visibility and optimize the design. New data developed within the involved disciplines provide the basis for continuing design and result in an optimized configuration (subject to constraints). Engineer involvement may range between 50 to 100 engineers and require 3 to 9 weeks of elapsed time.

DESIGN LEVEL V - CONFIGURATION VERIFICATION

The primary goal of this task is to verify candidate configuration characteristics. These characteristics provide the background for commitment of the product to go ahead with minimum risk. Configuration verification is accomplished by tests and analyses that are within the aircraft weapons systems concepts, and propulsion systems. The design and analysis is as rigorous as possible, with preliminary detail parts designed wherever needed to develop confidence in the overall design.

Engineer involvement varies from 100 to 500 engineers with 2 to 6 months of elapsed time. The output of this level is the design requirements baseline and the proposal that is submitted for the aircraft system. This development, after the customer has made the decision to commit the activity to design manufacturing. It is to be noted, that 1 to 6 months may pass before the completion of Design Level V and the initiation of Design Level VI.

CONCEPTUAL DESIGN OPTIMIZATION (CURRENTLY)

In theory, conceptual designs can be optimized around any figure-of-merit (takeoff gross weight, radar cross section, etc.). In practice conceptual designs are hopefully an optimum compromise of many competing figures-of-merit. The global figure of merit is often mission effectiveness. The mix of figures-of-merit that make up mission effectiveness changes with each conceptual design. Further, for a variety of reasons, the mix is rarely rigorously defined in terms of either actual numbers or relative priorities. There may also not be general agreement among the customer's competing factions of what the mix of figures-of-merit should be. Also, the mix may change significantly over the course of developing the aircraft. Finally, even if the desired mix of figures-of-merit were quantified, it is entirely possible that they all cannot be achieved. At that point difficult decisions must be made as what figures-of-merit to reduce and by how much. All of these uncertainties constitute risks to the success of a program. One of the traditional means for dealing with these risks is to strive to come up with a conceptual design in which the figures-of-merit are not particularly sensitive to each other or to small changes in the design. A design of this type is often not considered an optimum design. And in many ways the goals of risk minimization are the antithesis of the goals of numerical optimization. Perhaps current work by Taguchi (reference 13) in Japan to incorporate off-design risk sensitivity factors into numerical optimization will someday eliminate this dichotomy.

The current design optimization capabilities at Rockwell focus around the more traditional trade studies of thrust to weight ratio versus wing loading, with a goal of minimizing the takeoff gross weight. Imposed on this are the traditional performance constraints of flying a specified mission, energy maneuverability, load factor, takeoff and landing distances, turn rate and radius, acceleration, maximum speed and altitude, range and radius for alternate missions, etc. These trade study results are most often presented graphically. Figure 2-5 is an example of this. The trade study process at Rockwell has been described in NA-85-1543 (reference 6).

For the types of conceptual design trade studies described above, Rockwell has developed a computer system called the Integrated Design and Analysis System (IDAS). This system consists of an integrated suite of computer programs to perform conceptual design geometry definition (layouts), aerodynamics and mass properties analysis, mission performance analysis, vehicle sizing, and a trade study crossplotting capability. A good description of how IDAS is used for conceptual design optimization appears in TFD-85-1453 (reference 8).

Most recently the scope of what figures-of-merit make up mission effectiveness has grown geometrically. For example, low observables, super-maneuverability, hypersonic capability, vulnerability and supportability all vie with the traditional performance figures-of-merit. This has been further complicated by a multitude of new materials, structures, aerodynamics, propulsion and flight controls technologies that need to be assessed in any given conceptual design study. Trade studies which incorporate these figures-of- merit are now being done primarily manually with the unfortunate reality that only a small portion of the potential design space can be explored for optimization. The primary difficulty has been to determine, with confidence, how a new technology affects relevant design variables and how lower level design variables and figures-of-merit affect higher level figures-of-merit. For example, quantitatively how does a new active flexible wing technology affect the drag, weight, and radar cross section of a conceptual design? Then how do these figures-of-merit affect the overall mission effectiveness of the design? A necessary condition for successful optimization is a complete, quantitative understanding of all of these relationships. The challenge here is tremendous.

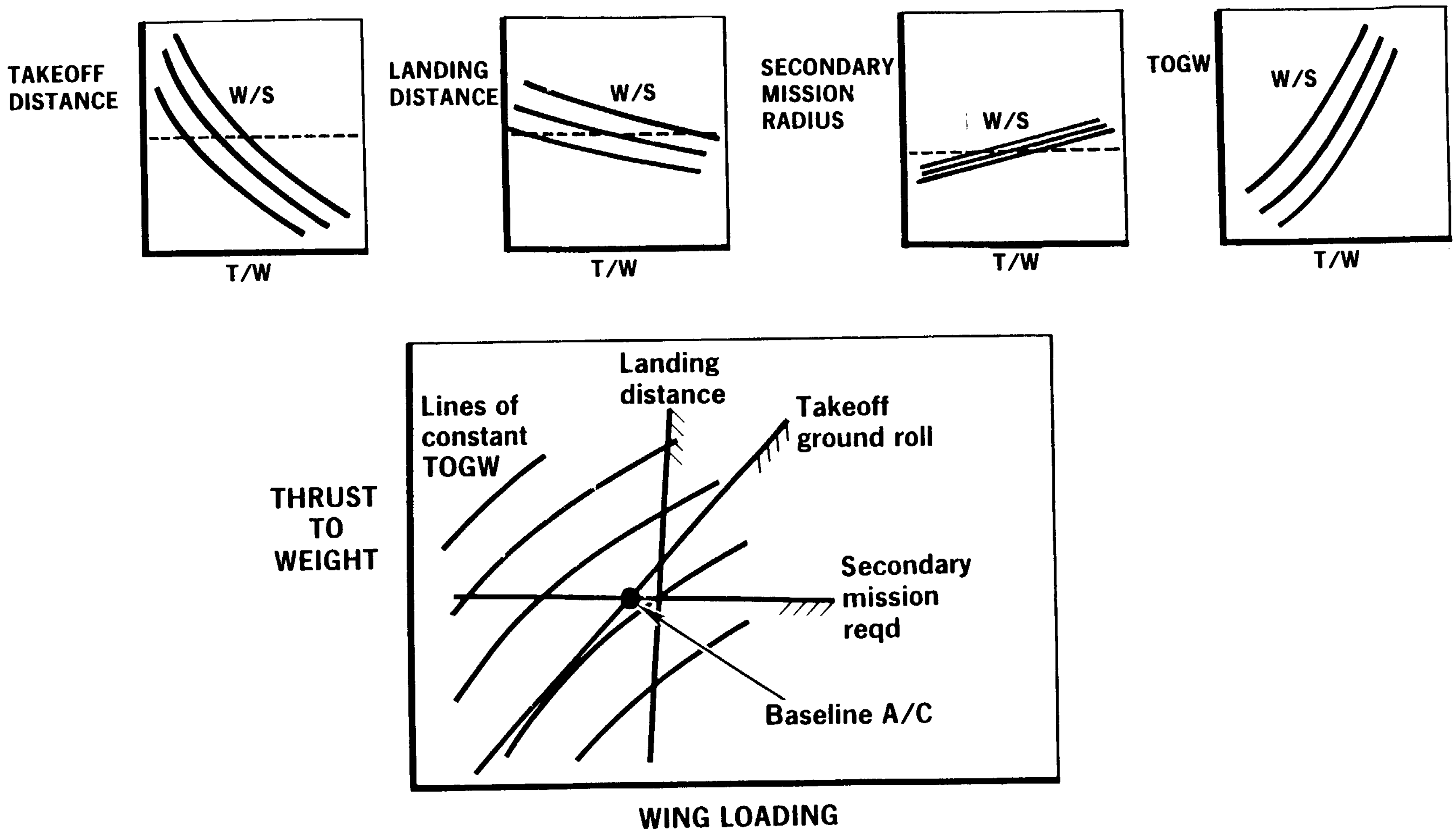

Figure 2-5. ***Graphical Trade Study Example - Sizing for Secondary Mission Requirement***

N^2 DIAGRAM DISCUSSION

In developing the N^2 diagram, the first step was to decompose the design process. A conceptual difficulty appears here. One must consider at least three types of decompositions that are occurring simultaneously. The design process itself decomposes relative to time (e.g., conceptual, preliminary, detail design phases) each of these phases is different, but some of the functional disciplines (or processes) like aerodynamics appear in all design phases. The design itself decomposes hierarchically into smaller and smaller subsystems, which all have to be designed at some time. This functional decomposition does not necessarily correlate to the process decomposition, but is not independent either. Finally, the design and analysis methods within a functional discipline also decompose in some fashion, usually with a loose correlation to the magnitude of input data they require and their computational cost. It is possible to conclude in general that early in the design process (conceptual) one is designing the whole airplane or maybe decomposed down to the major subsystem level and using empirically based methods that require minimal input and are cheap to run.

Unfortunately, with the conceptual design environment the way it is today, there are many exceptions to this general rule. For example, conceptual design of a hypersonic aircraft which uses air-breathing propulsion, because of limited historical data, requires the use of computational fluid dynamics right from the start. Nevertheless, this report makes that simplifying conclusion.

The design process is represented by a single N^2 diagram with conceptual, preliminary, and detailed design being processes along the diagonal. Each of these processes is further decomposed into an N^2 diagram with the design levels along the diagonal. For example, conceptual design is composed of Design Levels I through III. Each of these design levels would then be expressed as an N^2 diagram showing processes within a design level along the diagonal. For this study, only Design Level III will be examined in detail. For Design Level III the processes chosen were directly out of the flow chart in Figure 2-4. As Task 1 is documenting the current design process at Rockwell, it was assumed that the functional decomposition of the design and the decomposition of the methods correlates directly with decomposition of the design process. Therefore, at Design Level III one is concerned about a certain level of functional decomposition (major subsystem) and is applying certain type of design/analysis methods (semi-empirical, linear, static). For Task 2 it will be desirable to make this assumption in order to more effectively deal with current design challenges.

At this level of decomposition the processes may or may not have more than one task that they perform. For example, the performance process consists of tasks to calculate mission performance, maneuverability, and takeoff and landing performance. In order to develop analytical expressions for the relationships between processes, it is necessary to decompose down to the individual task level. Or more precisely, to the individual equation level. At this point one encounters another conceptual difficulty. Even at the conceptual design level the linkages are highly complex and many are not readily posed as closed form analytical equations. Certainly no one at Rockwell has ever attempted to explicitly state all the linkages for the conceptual design process. Dr Sobieski has suggested in NASA-TM-86377 (reference 2) that further functional decomposition of the conceptual design into lower and lower levels of subsystems would make these complex linkages more manageable. The problem here is that, as stated earlier, further functional decomposition is associated with preliminary and detailed design levels. The implication then is that either detailed design must be accomplished prior to conceptual design, or information from a previous detailed design study must be applied to the current conceptual design study. Since conceptual design is characterized by tens of engineers optimizing many designs and detailed design is characterized by hundreds of engineers optimizing a single design, clearly, it is not feasible to do detailed design first. While it is feasible to use the results from a previous detailed design on a current conceptual design study, they may not be applicable due to new technologies or the design being outside the range of validity for the previous coupling expressions (or sensitivities). Indeed the conceptual design process at Rockwell currently makes extensive use of data from previous (actual) aircraft designs in the form of empirical equations (or correlations) for processes such as aerodynamics and mass properties. Finally, one must conclude that if a complete description of all of the linkages from the lowest level subsystems up through the complete aircraft were required then the level of effort would be roughly on the order of detailed design (i.e., hundreds of engineers).

Once the design process has been properly decomposed, it is necessary to identify for each process :

- A statement of the problem or task to be accomplished,
- constraints,
- figures-of-merit,
- control variables,
- input data,
- output data.

Next, all of the relationships between each of the processes must be explicitly and completely defined. The functional approach used by Lockheed in NASA-CR-178239 (reference 9) appears to be most appropriate for this. There will necessarily be a hierarchy of functional relationships. It appears that these functions need to be posed in the form of figure-of-merit = f(control variables, constraints). Further, it appears that input and output data must necessarily be composed only of combinations of figures-of-merit, control variables, and constraints. Finally, once the functional relationships have been identified, then quantitative sensitivity derivatives must be established for each of the relationships. As discussed by Dr Sobieski in Structural optimization: challenges and opportunities (reference 10), the derivatives can be calculated by either finite difference methods or semi-analytical methods. However, for either of these to be applied, the relationships must be expressed in terms of governing equations. It is not yet clear whether it will be possible to derive governing equations for each of the relationships.

Appendix B decomposes the existing Rockwell design process using N^2 diagrams and focuses on the decomposition of conceptual Design Level III. For each of the processes a task statement, control variables, figures-of-merit, inputs, outputs and constraints have been identified. These data have been organized using Apple Macintosh "HyperCard." The hard copy output in this Task 1 report is necessarily limited to one "view" of the data. It is organized such that following the overall N^2 diagram for Design Level III, each function is expanded. Then, following each function is its input data categorized by the process which outputs it.

Section III

TASK 2 RESULTS

INTRODUCTION

PURPOSE

The purpose of Task 2 was to define a modified conceptual design process which incorporates the functional decomposition approach to configuration optimization developed by Dr. Sobieski.

SCOPE

To this end, this section defines a new capability to perform tradeoffs and optimizations of major wing design parameters. The objective function is to minimize design mission take-off gross weight, subject to mission, maneuver, and take-off/landing constraints. This is accomplished by modifying the Parametric Synthesis Module (PSM) of the Rockwell Integrated Design and Analysis System (IDAS) to accept wing aerodynamic and mass properties sensitivity derivatives. Using these sensitivities, PSM generates the needed mission performance parameters, which enables IDAS to perform an optimization of the wing design parameters. This section addresses decomposition of the wing, needed modifications to the IDAS scaling models (geometry, aerodynamics, mass properties), what sensitivity derivatives need to be generated and how the disciplines (aerodynamic and mass properties) will generate them, modifications to IDAS to generate and save the additional problem parameters, how the optimization will be done with IDAS, and any difficulties that must be resolved before this capability can be implemented.

DECOMPOSITION

Figure 3 - 1 shows a functional decomposition of an aircraft, with emphasis on the wing. Figures 3 - 2 and 3 - 3 continue this decomposition for the theoretical trapezoid wing down through structures and aerodynamics, respectively. Figures 3 -4 and 3 - 5 do the same thing for flaps. The concept of functional decomposition, as put forth by Dr. Sobieski, is that the aircraft is decomposed down to the very lowest entity or part (such as a rib or wing skin) by establishing multiple intermediate levels. At a given level, sensitivities (or partial derivatives) are generated for each entity. These sensitivities show how that entity affects the next level up (or parent).

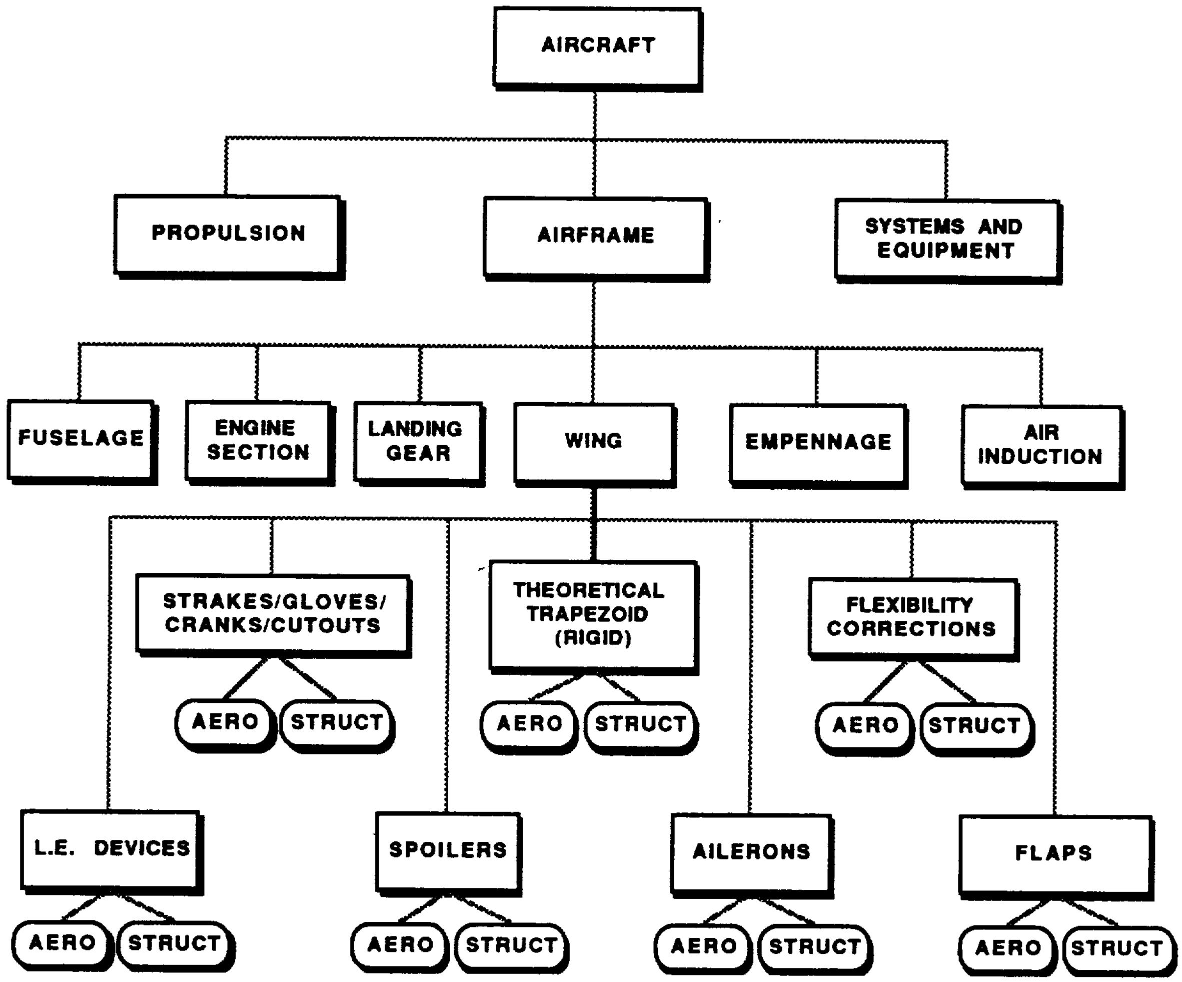

Figure 3 - 1. Functional Decomposition, Top Level

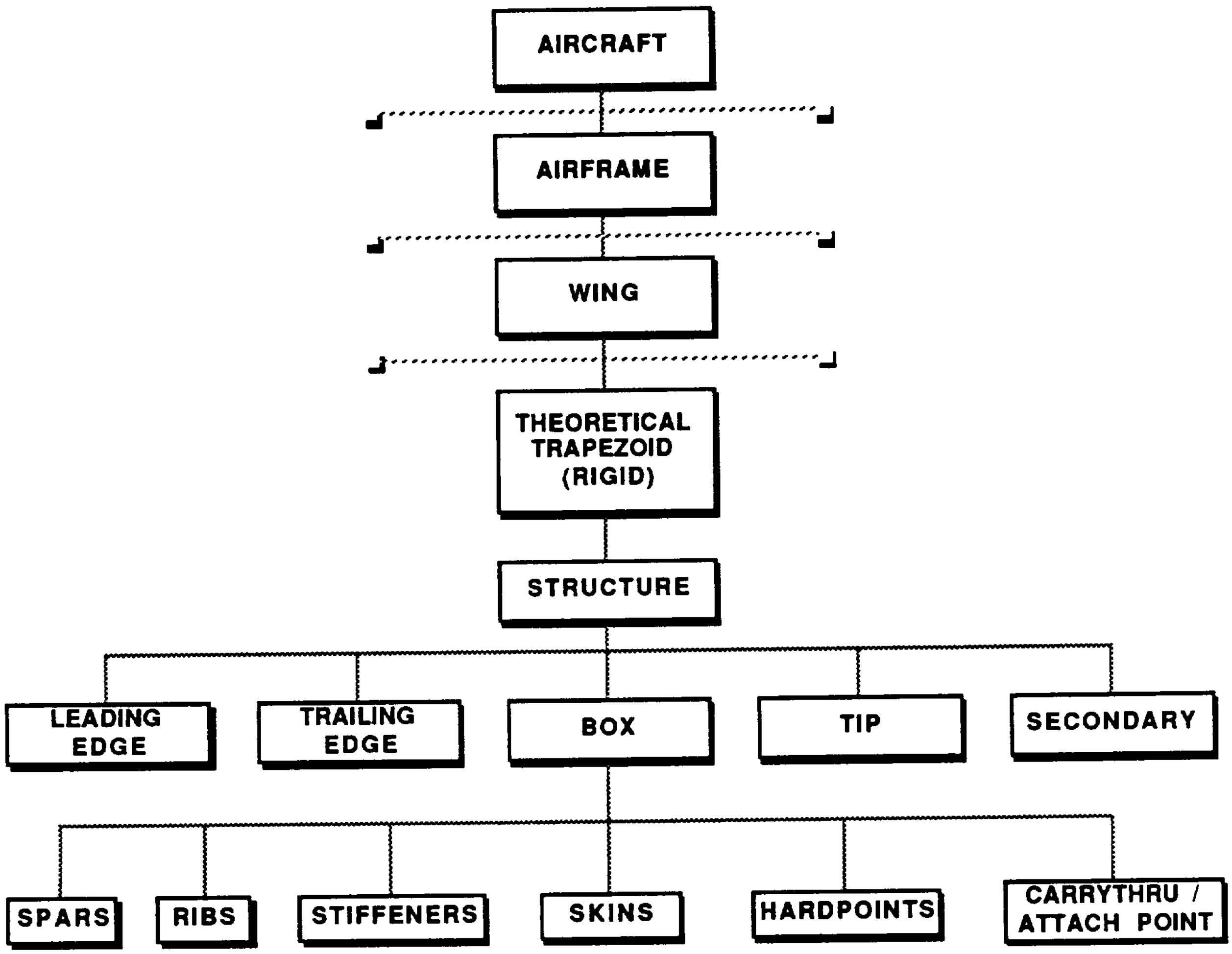

Figure 3 - 2. Functional Decomposition, Theoretical Trapezoid Wing - Structure

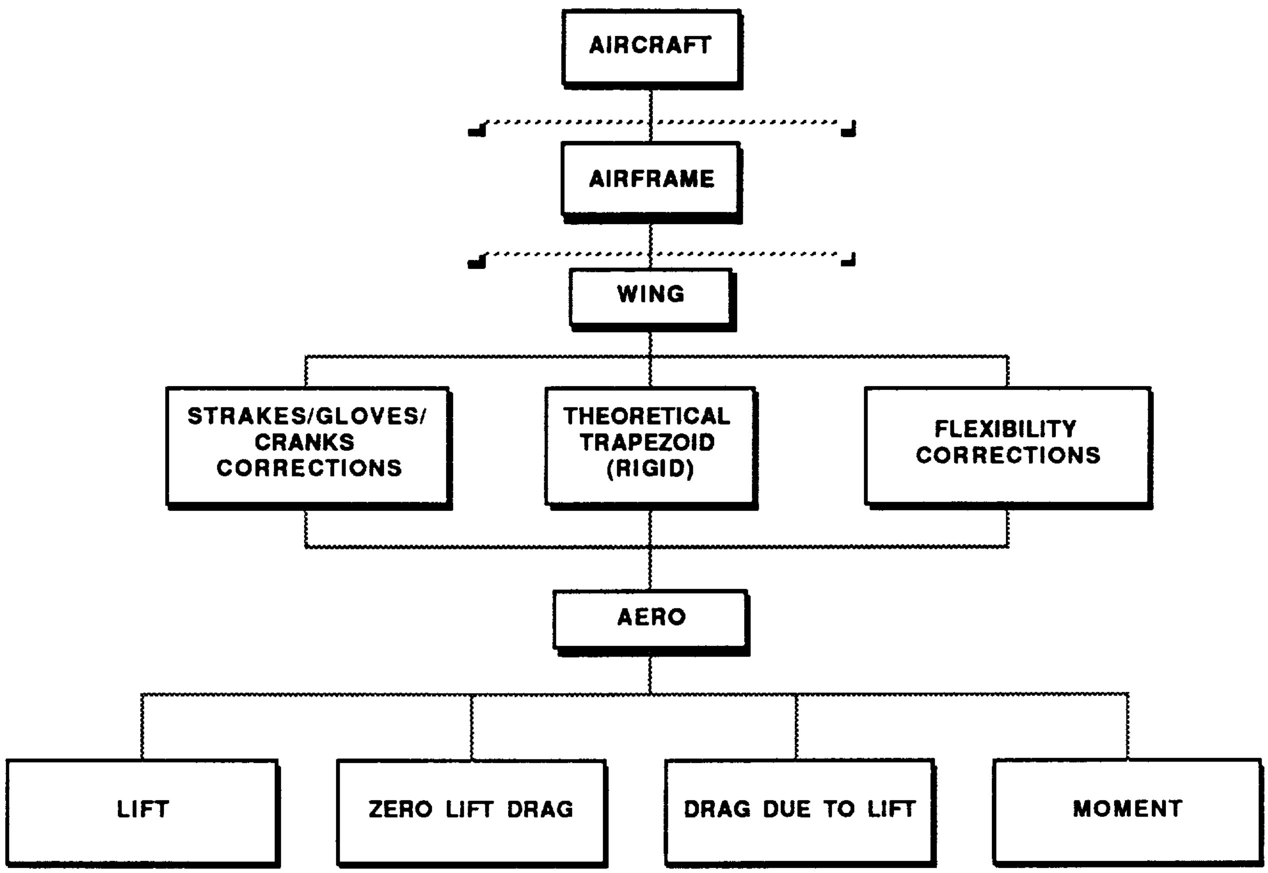

Figure 3 - 3. Functional Decomposition, Theoretical Trapezoid Wing - Aero

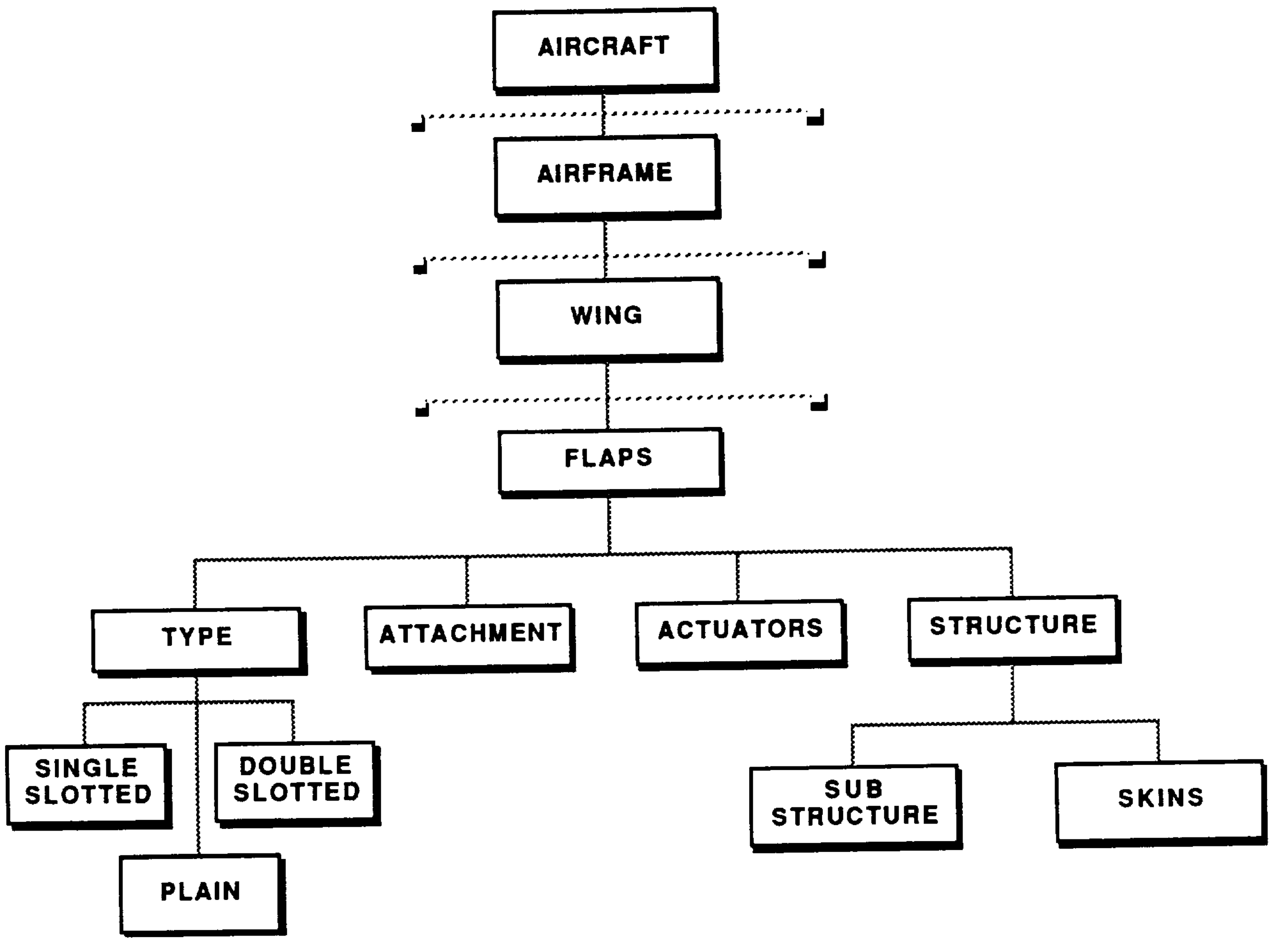

Figure 3 - 4. Functional Decomposition, Flaps - Structure

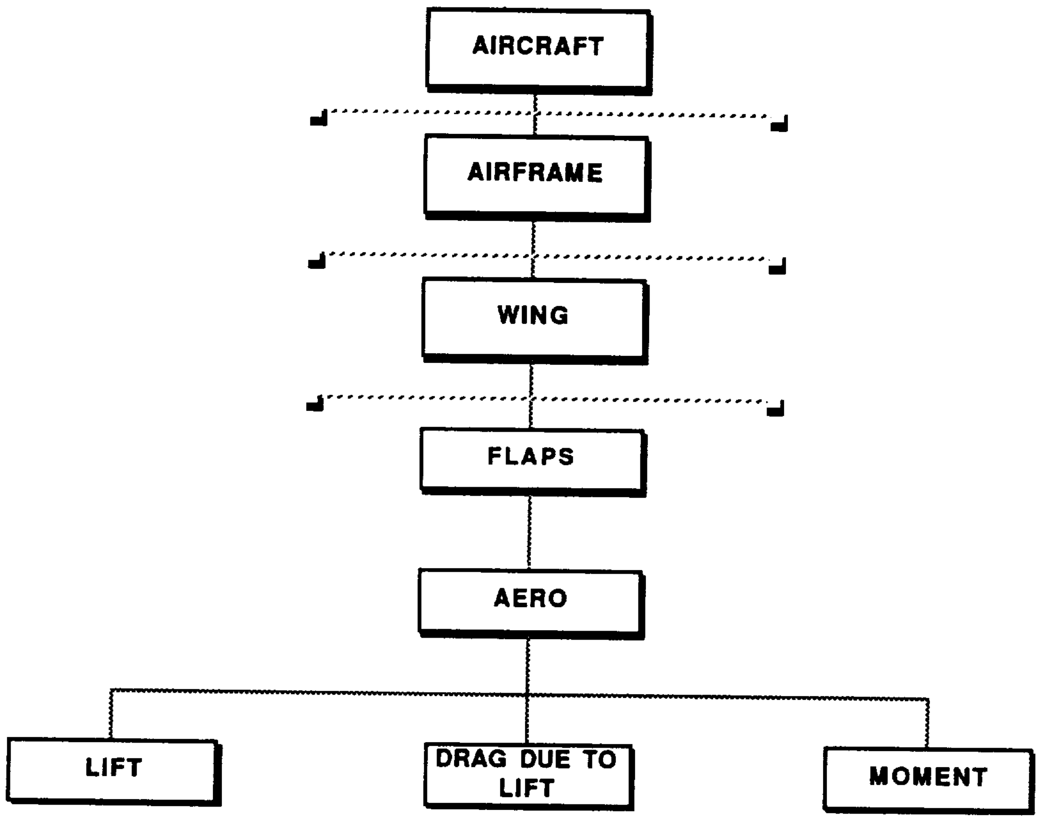

Figure 3 - 5. Functional Decomposition, Flaps - Aero

Using the chain rule, the partial derivatives for each entity can be combined to create a partial derivative for the parent. For example, the wing box weight is composed of cover weight, spar weight, ribs weight, stiffener weight, hardpoints weight, and carry through or attachment point weight. Total wing weight is composed of box weight, leading edge weight, trailing edge weight, tip weight, flap weight, etc. If we wanted to know what a 25 percent increase in rib material thickness did to the overall wing weight, we would use the following expression:

$$\Delta \text{WING WEIGHT} = \frac{\partial \text{WING WEIGHT}}{\partial \text{BOX WEIGHT}} \cdot \frac{\partial \text{BOX WEIGHT}}{\partial \text{RIB WEIGHT}} \cdot \frac{\partial \text{RIB WEIGHT}}{\partial \text{ THICKNESS}} \cdot \Delta \text{THICKNESS}$$

When more than one variable changes simultaneously, this can be represented by summing the partial derivatives. The key to this concept is that one does not have to know a priori how the thickness of a rib affects the total wing weight.

SCALING MODELS

IDAS currently has the capability to size a parametric aircraft (shrink or grow in physical size and weight) so that the fuel required to perform a specified mission is equal to the fuel available within the aircraft. In order to perform the sizing, IDAS contains scaling models for geometry, aerodynamics, weights, propulsion, and trim/control power. At the present time, IDAS has the capability of optimizing thrust-to-weight ratio and wing loading (T/W and W/S), subject to constraints, to minimize take-off gross weight. This means that the geometry and aerodynamic scaling models can scale fuel capacity, engine size and wing area (S), but not the other wing design parameters. The weight model, however, can scale wing weight based on empirical relations which include wing design parameters. This section will discuss modifications to the geometry, weight and aerodynamics scaling models to allow scaling of the following wing design parameters: area (S), aspect ratio (AR), sweep (Λ), taper ratio (λ), thickness ratio (t/c), twist and camber. These wing design parameters are graphically depicted in Appendix C.

GEOMETRY SCALING MODEL

The current geometry model accepts as input the wing design parameters for the base point vehicle. From these, PSM calculates the wing volume and surface area. The wing volume is used to determine how much fuel can be put in wing tanks. As stated earlier, the only wing design parameter that can be scaled is area. Scaling the area results in a new wing volume and surface area. The new model will have to: 1) calculate deltas to the wing design parameters (if a new wing parameter is input) or new wing design parameters (if a delta is input); 2) calculate changes to the wing volume, total aircraft wetted area and frontal area due to changes to the wing parameters; and 3) calculate quarter chord sweep if leading edge sweep in input and vice versa.

WEIGHT SCALING MODEL

As stated above, the wing weight equation currently in PSM is sensitive to all the wing design parameters, except camber and twist. This equation is as follows:

$$\text{WING WEIGHT} = \text{CWCI}\ \text{CCW}\ \text{WK1}\ \text{WK2}\ \text{WK3}\ \text{WGAM} + \text{WWFIX}$$

$$\text{WGAM} = \frac{(\text{WZDES}\ \text{NZ})^{.437}\ \text{QMAX}^{.132}\ \text{SPLAN}^{.758}\ \text{GWTPRE}^{.04}\ (\text{GWARE} / \text{COS4E}^2)^{.60}}{(\text{GWTCR} / \text{COS4E})^{.296}}$$

Where:

WGAM	=	Intermediate calculation, wing weight
WZDES	=	Basic flight design gross weight, lb
NZ	=	Ultimate design load factor
QMAX	=	Maximum dynamic pressure, psf
SPLAN	=	Exposed wing planform area, ft^2 (excludes area covered by fuselage)
GWTPRE	=	Exposed wing taper ratio ($\geq$0.015)
GWARE	=	Exposed wing aspect ratio
COS4E	=	Cosine of the sweep angle of the wing quarter chord
GWTCR	=	Wing thickness to chord ratio at root
CWCI	=	Correlation constant 0.009247
CCW	=	Ratio of weight of advanced material structure to aluminum structure (all aluminum = 1.0)
WK1	=	Wing weight parameter 1.0 if wing mounted on side of body 1.245 if wing with complete carry through structure
WK2	=	Landing gear increment 1.0 if gear mounted on fuselage or nacelle 1.05 if gear mounted on wing
WK3	=	Pivot increment 1.0 if fixed wing 1.17 if variable sweep wing
WWFIX	=	Wing weight penalty, lb (nominally 0.0)

It is based on empirical relations derived from a database of previous aircraft designs. It is possible that the current wing weight equation may not adequately predict the weight sensitivities of new technology wings. Therefore, a new expression appears below which embodies the functional decomposition approach. This approach allows more flexibility in defining sensitivities and may result in more accurate wing weight scaling, assuming it is possible to generate the needed sensitivities. The sensitivities section will describe how Mass Properties will generate these sensitivities.

Two candidate expressions were developed for the wing weight. The first is a simplified expression which does not decompose the wing into its various parts. This is the recommended approach for conceptual design. The simplified expression is as follows:

$$WW = WW_{BP} + \frac{\partial WW}{\partial \psi} \cdot \Delta\psi + \frac{\partial WW}{\partial LOAD} \cdot \Delta LOAD$$

$$+ \frac{\partial WW}{\partial\, NEW\ MATERIAL} \cdot \Delta NEW\ MATERIAL + \frac{\partial WW}{\partial Xcp} \cdot \frac{\partial Xcp^{*}}{\partial \psi} \cdot \Delta\psi + \frac{\partial WW}{\partial Ycp} \cdot \frac{\partial Ycp^{*}}{\partial \psi} \cdot \Delta\psi$$

$$+ WW_{FIX}$$

where:

WW	=	wing weight (for current parametric design)
WW_{BP}	=	wing weight of the base point vehicle
ψ	=	wing design parameters (S, AR, Λ, λ, t/c, twist, camber)
LOAD	=	product of wing design gross weight and ultimate load factor (N_z)
ΔNEW MATERIAL	=	incremental fraction of wing weight due to new material
Y_{cp}	=	spanwise location of center of pressure
X_{cp}	=	chordwise location of center of pressure
WW_{FIX}	=	weight penalty

In this expression the base point weight would include things like types of carry-through structure, landing gear mounting, whether fixed or variable sweep, etc. These fundamental design concepts would not be changed when the wing is scaled, but certain component weights (e.g. landing gear) will change. It is also important to note that a change in design gross weight will change the wing weight, which in turn will change the design gross weight. This requires an iteration loop to converge on the wing weight.

The second, more complex expression decomposes the wing into its parts, and sums the weight sensitivities of each part to arrive at a total wing weight sensitivity. The parts making up the wing are: wing box, leading edge (fixed), trailing edge (fixed), tip, secondary structure, strakes/gloves/cranks, leading edge devices, flaps, spoiler and ailerons. The wing box is further broken down into spars, ribs,

* These sensitivity derivatives come from aerodynamics

stiffeners, wing skins, hardpoints and carry-through/attachment points. The resulting expression (and intermediate expressions are as follows):

$$WW = WW_{BP} + \frac{\partial WW}{\partial BOX} \cdot \Delta BOX + \frac{\partial WW}{\partial LE} \cdot \Delta LE + \frac{\partial WW}{\partial TE} \cdot \Delta TE + \frac{\partial WW}{\partial TIP} \cdot \Delta TIP$$

$$+ \frac{\partial WW}{\partial SS} \cdot \Delta SS + \frac{\partial WW}{\partial GLOVE} \cdot \Delta GLOVE + \frac{\partial WW}{\partial LED} \cdot \Delta LED + \frac{\partial WW}{\partial FLAP} \cdot \Delta FLAP$$

$$+ \frac{\partial WW}{\partial SPOILER} \cdot \Delta SPOILER + \frac{\partial WW}{\partial AILERON} \cdot \Delta AILERON + WW_{FIX}$$

where:

WW	=	wing weight (for current parametric design)
WW_{BP}	=	wing weight of the basepoint vehicle
BOX	=	wing box weight
LE	=	leading edge (fixed) weight
TE	=	trailing edge (fixed) weight
TIP	=	tip weight
SS	=	secondary structure weight
GLOVE	=	strake/glove/crank weight
LED	=	leading edge device weight
FLAP	=	flap weight
SPOILER	=	spoiler weight
AILERON	=	aileron weight
WW_{FIX}	=	weight penalty

Since none of these derivatives directly expresses sensitivities due to design variables, additional intermediate expressions are needed, as follows:

$$\Delta BOX = \frac{\partial BOX}{\partial SPAR} \cdot \frac{\partial SPAR}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial RIBS} \cdot \frac{\partial RIBS}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial STIFFENERS}$$

$$\cdot \frac{\partial STIFFENERS}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial SKINS} \cdot \frac{\partial SKINS}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial HARDPOINTS}$$

$$\cdot \frac{\partial HARDPOINTS}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial CARRYTHROUGH} \cdot \frac{\partial CARRYTHROUGH}{\partial \psi} \cdot \Delta\psi$$

$$+ \frac{\partial BOX}{\partial X_{cp}} \cdot \frac{\partial Xcp^*}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial Y_{cp}} \cdot \frac{\partial Ycp^*}{\partial \psi} \cdot \Delta\psi + \frac{\partial BOX}{\partial LOAD} \cdot \Delta LOAD$$

*These sensitivity derivatives come from aerodynamics

$$\Delta LE = \frac{\partial LE}{\partial \psi} . \Delta\psi \qquad \Delta LED = \frac{\partial LED}{\partial \psi} . \Delta\psi$$

$$\Delta TE = \frac{\partial TE}{\partial \psi} . \Delta\psi \qquad \Delta FLAP = \frac{\partial FLAP}{\partial \psi} . \Delta\psi$$

$$\Delta TIP = \frac{\partial TIP}{\partial \psi} . \Delta\psi \qquad \Delta SPOILER = \frac{\partial SPOILER}{\partial \psi} . \Delta\psi$$

$$\Delta SS = \frac{\partial SS}{\partial \psi} . \Delta\psi \qquad \Delta AILERONS = \frac{\partial AILERONS}{\partial \psi} . \Delta\psi$$

$$\Delta GLOVE = \frac{\partial GLOVE}{\partial \psi} . \Delta\psi$$

where:

ψ	=	wing design parameters (7)
SPAR	=	weight of wing spars
RIBS	=	weight of wing ribs
STIFFENERS	=	weight of wing stiffeners
SKINS	=	weight of wing skins
HARDPOINTS	=	weight of wing hardpoints
CARRYTHROUGH	=	weight of wing carry-through structure or attachment points
LOAD	=	product of wing design gross weight and ultimate load factor (N_Z)

In developing this second expression, it was assumed that the sensitivities of the wing box components (skins, ribs, etc.) to the wing design variables implicitly accounted for the weight change required for the wing to maintain the same load carrying capability. For example, a decrease in the wing t/c design variable would likely cause the wing skins to become thicker (and hence heavier) in order for the wing to have the same load carrying capability. If this assumption is not valid, then load sensitivities will have to be explicitly included with each of the wing box components. This would increase the number of sensitivities that would have to be calculated. It was also assumed that load (product of design gross weight and ultimate load factor) was a design variable independent from the other wing design variables. The validity of this assumption will require further investigation. Finally, it may be possible to assume that the movable wing surfaces weights (flaps, etc.) and gloves/strakes weights are not sensitive to changes in basic wing design parameters (with the exception of wing area, to which they are clearly sensitive). This would significantly reduce the number of sensitivities to be calculated for the more detailed expression.

Since there are seven wing design parameters, performing an optimization or trade study on all the wing design parameters will require the Mass Properties group to calculate 11 sensitivity derivatives for the simple (preferred) expression or 124 for the more complex expression (plus 14 aero sensitivity derivatives in either expression). This latter number of sensitivity derivatives is more appropriate to late preliminary and detail design phases than it is to conceptual design. Therefore, the remainder of this report will use the simplified expression.

AERODYNAMIC SCALING MODEL

PSM accepts aerodynamic coefficient (C_L, C_D, etc.) for the basepoint vehicle in the form of tables, usually as a function of Mach number. The current PSM aerodynamic scaling model allows wing area to be scaled, but not the other wing design parameters. This current capability will be retained. This section develops new expressions to be added to PSM that will allow the aerodynamic coefficients to be scaled with respect to the rest of the wing design parameters. These expressions will incorporate sensitivity derivatives for each of the aerodynamic coefficients needed by PSM. The sensitivities section will describe how Flight Sciences will generate these sensitivities.

Zero Lift Drag

PSM calculates a skin friction drag coefficient (C_{DSF}) based on vehicle length and the wetted area of the current parametric vehicle. This C_{DSF} is indexed from an input baseline value. PSM scales the wetted area as a result of scaling the wing area, scaling the nacelles to accommodate a higher or lower thrust engine, or scaling the fuselage to accommodate more or less fuel. PSM also scales drag coefficients as a result or increases/decreases in engine sizes (C_{DBASE}, $C_{DBOUNDARY\ LAYER\ DIVERTER}$, C_{DWAVE}). Finally, PSM will scale the C_{DWAVE} of the fuselage due to changes in the frontal area (while maintaining a constant fineness ratio). Any modification to the aerodynamic scaling model to accommodate wing design parameters must retain the current fuselage/nacelle scaling capability. This requirement imposes the following constraints on the new aerodynamic scaling model.

1. Proposed approach: skin friction drag coefficient will continue to be calculated internal to PSM based on vehicle length and wetted area, but the expression will be modified to break out wing skin friction drag separately, as follows:

$$C_{D_{SF}} = C_{D_{SF_{WING}}} + C_{D_{SF(TOTAL - WING)}}$$

$$C_{D_{SF_{WING}}} = K_{ALT} \cdot C_{f\,WING} \cdot FF_{WING} \cdot SWET_{WING}/S$$

$$C_{D_{SF(TOTAL - WING)}} = K_{ALT} \cdot C_{f\,(TOTAL - WING)} \cdot FF_{(TOTAL - WING)} \cdot SWET/S$$

where:

$C_{D_{SF}}$	=	skin friction drag coefficient
K_{ALT}	=	constant used to index friction drag at the reference mach number and altitude
$C_{f\,WING}$	=	friction coefficient - f (mach, altitude, mean aerodynamic chord length, wall temperature, emittance, roughness)
$C_{f\,(TOTAL - WING)}$	=	friction coefficient - f (mach, altitude, weighted average characteristic length, wall temperature, emittance, roughness)
FF_{WING}	=	form factor for wing (currently f (t/c))
$FF_{(TOTAL - WING)}$	=	form factor for rest of aircraft
$SWET_{WING}$	=	wing wetted area (approximately twice the exposed wing area)
SWET	=	total wetted area minus wing wetted area
S	=	reference wing area

Other candidate approaches will be evaluated during development. These include: (1) incorporating the sensitivity derivative $\partial C_{D_{SF_{WING}}} / \partial \psi$, which will be determined by Flight Sciences. The main issue here is that this relationship is non-linear with respect to t/c. (2) A complete skin friction drag component buildup, instead of lumping the rest of the aircraft together. The main issue here is determining the most appropriate form factor(s) and characteristic length(s).

2. Zero lift drag coefficient (C_{D_0}) will continue to be decomposed into components, as is currently in PSM.

$$C_{D_0} = C_{D_{WAVE}} + C_{D_{SF}} + C_{D_{BASE}} + C_{D_{BOUNDARY\ LAYER\ DIVERTER}}$$

(Note: 1. PSM terminology uses $C_{D_{PARASITE}}$ for C_{D_0},
2. $C_{D_{BASE}}$ and $C_{D_{BOUNDARY\ LAYER\ DIVERTER}}$ are not changed by wing design parameters)

3. Wave drag coefficient currently has two scaling options external and internal. The external option requires tables which are functions of mach number, wing area, fuselage fineness ratio, engine volume. The internal option is a function of the ratio of total cross section area/wing area. The internal option will be modified to break out the wing contribution, as follows:

$$C_{DWAVE} = [C_{DWAVE\ (TOTAL\ -\ WING)BP}] \cdot [S/S_\pi]_{BP} \cdot [S_\pi/S]$$

$$+ [C_{DWAVE(WING)BP} + \frac{\partial C_{DWAVE(WING)}}{\partial \psi} \cdot \Delta\psi] \cdot [S/S_{EXP}]_{BP} \cdot [S_{EXP}/S]$$

where:

C_{DWAVE}	=	wave drag coefficient (of current parametric vehicle)
$C_{DWAVEBP}$	=	wave drag coefficient (of basepoint vehicle)
ψ	=	wing design parameters (except wing area)
S	=	wing reference area
S_π	=	total aircraft cross section area minus wing frontal area
S_{EXP}	=	exposed wing area

4. Drag divergence mach number ($C_L = 0$) is currently not scaled, but wing parameters such as sweep will significantly affect it. So a new expression must be developed, as follows:

$$MDD_0 = MDD_{0BP} + \frac{\partial MDD_0}{\partial \psi} \cdot \Delta\psi$$

where:

MDD_0	=	drag divergence mach number at $C_L = 0$ (of current parametric vehicle)
MDD_{0BP}	=	drag divergence mach number at $C_L = 0$ (of basepoint vehicle)
ψ	=	wing design parameters (except wing area)

note: a correction will be added to this expression for $C_L > 0$ in the next section

Drag Due to Lift

PSM can accept as input a total drag polar $C_{DTOTAL} = f(CL, mach)$, drag due to lift $C_{DL} = f(C_L, mach)$, or drag due to lift factor $K = f(C_L, mach)$. Additional input includes lift coefficient for minimum drag CLK = f(mach) and drag due to lift coefficient at minimum drag CDK = f(mach). Internally PSM works with the drag due to lift factor only. So it converts an input C_{DTOTAL} or C_{DL} to a K. PSM does not, however, scale K. It would be possible to develop a scaling expression for either C_{DL} or K. However, the expressions developed here will be using K with the sensitivity derivatives being a function of both mach and C_L. The various merits of either approach will be revisited before the actual implementation is done. The expressions are:

$$C_{D_L} = K\,(C_L - CLK)^2 + CDK$$

$$K = K_{BP} + \frac{\partial K}{\partial \psi} \cdot \Delta\psi$$

$$CLK = CLK_{BP} + \frac{\partial CLK}{\partial \psi} \cdot \Delta\psi$$

where:

C_{D_L}	=	drag due to lift coefficient
CLK	=	lift coefficient for minimum drag
CDK	=	drag due to lift coefficient at minimum drag
K_{BP}	=	drag due to lift factor for base point vehicle
ψ	=	wing design parameter (except wing area)

Compressible Drag Due-to-Lift Correction

PSM calculates the drag divergence mach number as a function of C_L . This is currently not sensitive to wing design parameters, so a new expression must be developed. The expression for MDD is shown below and includes the $C_L = 0$ portion developed above. The sensitivity derivatives ($\partial MDD/\partial\psi$) will also be a function of C_L. The derivative $dMDD/dC_L$ is presently constant and the change of that derivative with respect to the design variables (ψ) may be small enough to neglect. Further research will be needed to make that determination and the expression below includes its effect.

$$MDD = MDD_0 + \left[\frac{dMDD}{dC_L}_{BP} + \frac{\partial(dMDD/dCL)}{\partial\psi} \cdot \Delta\psi \right] \cdot C_L$$

Total Drag

The current expression for total drag coefficient will be used:

$$CD_{TOTAL} = CD_0 + CDK + K(CL - CLK)^2$$

with the added expression:

$$CDK = CDK_{BP} + \frac{\partial CDK}{\partial \psi} \cdot \Delta\psi$$

Lift

PSM does not currently scale any of the lift related parameters. Therefore, the following relations will be developed.

$$C_{L0} = [C_{L0}]_{BP} + \frac{\partial C_{L0}}{\partial \psi} \cdot \Delta\psi$$

$$C_{L\alpha} = [C_{L\alpha}]_{BP} + \frac{\partial C_{L\alpha}}{\partial \psi} \cdot \Delta\psi$$

$$C_{LMAX} = [C_{LMAX}]_{BP} + \frac{\partial C_{LMAX}}{\partial \psi} \cdot \Delta\psi$$

$$\alpha_{C_{LMAX}} = [\alpha_{C_{LMAX}}]_{BP} + \frac{\partial \alpha_{C_{LMAX}}}{\partial \psi} \cdot \Delta\psi$$

where:

C_{L0}	=	lift coefficient at $\alpha = 0$
$C_{L\alpha}$	=	lift curve slope
C_{LMAX}	=	maximum lift coefficient
$\alpha_{C_{LMAX}}$	=	α at maximum lift coefficient
ψ	=	wing design parameters (except wing area)

Finally, two aerodynamic sensitivity derivatives are necessary for weight analysis. Those are:

$$\frac{\partial X_{CP}}{\partial \psi} \quad \text{and} \quad \frac{\partial Y_{CP}}{\partial \psi}$$

where Y_{cp} is the spanwise location of the center of pressure and X_{cp} is the chordwise location of the center of pressure.

If we have a total of six wing design parameters (AR, Λ, λ, t/c, twist, camber) for each of the sensitivity derivatives, a total of 78 derivatives will have to be calculated for the aerodynamic scaling model. Here a significant complication arises in that all these derivatives are also a function of mach number (except MDD, which instead is a function of C_L). This becomes particularly apparent if we look at $\partial C_{D_{WAVE}}/\partial \Lambda$ in the region of mach 0.8 to 1.5. At the low mach numbers the sensitivity would be zero. At the high mach numbers the sensitivity would be higher, but around mach 1 the sensitivity would be very high. There appears to be two approaches to deal with this complication. First, would be to calculate sensitivity derivatives at each mach number where the base point aerodynamic coefficients are calculated. PSM could handle this in the form of an input table, but it would result in Flight Sciences having to generate quite a lot of sensitivity derivatives. The second approach would be to calculate another partial derivative with respect to mach number for each aerodynamic coefficient and incorporate it into the expressions. An example of this is the wave drag coefficient expression:

$$C_{D_{WAVE}} = [C_{D_{WAVE}}]_{BP} + \frac{\partial C_{D_{WAVE}}}{\partial \psi} \cdot \Delta\psi + \frac{\partial C_{D_{WAVE}}}{\partial MACH} \cdot \Delta MACH$$

Two problems immediately become apparent. First is that $C_{D_{WAVE}}$ is strongly nonlinear with respect to mach number. Second, $\partial C_{D_{WAVE}}/\partial MACH$ is not really independent of ψ (e.g. Λ). So a different value of $\partial C_{D_{WAVE}}/\partial MACH$ would be needed for each different value of the various design parameters. Figure 3 - 6 illustrates this coupled relationship, as well as the nonlinearity of the data (reference 11). Perhaps some advanced mathematical techniques can be applied to solve this problem of mach number dependency. For now, the recommended approach is for Flight Sciences to generate the aerodynamic sensitivities at all relevant mach numbers (and C_L values, as appropriate).

SENSITIVITIES

The previous section identified which aerodynamic and weight sensitivities were needed by PSM, without regard for relative importance of sensitivities to the various wing design parameters. The number of sensitivities required for the weight scaling model (11) is probably manageable; however, the number of sensitivities required for the aerodynamic scaling model (78 or more) may be difficult to generate on a routine basis. One approach to alleviate this problem would be to eliminate those sensitivities which are not important. This section will discuss weight, aerodynamic and performance sensitivities. It will identify what sensitivities need to be generated and how they will be generated. The Aerodynamic Sensitivities section will also discuss the relative importance of the sensitivities.

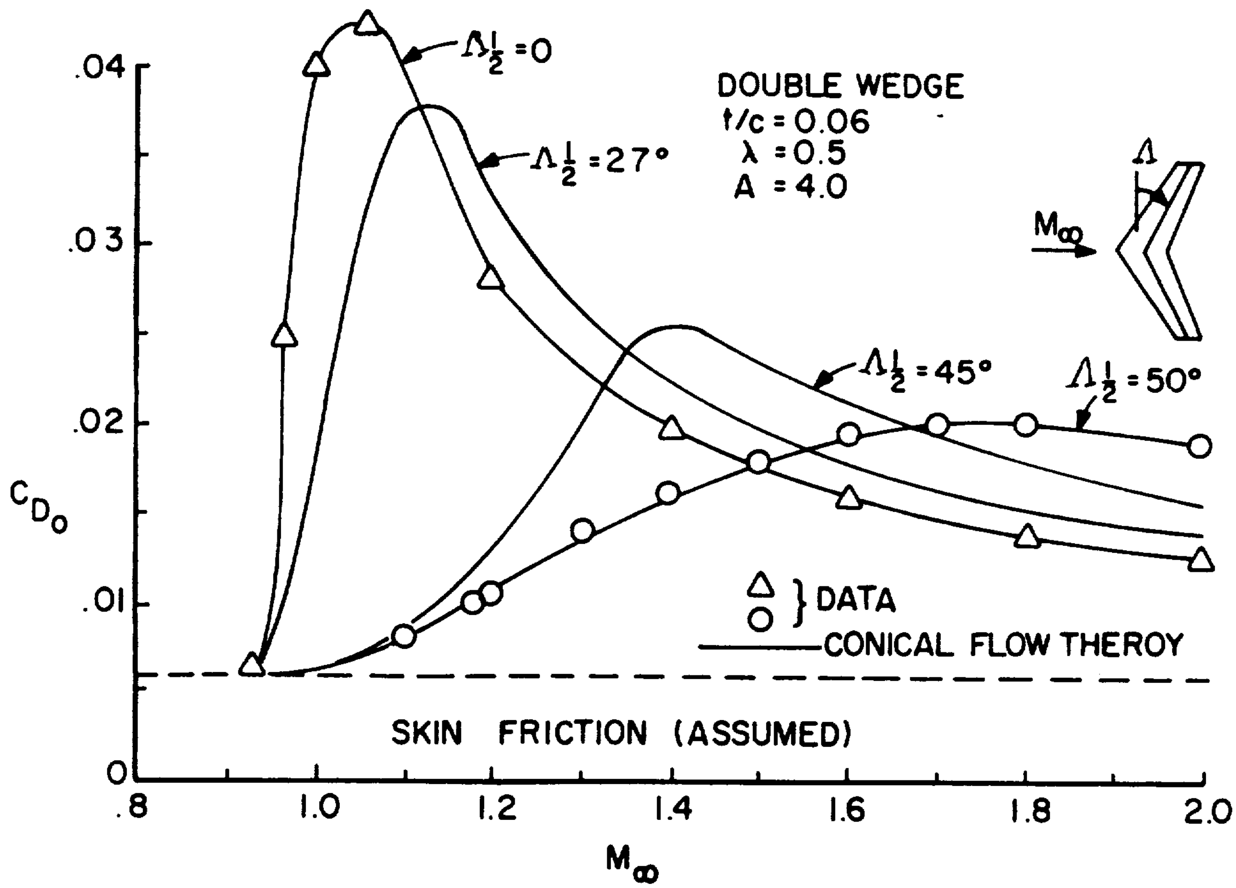

Figure 3 - 6. Effect of Wing Sweep on C_{D_0}

WEIGHT SENSITIVITIES

As identified in the previous section, the following weight sensitivity derivatives need to be generated for the simplified wing weight expression:

$$\frac{\partial WW}{\partial \psi}, \frac{\partial WW}{\partial LOAD}, \frac{\partial WW}{\partial NEW\ MATERIAL}, \frac{\partial WW}{\partial X_{cp}}, \frac{\partial WW}{\partial Y_{cp}} \qquad \text{(total of 11)}$$

where:

W W	=	wing weight
ψ	=	wing design parameters (S, AR, Λ, λ, t/c, twist, camber)
LOAD	=	product of wing design gross weight and ultimate load factor (N_z)
NEW MATERIAL	=	new wing structural material (from base point)
Ycp	=	spanwise location of center of pressure
Xcp	=	chordwise location of center of pressure

In the conceptual phases of vehicle design, structure weight has traditionally been one of the more difficult of the sensitivity parameters to determine. Empirical weight estimation methods have been employed in the concept optimization process. The accuracy of these empirical derivations decreases rapidly as one departs from the statistical data base. Furthermore, statistical formulations are dependent on the existing population of data points which limits the ability to assess the merits of new materials or load changes due to shifts in center of pressure etc. Therefore, development of the 11 desired sensitivity derivatives will require use of analytical models and approaches. Wing weight derivatives with respect to area, aspect ratio, thickness ratio, taper ratio, sweep, load factor and material can then be derived directly. It is envisioned that wing weight derivatives with respect to twist and camber will be derived as partial derivatives with respect to center of pressure (spanwise/chordwise load distribution) taking the form shown below:

$$\Delta \text{ Wing Weight} = \frac{\partial \text{Wing Weight}}{\partial CP} \cdot \frac{\partial CP}{\partial \psi} \cdot \Delta\psi$$

where:

CP	=	the center of pressure $\left(\frac{Ycp}{b/2}\right)$
ψ	=	design parameter (twist, camber)

and $\partial CP/\partial\psi$ is generated by aerodynamics

Generation of the wing weight sensitivity derivatives will consist of the following:

1. Selection/development of analytical process
2. Calibration/correlation of the analytical process
3. Employ analytical process to generate parametric data base and develop approach for extracting partial derivatives
4. Validate the proposed conceptual design approach by comparison with point design analysis results.

The Structural Weight Estimation Program (SWEEP) is an analytical procedure developed for utility in the concept formulation and validation phases of preliminary design. The approach to wing weight estimation is based on a multi-station analysis/sizing of structural elements. The process evaluates a spectrum of vehicle flight and ground loading conditions and synthesizes elements to satisfy these loadings based on material properties, temperature, type of construction and fabrication constraints, geometry, strength, local and general stability, lifting surface flutter and manufacturing requirements. SWEEP can be used to generate the sensitivity derivatives by performing parametric trades over the range of interest.

In most cases, time lines associated with concept formulation studies conflict with the utilization of analytical tools. Therefore, at times, program goals will preclude the use of SWEEP and related methods to develop sensitivity derivations. A predeveloped data base can be employed in these instances to generate the required sensitivities. This data base will consist of parametric matrices generated analytically about existing statistical points and projected alternate planforms.

AERODYNAMIC SENSITIVITIES

As identified in the previous section, the following aerodynamic sensitivity derivatives need to be generated:

$$\frac{\partial CDWAVE(WING)}{\partial\psi}, \frac{\partial MDD}{\partial\psi}, \frac{\partial (dMDD/dCL)}{\partial\psi}, \frac{\partial K}{\partial\psi}, \frac{\partial CLK}{\partial\psi}, \frac{\partial CDK}{\partial\psi}, \frac{\partial CDSF}{\partial\psi}$$

$$\frac{\partial C_{L_0}}{\partial\psi}, \frac{\partial C_{L_\alpha}}{\partial\psi}, \frac{\partial C_{L_{MAX}}}{\partial\psi}, \frac{\partial \alpha_{C_{L_{MAX}}}}{\partial\psi}, \frac{\partial Xcp}{\partial\psi}, \frac{\partial Ycp}{\partial\psi} \quad \text{(Total of 78)}$$

where:

CDWAVE(WING) = wave drag coefficient of wing

ψ = wing design parameters (AR, Λ, λ, t/c, twist, camber),except S

MDD	=	drag divergence mach number f (C_L)
$dMDD/dC_L$	=	slope of MDD curve
K	=	drag due to lift factor f (C_L)
CLK	=	lift coefficient for minimum drag
CDK	=	drag due to lift coefficient at minimum drag
CDSF	=	skin friction drag coefficient
C_{L0}	=	lift coefficient at $\alpha = 0$
$C_{L\alpha}$	=	lift curve slope
C_{LMAX}	=	maximum lift coefficient
αC_{LMAX}	=	α at maximum lift coefficient
Ycp	=	spanwise location of center of pressure
Xcp	=	chordwise location of center of pressure

All of these sensitivities are functions of mach number (except MDD), which could result in an order of magnitude increase in the number of sensitivities that must be generated.

Wing geometry variables, and the aerodynamic parameters needed for sizing , are shown in Table 3-I. The X (longitudinal) and Y (spanwise) location of the center of pressure, shown in the last two columns, are needed in order to determine the effect of wing loads on structural weight. (Note that the total load as expressed in g's does not vary with wing size.) The relative importance of each configuration variable is shown in each case. All of the dependent variables are also dependent on free stream mach number, and certain of the variables (K, MDD, Xcp, Ycp) may also vary with lift coefficient.

The aerodynamic data required for the sizing process are currently estimated using several sources. The lift curve slope, drag due-to-lift, and loads data are initially estimated using a linear panel method. Usually, the twist and camber are then determined using a linear optimization code, which will alter the drag due-to-lift and the loads. Volume wave drag is determined using a linear method when possible (depending on wing sweep and mach number). For some conditions, the linear answer must be adjusted empirically, or estimated using a nonlinear formulation. The limit lift coefficient and its angle of attack may arise from wing stall, or may result from other limitations such as buffet or loss of stability. These parameters are also usually determined empirically.

Since none of the dependent variables are computed from closed-form algebraic algorithms, partial derivatives with respect to the independent variables cannot be formed explicitly. In such cases, the sensitivities can be determined numerically by computing results for the baseline case and a series of variations around the baseline. This process will have to be repeated at a sufficient number of mach numbers to accurately describe the flight envelope. As indicated in the preceding paragraph, provisions for variation with lift coefficient should be included in a few cases. Many of the variables also exhibit nonlinear behavior in some regions of interest. Therefore, the range of parametric variations must be limited somewhat, or large errors will be introduced, or else provision for nonlinear variations will have to be included in the method.

The process as outlined above would be very computationally intensive, which would defeat the purpose of the proposed overall approach. Some alternate schemes which may reduce the computational workload with minimal loss of accuracy are outlined in the following paragraphs.

Table 3 - I shows that several of the required terms are of secondary importance, or have negligible effects. Through case-by-case inspection, some terms could be limited to linear variations, and others could be eliminated altogether.

Another possible alternative would be to generate data for only the baseline configuration using the methods described above. Empirical methods would then be used to generate sensitivities. Suitable empirical methods have already been coded in the IDAS Configuration Analysis Module CAM aero routines. Similar empirical methods are also available in the Digital DATCOM Program, which could be used in place of CAM.

The computational effort could also be reduced by generating a master database - one time only - and determining sensitivities to be used thereafter for all applications. The master database could be generated in well-defined parameter ranges, using wing characteristics typical of various classes of aircraft; i.e., supersonic high performance aircraft (fighters), subsonic high performance aircraft (trainers), subsonic transport in tanker aircraft, etc. This would allow the database to be more accurate for various applications, and allow for the effort to be completed in stages, although at some loss of generality. The loss of generality should not be overstated, however, since stability and control requirements and wave drag considerations limit the usable range of wing parameters in any case. The approach of tailoring design codes for specific applications has been used successfully before, and does enhance the accuracy of the solutions.

PERFORMANCE SENSITIVITIES

The performance calculations in PSM can be divided into two major categories: mission performance and point performance. Mission performance includes calculating the range/radius that a vehicle is capable of achieving when flying a design mission, or sizing a vehicle (increasing/decreasing physical size and weight) so that the new sized parametric vehicle will achieve a design mission (which includes a specified range/radius). Mission performance evaluates the aircraft performance over a range of mach number/altitude/weight combinations. Point performance on the other hand evaluates the aircraft at a specific mach number/altitude/weight combination. Point performance includes specific excess power (P_S), maximum sustained and instantaneous load factor (N_Z), as well as take-off and landing distance calculations.

Table 3 - I. AERODYNAMIC SENSITIVITIES REQUIRED FOR WING SIZING

	C_{L0}	C_{LA}	C_{LLIMIT}	αC_{LLIMIT}	C_{DSF}	K	C_{LK}	C_{DK}	M_{DD}	C_{DWAVE}	X_{cp}	Y_{cp}
AR	2	1	1	1	2	1	2	2	1	1	1	1
Λ	2	1	1	1	2	1	2	2	1	1	1	1
λ	2	1	1	1	2	1	2	2	1	1	1	1
t/c	0	0	1	1	1	1	2	2	1	1	2	0
Twist	1	0	1	1	0	1	1	1	2	2	0	1
Camber	1	0	1	1	0	1	1	1	2	2	0	1

Legend: 1 = Primary Variable
2 = Secondary Variable
0 = Small impact

In order to calculate mission and point performance of a vehicle, PSM needs to know the aerodynamic coefficients, take-off gross weight and fuel weight, and propulsion thrust and fuel flow. The primary geometric information needed is reference wing area (S), a representative length, α limits and miscellaneous other items. Decomposition theory indicates that the most appropriate sensitivity derivatives would be those relating performance figures of merit (take-off gross weight, range, P_S, N_Z, take-off distance, landing distance) to the aerodynamic coefficients, weight increments, thrust and fuel flow. Then using the chain rule, these performance variables could be related to the design variables. For example:

$$\frac{\partial RANGE}{\partial \psi} = \frac{\partial RANGE}{\partial TOGW} \cdot \frac{\partial TOGW}{\partial WW} \cdot \frac{\partial WW}{\partial \psi} \cdot \Delta\psi + \ldots.$$

or

$$\frac{\partial TOGW}{\partial \psi} = \frac{\partial TOGW}{\partial C_{D0}} \cdot \frac{\partial CD0}{\partial \psi} \cdot \Delta\psi + \ldots + \frac{\partial TOGW}{\partial RANGE} \cdot \Delta RANGE$$

Notice that $\partial WW/\partial\psi$ and $\partial C_{DPARASITE}/\partial\psi$ were calculated previously by Mass Properties and Flight Sciences.

This approach has the advantage of allowing the design variables to be changed without affecting the performance sensitivity expression. Unfortunately there is a very serious problem with this approach when applied to mission performance sensitivities (and hence optimization). As stated previously mission performance measures the vehicle performance over a range of mach numbers. Unfortunately the aerodynamic sensitivity derivatives ($\partial C_{D0}/\partial\psi$, etc.) are valid for only one mach number. In addition, while PSM currently has the capability of generating mission sensitivities (e.g. $\partial TOGW/\partial C_{D0}$) to some of the drag components (C_{D0}, C_{DL}, C_{DWAVE}), PSM assumes that the sensitivity is uniform across all mach numbers. It is well known that this assumption would not be valid for a change in C_{DWAVE} due to a change in Λ. This results in the classic optimization problem: optimizing the vehicle for one mach number will likely degrade its performance at another mach number. It has been proposed that some form of mach number weighting be applied to mission performance optimization. This weighting introduces an undesirable artificiality into the mission performance sensitivities. Also the weighting must be reassessed and changed each time the mission is changed. Finally, mach number weighting still requires the aerodynamic derivatives to be calculated at several mach numbers.

A hybrid approach is being proposed for this implementation of PSM. For this approach sensitivity derivatives will be calculated externally and applied to the aerodynamic and weight calculations in PSM. PSM will use these sensitivities to generate performance (mission and point) data for each parametric vehicle and store results in the current PSM summary matrix. Performance sensitivities will not be generated. However, they could be generated from the summary matrix if further evaluation deems this to be necessary.

OPTIMIZATION

Stated mathematically, the optimization problem to be solved is:

Minimize take-off gross weight $F(\psi)$ for a fixed "design" mission

Subject to:

CONSTRAINTS:	Alternate mission radius $\geq$ required radius	$G1(\psi)$
	Take-off distance $\leq$ required distance	$G2(\psi)$
	Landing distance $\leq$ required distance	$G3(\psi)$
	Max sustained load factor $\geq$ required load factor	$G4(\psi)$
	Specific excess power (SEP) $\geq$ required SEP	$G5(\psi)$
	SEP at specified load factor ≥ 0	$G6(\psi)$

DESIGN VARIABLE LIMITS:

$$\psi_i^l < \psi_i < \psi_i^u \qquad i = 1, 8$$

where: ψ = design variables (S, AR, Λ, λ, t/c, twist, camber, thrust).

This optimization minimizes the take-off gross weight for a vehicle flying a fixed "design" mission. Within PSM it is also possible to maximize the range, radius or time of a specified mission leg (or group of legs) for a fixed take-off gross weight vehicle.

PSM is currently interfaced to the CONMIN optimization program. This section will discuss how that interface works. Also discussed are several alternative approaches to performing optimization of wing design parameters and possible ways to present the results to the user. Finally, an optimization approach will be recommended.

In the current interface of PSM with CONMIN, control of PSM execution is transferred to CONMIN. The user must specify the objective parameter, the constraint parameters and their required values, the design parameters (up to 20) and their upper and lower bounds. The user selects the parameters by identifying appropriate PSM data locations using the PSM editor. CONMIN executes PSM to size the baseline vehicle to the specified mission. CONMIN then chooses new values for the design variables using its own internal logic and re-executes PSM to size the new parametric vehicle to the specified mission. This process continues until CONMIN arrives at an optimum solution. The current implementation does not allow input of externally derived gradients (although CONMIN does have the capability of using them if they were input). As far as output is concerned, all of the normal PSM mission printout is retained when executing under CONMIN control and the parametric vehicles generated by PSM are each added to the PSM summary matrix. Also, any of the normal CONMIN print modes can be specified when setting up an optimization using the PSM editor. This approach to interfacing PSM with CONMIN has the advantage of requiring less PSM executions than would be required to fully populate the design space. In terms of time savings, this can be a very significant advantage as each PSM sizing can take 10 minutes to several hours to execute, depending on the complexity of the mission and the computer. The disadvantage is that each time the optimization is changed (objectives, constraints) all the PSM executions must be repeated.

The first alternative approach is to fully populate the design space, save each case in the PSM summary matrix and have CONMIN perform its optimization using data on parametric vehicles contained in the summary matrix. This approach has the disadvantage of requiring many PSM runs to initially populate the design space. However, after this is done the actual optimization would proceed relatively quickly. Also, different optimizations could be done using the same summary matrix. The summary matrix in PSM stores the objective function, each design variable and each constraint in a separate column. To implement this approach, the summary matrix would have to be expanded from its current limit of 18 columns to something like 36 columns, and PSM would have to be extended to handle something like 10 design variables (currently it is limited to 3). In actual practice, the computer time required to fully populate the design space makes this approach unrealistic. For example, when there are 10 design variables and 3 values for each design variable one would have to run 3^{10} or 59,049 cases. It might be possible to incorporate an experimental design technique such as Central Composite Design or Latin Squares into PSM to create a "reduced" summary matrix, but it is still very likely that the number of cases would be execessive. Finally, additional code would have to be added to IDAS to allow CONMIN to read the summary matrix. It is recommended that if this capability is desired, it be added to the Summary Report Module (SRM) of IDAS, rather than PSM. This is because SRM already has curve fitting and cross plotting capabilities to perform three design variable optimization.

The second alternative approach would be to generate global performance sensitivities ($\partial TOGW/\partial\psi$, $\partial RANGE/\partial\psi$, $\partial LANDING\ DISTANCE/\partial\psi$, etc.) directly from the PSM summary matrix. These global sensitivities would then be input into CONMIN as gradients to the objective and constraint functions. The CONMIN optimization could then take place independently, or CONMIN could be interfaced to accept these gradients directly from PSM. Implementing this alternative would require extensive additional code to generate the local and global sensitivities in PSM. The summary matrix would have to be expanded to handle the additional design variables. Another requirement of this approach is the need to run $2\psi+1$ cases (central difference method) or $\psi+1$ cases (one-sided difference method) in order to calculate all the global sensitivities.

The third alternative would be to generate local performance sensitivities (∂TOGW/∂C_D, ∂TOGW/∂WING WEIGHT, ∂RANGE/∂WING WEIGHT, etc.) in PSM, combine these with the aerodynamic and weight sensitivities which are externally generated, setup a series of simultaneously equations and solve for the global sensitivities using matrix techniques. These global sensitivities would be input to CONMIN as in the previous alternative. To implement this alternative would require substantial coding revisions to PSM to calculate the local sensitivities. In addition, difficulties discussed in the previous section with the aerodynamic coefficients also being functions of mach number would result in performance sensitivities to aerodynamic coefficients being of questionable utility. This alternative would require the greatest level of effort to implement and would not provide any advantage over the previous alternatives.

Our experience with using CONMIN for optimization is limited. However, a preliminary evaluation indicates that CONMIN is entirely adequate for the optimization problem identified at the start of this section. CONMIN is also available on the Prime minicomputers, which is the primary type of computer used at Rockwell to execute PSM. An alternative to CONMIN is the Automated Design Synthesis (ADS) optimization program. ADS is much more sophisticated than CONMIN and has many more optimization options. It has the disadvantage of being non-public domain program which is currently only available at Rockwell on the IBM computer. Since the added sophistication of ADS does not appear necessary for the optimization problem at hand, it is recommended that CONMIN continue to be used for the conceptual design optimization problem.

One of the principal draw-backs to numerical optimization is that the user ends up with an "optimum" design, but very little insight into how the design parameters can be traded off, what constraints are driving the solution and if those constraints were relaxed, what the next constraint would be. In the case of three design variable optimization the interactions can be depicted graphically. Figure 3 - 7 shows an example of a three variable optimization depicted graphically (reference 11). This graphical technique is the way IDAS (SRM) currently does a three design variable optimization, and it has the advantage of providing considerable additional information about the nature of the design space. Unfortunately, three design variables are the upper limit for graphical techniques such as this and the optimization being proposed in this report will have eight or more design variables. Neither CONMIN nor ADS have any graphical output capability. We feel that some form of graphical post processing should be added to CONMIN. The proposed approach for graphical post processing uses a "strip-chart" concept where the objective function, all the design variables and all the constraint variables would be plotted versus iteration. This concept is certainly not ideal, but does provide more useful information than just an optimized point design.

A capability already exists within PSM to do sensitivity studies on each of the design variables. To do this, first the optimum design is determined. Next, cases totalling twice the number of design variables ($\pm\Delta\psi$ for each design variable) must be executed and stored in the summary matrix. Finally, the generic plotting capability of PSM can be used to plot the sensitivity of the objective function to each of the design variables. This technique will be used in combination with the concept described above.

In conclusion, the recommended approach to optimization is to retain the current CONMIN interface to PSM. In addition the summary matrix will be expanded to 36 columns maximum, and a graphic post processing capability will be added to PSM (or alternatively SRM) to display CONMIN results.

TASK 2 CONCLUSIONS

Applying the functional decomposition techniques to the conceptual design phase appears to be feasible. While many of the variables needed to perform detailed analysis (e.g. structural design of a wing rib) are not known at conceptual design, it is possible to get around this problem. This can be done by limiting the functional decomposition to a level at which the needed variables are available.

This task selected the major wing design parameters as the starting point for a modified conceptual design process using the functional decomposition approach. The wing has been decomposed to a level appropriate to conceptual design. Mass properties and aerodynamic scaling models, using sensitivity derivatives available at conceptual design, have been developed. A hybrid approach has been chosen for performance analysis/optimization. For this approach the performance analysis computer program will accept sensitivity derivatives from mass properties and aerodynamics. It will calculate a new value for the objective function in response to a variation in one of the design variables. The optimization program will determine the most efficient way to change the design variables in order to arrive at the optimum solution.

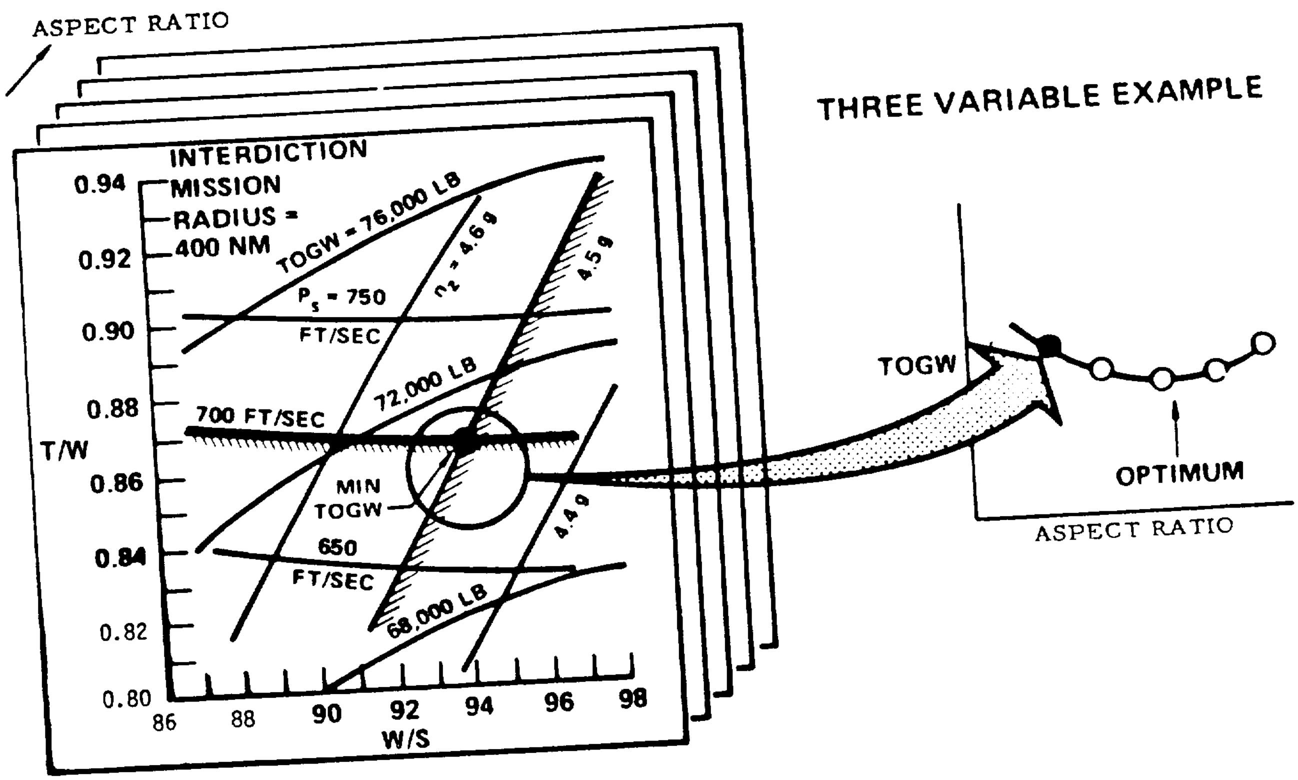

Figure 3 - 7. Three Design Variable Optimization, Graphical Depiction

Section IV

TASK 3 RESULTS

INTRODUCTION

PURPOSE

The purpose of Task 3 was to develop a plan for implementation of the optimization system defined in Task 2 for advanced fighter type aircraft.

SCOPE

This section: 1) defines a test case for the new optimization capability, 2) defines the initial optimization system (summary of Task 2 results), 3) identifies development requirements, 4) presents an overall schedule, and 5) identifies potential future enhancements. A task description, level of effort and calendar time estimates are provided for each of these tasks.

DEFINE TEST CASE

The test case to be used for this project will be the Advanced Technology Multi-role Fighter (ATMF) (reference 12). This was an early Rockwell concept for the Air Force Advanced Tactical Fighter (ATF). It has the advantage of having a complete set of analysis data (aero, weights, propulsion, performance, etc.) and is unclassified. Figure 4-1 shows a three-view drawing of this concept. Table 4-I shows a summary of the ATMF dimensional data. Table 4-II shows a weight statement. Figure 4-2 shows the mission profiles to which this concept was sized. Mission 2 had a radius design constraint of 150 n mi or greater and Mission 3 had an acceleration time design constraint of 45 seconds or less. In addition, there were several point performance design constraints. They were:

1. Landing ground roll $\leq$ 2,000 feet,
2. Max sustained load factor at mach = 1.6 and 50,000 feet altitude $\geq$ 4g,
3. Specific excess power (Ps) at M = 0.9, alt = 30,000 feet, 1g $\geq$ 400 fps,
4. Ps at M = 1.6, alt = 30,000 feet, 1g $\geq$ 950 fps,
5. Ps at M = 1.6, alt = 30,000 feet, 5g $\geq$ 450 fps,
6. Ps at M = 1.8, alt = 50,000 feet, 1g $\geq$ 400 fps,

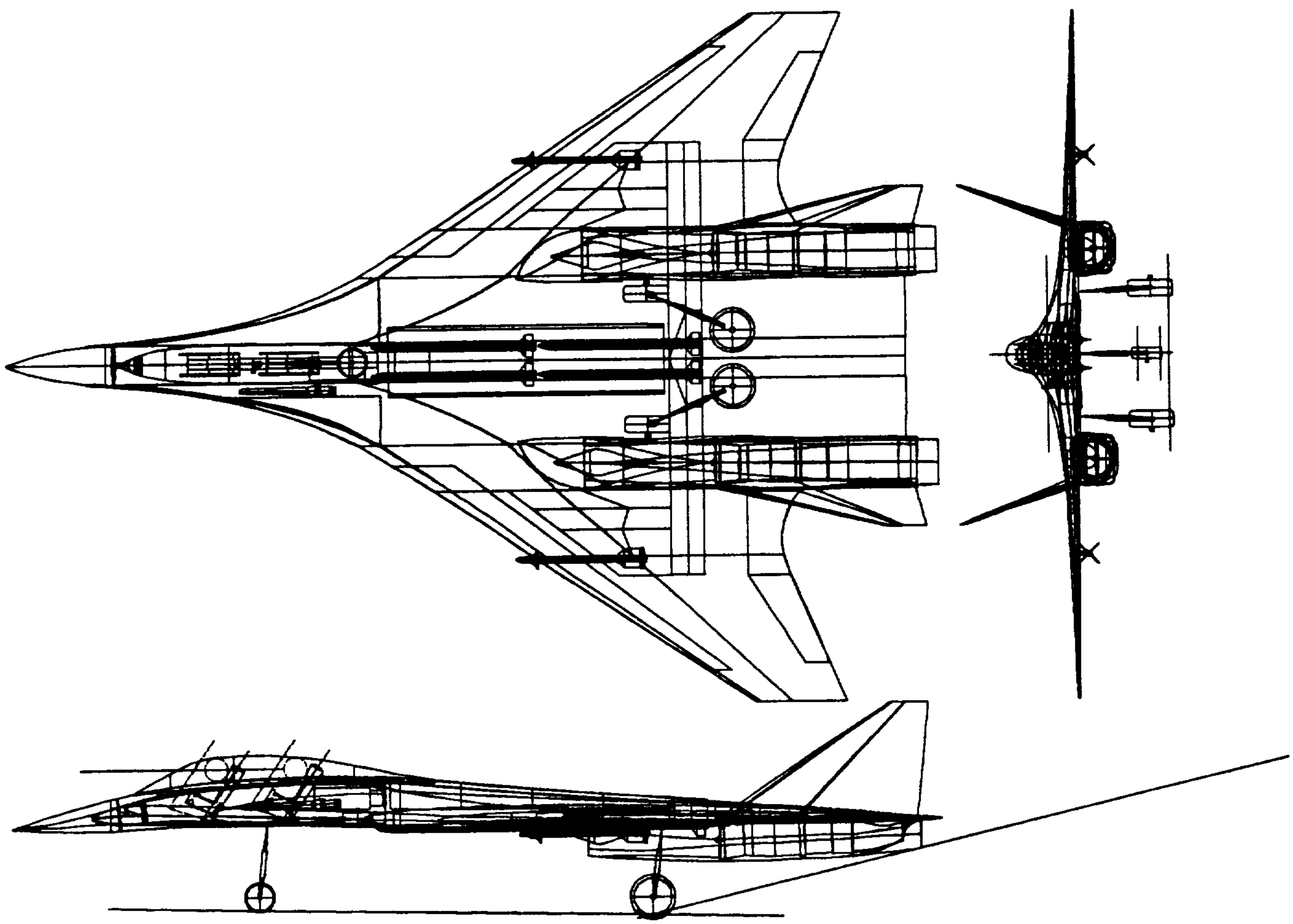

Figure 4 - 1. ATMF Three - View

Table 4 - I(a). ATMF DIMENSIONAL DATA

PROJECT: IDAS TEST CASE NUMBER TWO
BASELINE: ADVANCED TACTICAL MULTIROLE FIGHTER, D783-21
TASK: INPUT DATA PREPARATION AND CHECKOUT
CASE: GEOMETRY, WEIGHT, AND AERODYNAMIC ANALYSIS

DIMENSIONAL DATA

- WING LOADING - (LBS/SQ FT)	46.58
AREA - MAX (SQ FT)	10.89
FINENESS RATIO - OVERALL	18.51

	BODY	CANOPY	NACELLE I	NACELLE II
- LENGTH (FT)	64.92	21.68	21.56	0.00
- DEPTH (FT)	3.57	2.59	3.61	0.00
- WIDTH (FT)	1.55	1.46	3.61	0.00
- WETTED AREA (SQ FT)	312.75	86.90	140.62	0.00
- AREA MAX (SQ FT)	9.84	1.91	10.21	0.00
- FINENESS RATIO	18.74	13.92	5.98	0.00
- NUMBER			2.00	0.00
- TOTAL BODY WETTED AREA	585.51			
- VOLUME-PRESSURIZED	80.66			
- -REQ'D(CU FT)	958.31			
- -AVAIL(CU FT)	246.34			

	WING	HORIZONTAL I	VERTICAL I	HORIZONTAL II	VERTICAL II
- AREA-BASIC (SQ FT)	1011.94	0.00	78.67	0.00	0.00
- AREA-GROSS (SQ FT)	1394.39				
- AREA EXPOSED (SQ FT)	1176.55	0.00	157.34	0.00	0.00
- AREA WETTED (SQ FT)	2292.09	0.00	313.70	0.00	0.00
- AREA CAMBERED (SQ FT)	1344.71				
- AREA SURFACE CONTROL	199.89	0.00	0.00	0.00	0.00
- ASPECT RATIO	2.52	0.00	1.83	0.00	0.00
- TAPER RATIO	0.20	0.00	0.19	0.00	0.00
- UNIT WEIGHT (PSF)	8.66	0.00	3.00	0.00	0.00
- SPAN (FT)	58.48	0.00	8.98	0.00	0.00
- M.A.C. (FT)	35.16	0.00	10.12	0.00	0.00
- M.A.C. BUTT LINE (FT)	8.26	0.00	3.47	0.00	0.00
- M.A.C. T/C	0.049	0.000	0.047	0.000	0.000
- SWEEP AT LE-DEG	56.01	0.00	54.97	0.00	0.00
- SWEEP AT 25P CHORD-DEG	50.58	0.00	47.56	0.00	0.00
- SWEEP AT 50P CHORD-DEG	43.53	0.00	37.26	0.00	0.00
- SWEEP AT TE-DEG	22.63	0.00	5.44	0.00	0.00
- ROOT CHORD (FT)	33.50	0.00	14.74	0.00	0.00
- ROOT CHORD (FT) GROSS	63.01				
- ROOT THICKNESS (FT)	1.55	0.00	0.76	0.00	0.00
- ROOT T/C	0.055	0.000	0.052	0.000	0.000
- ROOT T/C GROSS	0.055				
- SOB CHORD (FT)	58.82	0.00		0.00	
- SOB THICKNESS (FT)	3.22	0.00		0.00	
- SOB BUTT LINE (FT)	1.44	0.00		0.00	
- BREAK BUTT LINE (FT)	12.62				
- BREAK CHORD (FT)	20.85				
- BREAK THICKNESS (FT)	0.79				
- BREAK T/C	0.039				
- TIP CHORD (FT)	6.59	0.00	2.78	0.00	0.00
- TIP THICKNESS	0.11	0.00	0.01	0.00	0.00
- TIP T/C	0.017	0.000	0.003	0.000	0.000
- VOLUME (CU FT)	147.35	0.00	12.62	0.00	0.00
- TWIST (DEG)	0.76				
- TYPE CAMBER	2				
- LE DEVICE SPAN RATIO	0.75				
- LE RADIUS (0/0 CHORD)	3.0000				
LOW WING AIRPLANE					
- LE APEX LOCATION (FT)	16.43	0.00	50.72	0.00	0.00
- TOTAL AIRPLANE WETTED AREA	3302.97				
- FUEL -GALLONS	1867.69				

Table 4 - I(b). ATMF DIMENSIONAL DATA

PROJECT: IDAS TEST CASE NUMBER TWO
BASELINE: ADVANCED TACTICAL MULTIROLE FIGHTER, D783-21
TASK: INPUT DATA PREPARATION AND CHECKOUT
CASE: GEOMETRY, WEIGHT, AND AERODYNAMIC ANALYSIS

DIMENSIONAL DATA (CONTD)

LOADS DATA

GUST LOAD INCREMENT	3.37220
FLIGHT LOAD FACTOR -ULT	11.00
GUST LOAD FACTOR -ULT	6.56
VERT TAIL LOAD -LIMIT GUST	15523.
VERT TAIL LOAD -LIMIT MANEUVER	17174.
VERT TAIL LOAD -ULTIMATE	25760.
HORZ TAIL LOAD -LIMIT MANEUVER	-108891.
HORZ TAIL LOAD -LIMIT GUST	0.
HORZ TAIL LOAD -ULTIMATE	0.
HORZ TAIL ASSYM MOMENT	0.

TYPE 1 ENGINE DATA

THRUST-TO-WEIGHT RATIO	0.90
ENGINE SCALE FACTOR	1.10
TOTAL INSTALLED THRUST (SLS)	23000.00
ENGINE DIAMETER (FT)	2.95
ENGINE LENGTH (FT)	8.54
ENGINE WEIGHT (LB)	3200.00
NUMBER OF ENGINES	2.00
INLET LENGTH (FT)	9.47
INLET CAPTURE AREA (SQ FT)	5.15

REFERENCE ENGINE DATA

INSTALLED THRUST (SLS)	19557.82
ENGINE DIAMETER (FT)	2.72
ENGINE LENGTH (FT)	8.09
ENGINE WEIGHT (LB)	2721.09
ENGINE INLET LENGTH (FT)	9.47
ENGINE INLET CAPTURE AREA (SQ FT)	4.38

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Table 4 - I(c). ATMF FUSELAGE REQUIRED VOLUME

PROJECT: IDAS TEST CASE NUMBER TWO
BASELINE: ADVANCED TACTICAL MULTIROLE FIGHTER, D703-21
TASK: INPUT DATA PREPARATION AND CHECKOUT
CASE: GEOMETRY, WEIGHT, AND AERODYNAMIC ANALYSIS

FUSELAGE REQUIRED VOLUME

WING CARRYTHRU STRUCTURE		0.00 CU FT
RADOME		16.51
FUSELAGE STRUCTURE		65.83
FORWARD SECTION	19.68	
MID SECTION	29.10	
AFT SECTION	17.05	
LANDING GEAR		104.56
MAIN WELLS	86.57	
NOSE WELLS	17.90	
ARRESTING GEAR	0.00	
COCKPIT		80.66
MAIN	80.66	
AFT	0.00	
PAYLOAD		321.01
INTERNAL	0.00	
SEMI-SUBMERGED	321.01	
TANGENTIAL	0.00	
SYSTEMS AND EQUIPMENT		207.36
APU	7.18	
AVIONICS	111.31	
ELECTRICAL	3.26	
ECS	30.42	
GUN	25.00	
AMMUNITION	30.19	
CONTROL AND DISTRIBUTION		41.96
AVIONICS	1.11	
ENGINE	0.64	
HYDRAULICS	6.34	
FLIGHT CONTROLS	11.39	
FUEL PLUMBING	16.52	
ARMAMENT WIRING	2.82	
ECS	3.15	
MISCELLANEOUS		0.00
UNUSABLE		14.16
FUEL SYSTEM		106.26
FUEL	106.09	
TANKS	0.16	
TOTAL VOLUME REQUIRED		955.31

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Table 4 - II. RESULTS OF WEIGHT ANALYSIS - SHIFTED WING

PROJECT: IDAS TEST CASE NUMBER TWO
BASELINE: ADVANCED TACTICAL MULTIROLE FIGHTER, D703-21
TASK: INPUT DATA PREPARATION AND CHECKOUT
CASE: GEOMETRY, WEIGHT, AND AERODYNAMIC ANALYSIS

SHIFTED WING	GROUP WEIGHT STATEMENT			GROUP C.G. STATEMENT STATION LOCATION	
				WING	BODY
STRUCTURE			16377.0 LB.		
WING		8760.5			
VERTICAL TAIL		472.0		847.376	
BODY		2935.9		1047.184	
LANDING GEAR		1522.3			379.630
MAIN GEAR	1260.5				310.351
NOSE GEAR	261.9				260.458
ENGINE SECTION		2077.3			597.086
INSTALLATION	83.2				
NACELLE	1994.1			1046.831	
AIR INDUCTION		688.9		1005.660	
PROPULSION			8384.7	736.462	
ENGINES		6400.0			
SUBSYSTEMS		1278.6		1046.032	
REMOTE GEAR BOX	186.2				
THRUST REVERSERS	962.2				
LUBRICATION	43.5			1184.757	
STARTING	86.7			1046.831	
CONTROLS		35.1		1046.831	
FUEL SYSTEM		671.0			227.491
SYSTEMS AND EQUIPMENT			6540.5		696.150
FLIGHT CONTROLS		991.2			
AUXILIARY POWER UNIT		229.7			618.560
INSTRUMENTS AND NAV		225.2			273.034
HYDRAULICS		348.1			243.541
ELECTRICAL		660.1			491.240
AVIONICS		2386.4			537.784
ARMAMENT		425.2			177.547
FURNISHINGS		354.3			285.052
AIR COND / ANTI-ICE		706.1			210.048
AUXILIARY GEAR		14.3			317.673
			----------		736.462
WEIGHT EMPTY			31302.2		
BASIC OPERATING ITEMS		1003.9			743.778
CREW	430.0				
OXYGEN	26.0				210.048
TRAPPED FUEL	182.0				210.048
OIL	90.9				696.157
GUN	275.0			1046.031	
					271.585

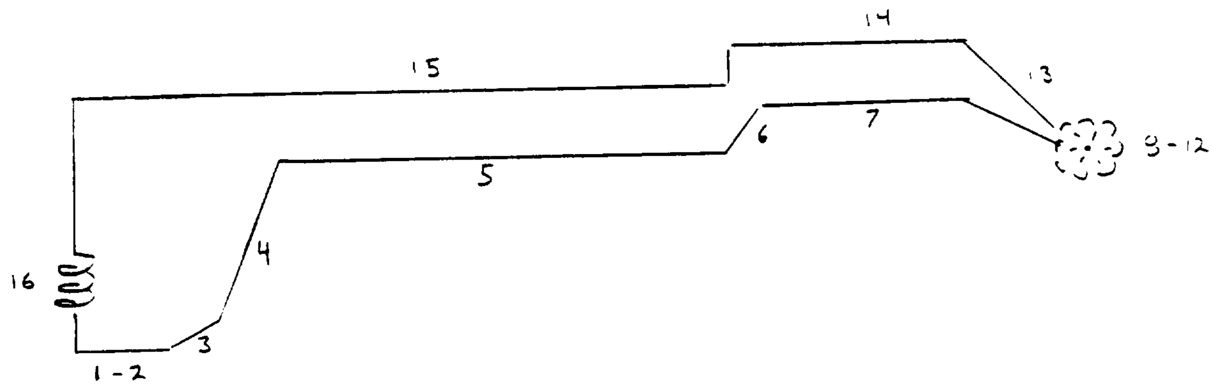
BASIC OPERATING WEIGHT			32306.1		
PAYLOAD		2690.0			736.462
EXTERNAL	2160.0				615.902
AMMUNITION	530.0				708.387
					271.585

ZERO FUEL WEIGHT			34996.1		
FUEL		12135.0			726.749
WING	7055.0				696.157
FUSELAGE	5080.0			893.672	
					421.855

TAKEOFF GROSS WEIGHT			47131.3		718.594
DESIGN WEIGHT			44704.3		
MAXIMUM WEIGHT			47131.3		
LANDING WEIGHT			36113.3		
AMPR WEIGHT			21487.4		

GROUP C.G. STATEMENT SUMMARY

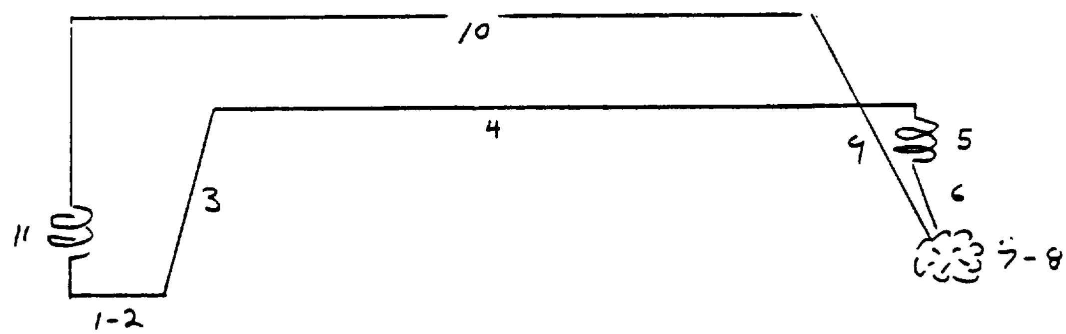
WEIGHT EMPTY	
STA LOC AIRPLANE C.G.	743.778
C.G. IN PCT WING MAC	26.734
BASIC OPERATING WT	
STA LOC AIRPLANE C.G.	736.462
C.G. IN PCT WING MAC	25.088
ZERO FUEL WEIGHT	
STA LOC AIRPLANE C.G.	726.749
C.G. IN PCT WING MAC	22.698
TAKEOFF WEIGHT	
STA LOC AIRPLANE C.G.	718.594
C.G. IN PCT WING MAC	20.764



Design Radius: 250 n.mi.

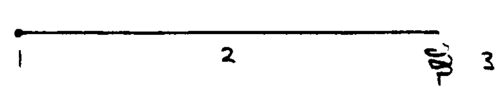
1.	Warmup		15 min idle power @ S.L.	
2.	Takeoff		fuel & time as required	
3.	Accel	to	climb condition	
4.	Climb	to	BCA/BCM	200 n.mi.
5.	Cruise		BCA/BCM	
6.	Accel	to	1.6M @ 50,000 feet	
7.	Dash		1.6M @ 50,000 feet	
8.	Combat		720° turn, .8M @ 30,000 feet	
9.	Drop		2xAMRAAM	
10.	Accel	to	1.6M @ 30,000 feet	
11.	Combat		720° turn, 1.6M @ 30,000 feet	
12.	Drop		2xAMRAAM	
13.	Climb	to	1.6M @ 50,000 feet	
14.	Dash		1.6M @ 50,000 feet	
15.	Cruise		BCA/BCM, 200 n.mi.	
16.	Loiter		20 min @ S.L.	

Figure 4 - 2(a). Mission 1 - Air Superiority (2 x AIM-9L, 4 x AMRAAM)



1.	Warmup		15 min idle power @ S.L.
2.	Takeoff		fuel & time as required
3.	Climb	to	.75M @ 5,000 feet
4.	Cruise		.75M @ 5,000 feet
5.	Loiter		15 min BLM @ 5,000 feet
6.	Accel	to	.8M @ S.L.
7.	Combat		720° turn, .8M @ S.L.
8.	Drop		12xMK82
9.	Climb	to	.75M @ 5,000 feet
10.	Cruise		.75M @ 5,000 feet
11.	Loiter		20 min @ S.L.

Figure 4 - 2(b). Mission 2 - Attack and Combat Air Patrol (12 x MK82 LDGP)



1.	Setup	remove 50% fuel load
2.	Accel	.8M to 1.6M @ 30,000 feet
3.	Dummy	cruise 1.6M @ 30,000 feet

Figure 4 - 2(c). Mission 3 - Acceleration (2 x AIM-9L, 4 x AMRAAM)

A thrust to weight (T/W) and wing loading (W/S) optimization was performed to minimize the take-off-gross weight, subject to the above design constraints. Table 4-III shows the PSM summary matrix for the T/W and W/S optimization. The design space consisted of three values each for T/W and W/S resulting in a matrix of 9 parametric designs, each of which were sized to the design missions. The optimization results are graphically depicted in Figure 4-3. They indicate that the optimum W/S = 63 psf and T/W = 1.015. This results in a take-off gross weight of approximately 43,500 pounds. Since PSM does not currently have the capability to perform an automated optimization of wing design parameters, this was not done. However, it will still be possible to compare the results of this optimization to those of the proposed enhanced PSM. It should be noted that considerable time was spent by both Flight Sciences and Mass Properties to identify the best wing for the design requirements. Therefore, It is anticipated that large changes in the wing design variables should not occur when the wing design variable optimization is done.

DEFINE AN INITIAL OPTIMIZATION SYSTEM

Task 2 defined an initial optimization system to be incorporated into the Rockwell Integrated Design and Analysis System. The detailed description appears in the previous section, and will be summarized here. The capability to be added to IDAS is to optimize the wing design variables: aspect ratio (AR), sweep (Λ), taper ratio (λ), thickness ratio (t/c), twist and camber (note: IDAS already optimizes wing area (S)) for specified mission, maneuverability, and takeoff/landing requirements. The approach will be to :

1. Modify the IDAS Parametric Synthesis Module (PSM) scaling models to use aerodynamic and structural weight sensitivities,

2. Generate the weight sensitivity derivatives externally to PSM using the Structural Weight Estimation Program (SWEEP),

3. Generate the aerodynamic sensitivities externally to PSM either a modified IDAS Configuration Analysis Module (CAM) or another appropriate technique (e.g. APAS),

4. Use the existing PSM mission performance model,

5. Expand the PSM summary matrix to store more design variables,

6. Use the existing PSM interface to CONMIN for optimization,

7. Add a new capability to graphically depict the optimization results.

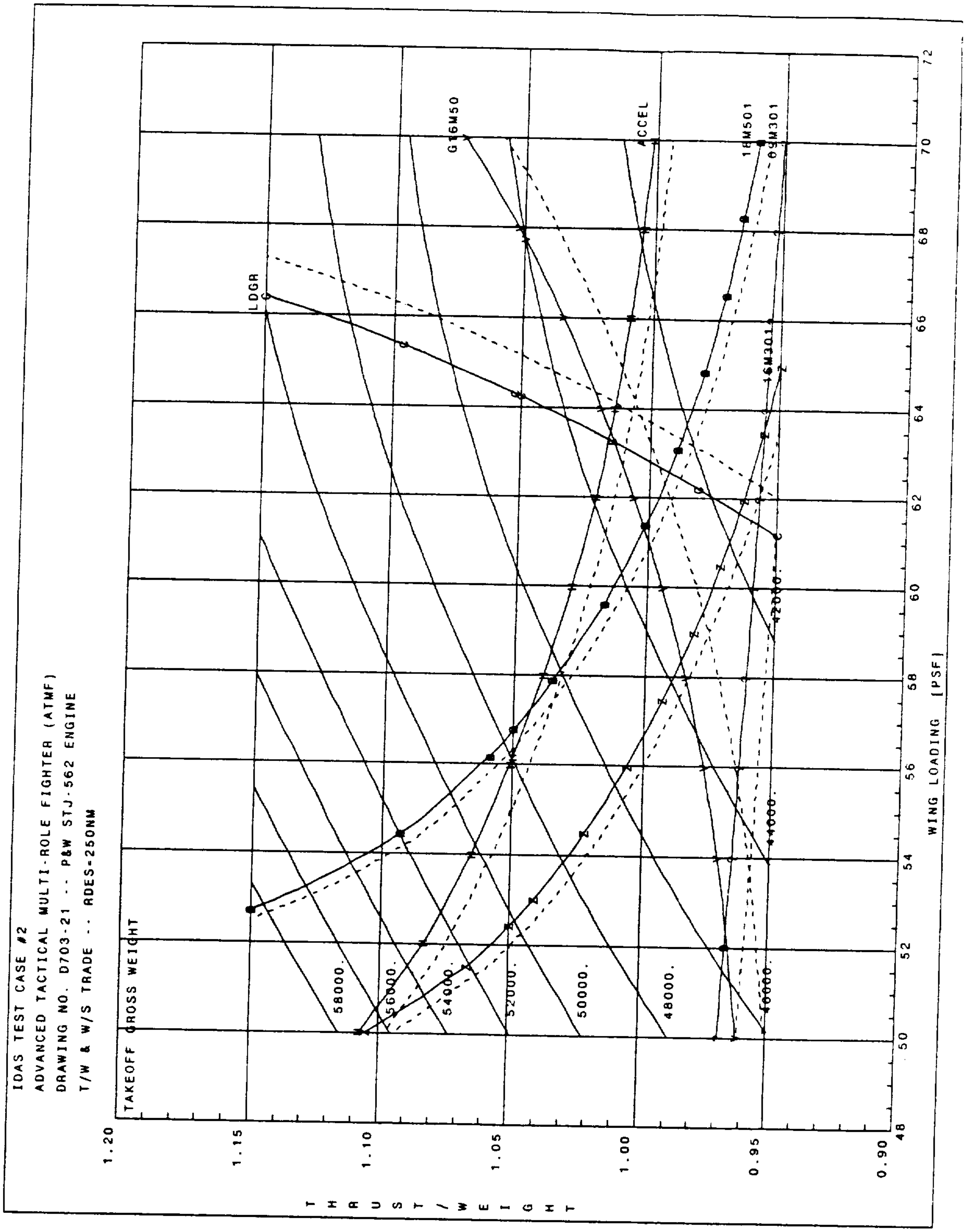


Figure 4 - 3. Design Trade Cross Plot

Table 4 - III. TRADE SUMMARY MATRIX

IDAS TEST CASE #2
ADVANCED TACTICAL MULTI-ROLE FIGHTER (ATMF)
DRAWING NO. D703-21 -- P&W STJ-562 ENGINE
T/W & W/S TRADE -- RDES=250NM

SUMMARY PRINT

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
N	TOGW	FUEL	SREF	ENGS	R1	WO/S	T/WO	R2	ACCEL	LDGR	G16M50	09M301	16M301	16M305	18M501
1	46095.	12243.	921.91	1.1195	250.27	50.000	0.95000	174.07	0.92281	1717.8	3.9560	389.57	763.15	582.46	260.79
2	41613.	10885.	693.55	1.0106	249.85	60.000	0.95000	175.52	0.84307	1972.6	3.8650	395.16	908.14	691.23	350.82
3	39609.	10423.	565.84	0.96199	250.36	70.000	0.95000	181.22	0.80987	2188.6	3.6605	399.53	976.76	724.16	391.14
4	43649.	11610.	623.56	1.1717	249.60	70.000	1.0500	177.84	0.69996	2126.3	3.9366	455.04	1155.1	903.51	475.88
5	46137.	12187.	768.95	1.2385	250.08	60.000	1.0500	172.68	0.72811	1904.9	4.1617	450.42	1075.5	859.39	428.57
6	52082.	13981.	1041.6	1.3981	250.03	50.000	1.0500	171.71	0.80107	1674.8	4.2697	444.77	894.36	714.31	315.09
7	61884.	16981.	1237.7	1.8194	250.26	50.000	1.1500	173.14	0.72010	1620.0	4.4742	501.34	987.40	808.32	343.74
8	52614.	14120.	876.90	1.5469	250.14	60.000	1.1500	171.39	0.64167	1854.8	4.4280	506.69	1237.4	1022.3	502.16
9	49166.	13275.	702.37	1.4455	250.18	70.000	1.1500	175.49	0.61605	2079.8	4.1911	511.39	1333.0	1082.7	559.86

The proposed CONMIN graphical output capability is to use a strip chart approach where the objective function, constraints and design variables are graphed versus iteration number.

IDENTIFY DEVELOPMENT REQUIREMENTS

The development requirements are broken down into the following tasks:

Task A - Background Research/Collect Data,

Task B - Refine Mathematical Models and Resolve Technical Problems,

Task C - Develop/Modify Computer Programs to Generate Aerodynamics and Mass Properties Sensitivities,

Task D - Modify the Integrated Design and Analysis System (IDAS),

Task E - Generate Sensitivity Derivatives,

Task F - Optimize Test Case Wing Design Variables,

Task G - Optimization - Second Iteration,

Task H- Documentation.

In addition, computer resources required for all of the tasks are identified.

TASK A - BACKGROUND RESEARCH/COLLECT DATA

As there are several uncertainties related to the feasibility and best approach to using sensitivity derivatives in IDAS, this task is to do the research and collect the data needed to allow the uncertainties to be intelligently resolved. This section discusses scope of research needed for weight sensitivity derivatives and aerodynamic sensitivity derivatives.

Aerodynamics

As indicated in the Task 2 discussion, the aerodynamic methods used in preliminary design generally preclude the explicit formulation of sensitivities, and using these methods to formulate sensitivities numerically requires prohibitive amounts of calculations. Therefore, the recommended approach is to use simple "handbook" methods to calculate the required sensitivities. Consistent with this approach, each term in Table 3 - I will be examined, and those which are determined to be negligible will be omitted.

Estimation methodology for all of the required aerodynamic data will be collected. It is anticipated that this collection will include simple exact methods, approximate methods, and empirical data. Sources to be searched will include the existing IDAS CAM, the USAF Stability and Control DATCOM, and other handbooks and source material in use by the technical staff. The ATMF test case will be analyzed using current aerodynamic preliminary design methods. This will establish a repeatable aerodynamic baseline to which the results of Tasks F and G can be compared.

Calendar time
6 weeks

Level of effort
400 hours

Mass Properties

Develop a data base of actual and in-house study aircraft wings. The data base will include actual and/or estimated weights, geometry and design information. As a minimum the parameter list will include the seven parameters (excluding twist and camber) selected for incorporation in the preferred simplified wing weight expression.

Explore the possibility of obtaining historical twist and camber data.

Calendar time
8 weeks

Level of effort
300 hours

TASK B - REFINE MATHEMATICAL MODELS AND RESOLVE TECHNICAL UNCERTAINTIES

This task uses the results of Task A to resolve those uncertainties identified in the previous section, and any other ones that have surfaced along the way. These uncertainties include (but are not limited to):

1. Automating sensitivity generation to the point that it will become routine with disciplines,

2. independence of sensitivities within a given discipline,

3. non-linearity of sensitivities and valid limits for linear derivatives

4. whether higher order derivatives need to be included,

5. mach number sensitivity of aero derivatives.

Aerodynamics

Algorithms will be formulated for each required aerodynamic parameter. It is anticipated that multiple candidates will be available in most cases, possibly yielding conflicting results. Further study of the original source material may be required to select the most appropriate method in each case. Where differing approaches give better agreement for different cases, multiple algorithms may be required and selected within the code based on each individual set of input parameters. For example, different techniques exist for estimating the maximum lift coefficient of low aspect ratio wing and high aspect ratio wings. Each works well within its own limits, but neither is accurate for all wings. Wherever this approach is selected, a smooth crossover must be assured in order to avoid discontinuities for those cases where input geometry may overlap two methods.

The methodology will be specific to wings only (no fuselage or empennage effects will be included). The wings will also be assumed to have fixed geometry - both camber and sweep.

Vehicle trim strategies can involve much more than the wing design; i.e., desired static margin, placement of wing, use of canard or aft tail, etc. Therefore, the methodology generated will be for the untrimmed condition. Since the data for the initial or starting configuration will normally be trimmed, the lack of trim effects in the sensitivities should not in general cause misleading results.

During this task, a scheme for treating sensitivities with two independent variables (mach number and lift coefficient) will also be developed. At the end of this task, the selected methodology and approach will be documented in an internal letter.

Calendar time
4 weeks

Level of effort
200 hours

Mass Properties

The Structural Weight Estimating Program (SWEEP), the SWEEP/ Aerodynamics Interface Program (SWAIP), and the Flexible Unified Distributed Panel program (FUDP) provide the necessary capability of evaluating loadings and loads effects for moderate and high aspect ratio wing planforms. However, should similar data br required for low aspect ratio (delta) wings, development effort will be required to modify SWEEP. This task will consist of exploring the problems associated with the modification. Specifications will be drafted for a computer program system which will integrate the analysis methods. The results of this task will be documented in the internal letter.

Calendar time
4 weeks

Level of effort
100 hours

Performance

This task will include monitoring the results of the aerodynamic and mass properties efforts under this task and modifying the performance/synthesis algorithms in IDAS to properly handle the aerodynamic and weight sensitivities.

Calendar time
4 weeks

Level of effort
80 hours

TASK C - DEVELOP/MODIFY COMPUTER PROGRAMS TO GENERATE SENSITIVITIES

Aerodynamics

A new code will be developed to generate the aerodynamic sensitivities using the methods and approach described above. This will be a stand-alone Fortran code compatible with the computer hardware available for advanced design use. Input will include the basepoint wing configuration (aspect ratio, taper ratio, sweep, thickness, twist, and camber) plus the mach numbers of interest. Output will consist of a matrix of sensitivities as shown in Table 3 - I, which will be repeated at each desired mach number. The output format will be compatible with the input format of the synthesis code, and will be made available in tabulated form and in a computer file.

Check cases will be compared with hand estimates to verify the code. The code will be documented in a TFD which will include a flow chart, input and output formats and guides, and any other necessary operating instructions.

Calendar time
10 Weeks

Level of effort
600 hours

Mass Properties

Develop a computer program system which will extract historical and analytically derived data as subsets of the data base and calculate the required sensitivity parameters. One operating mode would consist of calculating the required sensitivities by finite differences. An alternate mode would establish sensitivities based on regression analysis of the appropriate subsets of the data base. The code will be documented in a TFD which will include a flow chart, input and output formats and guides, and any other necessary operating instructions.

Calendar time
16 weeks

Level of effort
240 hours

TASK D - MODIFY IDAS

This task consists of the following subtasks:

1. modify the PSM geometry scaling model and add additional parameters to the PSM editor,
2. modify the PSM weight scaling model,
3. modify the PSM aerodynamic scaling model to accept sensitivity derivatives as a function of mach number,
4. Expand the size of the PSM summary matrix, and
5. add a graphic capability for CONMIN output.

Each of these subtasks has been described in the Task 2 results section. The uncertainty in this task is with the level of effort required for the additional graphical capability. The code changes will be documented by updating existing PSM users (NA-82-467 Vol IV) and maintenance (NA-82-468 Vol IV) manuals.

Calendar time
26 weeks

Level of effort
520 hours

TASK E - GENERATE SENSITIVITIES FOR TEST CASE

Aerodynamics

Using the code generated in Task C, a full set of aerodynamic sensitivities will be generated for the test case. These data will be transmitted to the performance group in the agreed upon format, and will be documented in an internal letter.

Calendar time
4 weeks

Level of effort
280 hours

Mass Properties

Develop baseline wing weight and sufficient data to generate the nine required sensitivity parameters. Table 4-IV, below, is a matrix of point solutions that will be calculated, the associated programs that will be executed and estimated hours to perform the task. Should the baseline combination of aspect ratio, sweep, taper ratio, and thickness ratio result in a severe flutter problem, the flutter optimization program (BEFO) will be used to address the flutter sensitivities. These data will be transmitted to the performance group in the agreed upon format, and will be documented in an internal letter.

Table 4 - IV. MASS PROPERTY EVALUATIONS

PARAMETERS	MAN HOURS	PROGRAM SOLUTIONS			
		SWEEP	SWAIP	FUDP	BEFO
Baseline	120	1	1	1	1
S	30	2	2	2	
AR	30	2	2	2	
λ	30	2	2	2	
Λ	30	2	2	2	
t/c	12	2	-	-	
Load	18	2	2	-	
Material	12	1	-	-	
Xcp	30	2	2	2	
Ycp	30	2	2	2	
TOTAL	342	18	15	13	1

Calendar time
9 weeks

TASK F - OPTIMIZE TEST CASE WING DESIGN VARIABLES

This task will consist of inputting the aerodynamic and weight sensitivities to PSM and performing a wing design variable optimization of the existing ATMF test case. The optimization will be done in two steps. First will be a fixed take-off gross weight vehicle, with the objective to maximize mission radius. The second step will be to minimize the vehicle take-off gross weight for a fixed mission. The final optimized vehicles for both these steps will be re-evaluated by aerodynamics and mass properties to determine if the improvements in aero and weight predicted by PSM can actually be attained. The level of effort needed on this task will be primarily for debugging and problem solving of the modified PSM, as well as the re-analysis of the two optimized vehicles by Flight Sciences and Mass Properties. It is estimated that this task will take one person two months to debug PSM and perform the optimization. One month will be required to perform the aerodynamic and weight evaluation requiring one person each from Flight Sciences and Mass Properties. The results of this task will be documented in internal letters from Aerodynamics, Mass Properties, and Performance.

Calendar time
12 weeks

Level of effort
640 hours

TASK G - OPTIMIZATION - SECOND ITERATION

One of the fundamental premises of using linear sensitivity derivatives to model phenomena that exhibits both linear and nonlinear behavior is that these derivatives will be valid only over a limited range. With this in mind, one approach to optimization is to define sensitivity derivatives over a limited range from the baseline, conduct the optimization, establish a new baseline, calculate sensitivity derivatives from the new baseline and re-optimize. This process would continue until no further improvements are possible.

Another approach would be to make the range of validity for the sensitivity derivatives unlimited. The optimization would then identify a theoretical optimum mix of design variables using essentially unconstrained sensitivities. The resulting optimum vehicle would be analyzed by Aero and Mass Properties to determine if any of the design variables had been scaled outside a reasonable (but unquantified) range. A new baseline would be established, analyzed and new sensitivity derivatives calculated. The new baseline would then be re-optimized using the new sensitivity derivatives. NASA's experience with this approach on non-linear optimization problems indicates that the solution often departs too far from the optimum solution, and convergence to the optimium solution fails. We do not recommend this approach.

The optimization in Task F will examine results with both constrained and unconstrained sensitivity derivatives, if limits can be quantified. Depending on the results of Task F, one of the two above approaches will be selected for re-optimization. Task G will be composed of four subtasks as follows. The results of each subtask will be documented in an internal letter.

Aerodynamics

Generating sensitivities for the new baseline will require the same time as it did in Task E.

Calendar time
4 weeks

Level of effort
280 hours

Mass Properties

Sensitivities generated for this task will be employing the database which was populated during Tasks A, C, and E. Therefore, substantially less time will be required for this subtask than was required for Task E. This assumes that the sensitivities needed for this task fall within the range of data in the database.

Calendar time
4 weeks

Level of effort
160 hours

Optimization

Much of the debugging required in Task F will not have to be repeated, so it is estimated that half the level of effort expended in Task F will be required here.

Calendar time
4 weeks

Level of effort
160 hours

Reanalysis

This will require the same level of effort as was required for aerodynamic and weight evaluation in Task F. The level of effort is two people (one aerodynamics and one mass properties) for four weeks.

Calendar time
4 weeks

Level of effort
320 hours

TASK H - DOCUMENTATION

This task will consist of preparing a single final report which documents the results of each of the tasks A through G. Updates to users manuals for all the computer programs that are to be modified (e.g., PSM) and documentation of any new computer programs written to generate sensitivity derivatives, will be documented under those tasks. Aerodynamics and Mass Properties will each be responsible for preparing sections of the final report. Performance will be responsible for the optimization sections as well as overall integration of the report.

Calendar time
12 weeks

Level of effort
500 hours

COMPUTER RESOURCES

The above tasks will require the use of a variety of computer programs located on various computers within Rockwell. In addition, new software developed under this effort will require a development computer system and will have a target computer(s) for its operation. The following paragraphs identify computer hardware and software needed by Aerodynamics, Mass properties, and Performance/Optimization, respectively.

Aerodynamics

The basic aerodynamic codes used in conceptual or preliminary design are small to moderate in size, and require relatively low execution time, even on the smaller mainframes. Currently, these codes reside in Rockwell's IBM mainframe at Seal Beach, which is where the codes are usually developed. For unclassified projects, the IBM is normally used to generate data as well. For classified programs, a number of small Prime mainframes are available. In addition, IBM-type personal computers which can execute most of these codes with reasonable efficiency are becoming available for both classified and unclassified use. The current codes and associated hardware appear in Table 4-V(a) below.

Table 4 - V(a). COMPUTER RESOURCE REQUIREMENTS - AERODYNAMICS

Software	Computers
UDP (linear aero)	IBM Mainframe, Primes, IBM P/C
OPT (twist and camber, C_{D_L})	IBM Mainframe, Primes
APAS (linear wave drag)	IBM Mainframe, Primes

The proposed new code for generating linear aerodynamic sensitivities will be developed in an unclassified environment, and the data for the test case will also be unclassified. Therefore, the IBM mainframe will be used to support this effort. In developing the code, however, its subsequent use on the other machines will be a ground rule for the programmer.

Mass Properties

The Mass Properties computer programs and the computers they execute on which will be required for this development effort appear in Table 4-V(b), below.

Table 4 - V(b). COMPUTER RESOURCE REQUIREMENTS - MASS PROPERTIES

Software	Computers
SWEEP	IBM Mainframe at Seal Beach
SWAIP	IBM Mainframe at Seal Beach
FUDP	IBM PC
BEFO	CDC at Seal Beach
New software	IBM PC (preferably) or IBM Mainframe

Performance/Optimization

The Performance/Optimization computer programs and the computers they execute on, which will be required for this development effort appear in Table 4-V(c), below.

Table 4 - V(c). COMPUTER RESOURCE REQUIREMENTS - PERFORMANCE

Software	Computers
PSM (performance)	Prime 850 at El Segundo
CONMIN (optimization)	Prime 850 at El Segundo
New software	Prime 850 at El Segundo

OVERALL SCHEDULE

The overall schedule for this effort appears in Figure 4 - 4. This effort is projected to take 17 months and require 5120 hours.

FUTURE SYSTEM DEVELOPMENT

Future development of the conceptual design optimization capability will be done in four phases. The wing design parameter optimization will be extended first to variable geometry wings (variable sweep, camber and maneuver load control). This optimization will include schedules for the variable geometry design parameters. Next, the optimization will be extended to include trim, control power, and other agility related design parameters. This phase will borrow heavily from the current Rockwell IRAD effort to design an advanced technology wing demonstration (TPA 150). This phase will include extended aerodynamics, mass properties and performance, and, if feasible, will add dynamic performance constraints along the lines of MIL-STD-8785C and incorporate a simplified optimum control algorithm (neglecting high order effects). The next phase will be to add fuselage design variables to PSM and calculate aerodynamic and mass properties sensitivities to these fuselage design variables. The fourth and final phase will be to add propulsion related design variables. The design variables inlet, nozzle and engine size variables, as well as some of the more basic engine cycle design parameters (i.e., compression ratio, bypass ratio, turbine inlet temperature, overall pressure ratio, bleed, etc.). This will be a fairly extensive modification to PSM as these design parameters will affect the aerodynamics, mass properties, installed thrust and installed fuel flow data, all of which are used by PSM for performance calculations. In addition, PSM now accepts only installed propulsion data. All the design tradeoffs on an engine cycle and inlet/nozzle design parameters are currently performed separately from PSM by the propulsion group. These sensitivities will have to be incorporated into PSM.

By the time this phase has been reached, the number of design variables will very likely have exceeded the limit of 20 for the current PSM/CONMIN implementation. It appears that the design variable arrays will have to be extended past the 20 variable limit. Also, the follow-on effort will consider integrating a more advanced optimization program into PSM, in place of CONMIN.

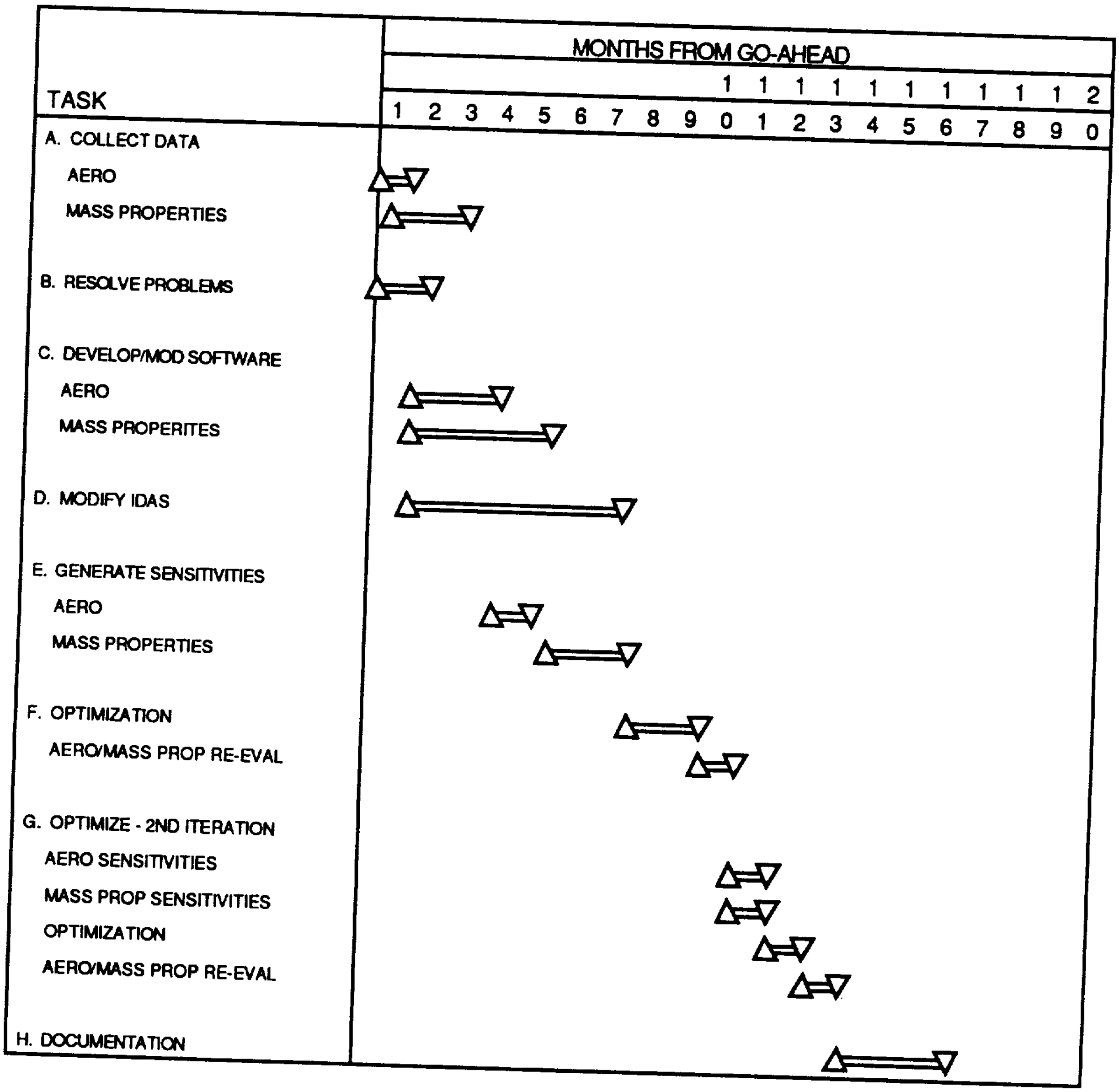


Figure 4 - 4. Conceptual Design Optimization - Overall Schedule

Section V

CONCLUSIONS

Rockwell breaks the aircraft design process into ten levels. This study focuses on Design Level III - Concept Selection. In theory, conceptual designs can be optimized around any figure-of-merit (takeoff gross weight, radar cross section, etc.). In practice conceptual designs are hopefully an optimum compromise of many competing figures-of-merit. The current design optimization capabilities at Rockwell focus around the more traditional trade studies of thrust to weight ratio versus wing loading, with a goal of minimizing the takeoff gross weight. Imposed on this are the traditional performance constraints of meeting a specified mission, energy maneuverability, load factor, takeoff and landing distances, turn rate and radius, acceleration, maximum speed and altitude, range and radius for alternate missions, etc. These trade study results are most often presented graphically. For these types of conceptual design trade studies, Rockwell has developed a computer system called the Integrated Design and Analysis System (IDAS). This system consists of an integrated suite of computer programs to perform conceptual design geometry definition (layouts), aerodynamics and mass properties analysis, mission performance analysis, vehicle sizing, and a trade study crossplotting capability. Recently the scope of what figures-of-merit make up mission effectiveness has grown geometrically. For example, low observables, super-maneuverability, hypersonic capability, vulnerability and supportability all vie with the traditional performance figures-of-merit. This has been further complicated by a multitude of new materials, structures, aerodynamics, propulsion and flight controls technologies that need to be assessed in any given conceptual design study. A necessary condition for successful optimization is a complete, quantitative understanding of all of these relationships. The challenge here is tremendous.

Applying the functional decomposition techniques to the conceptual design phase appears to be feasible. While many of the variables needed to perform detailed analysis (e.g. structural design of a wing rib) are not known at conceptual design, it is possible to get around this problem. This can be done by limiting the functional decomposition to a level at which the needed variables are available.

The initial implementation of the modified design process will optimize wing design variables (area, sweep, taper ratio, aspect ratio, thickness ratio, twist and camber). The modified design process proposed in this study is a hybrid approach. It combines functional decomposition techniques for generation of aerodynamic and mass properties linear sensitivity derivatives with existing techniques for aircraft sizing, mission performance and optimization. This hybrid approach will require 78 aerodynamic sensitivity derivatives, with most of these required at multiple mach numbers. Proposed approaches for generating the aerodynamic sensitivity derivatives include: (1) using empirical methods such as those available in the Configuration Analysis Module computer program, (2) generating a master database of wing sensitivity derivatives which can be used on multiple design studies, or (3) generating aerodynamic coefficients for a series of variations around the baseline design using existing methods. The hybrid approach will require 11 weight sensitivity derivatives. Proposed approaches to generating weight sensitivity derivatives include: (1) applying analytical techniques of the Structural Weight Estimation Program (modified as required) to the baseline design and parametric variations to it, and (2) generating a master database of wing sensitivity derivatives which can be used on multiple design studies. The hybrid approach will require modifications to the Parametric Synthesis Module computer program to accept aerodynamic and weight sensitivity derivatives and calculate resulting performance data. The hybrid approach will use the existing interface between the Parametric Synthesis Module and the CONMIN computer program for optimization of the wing design variables.

Several uncertainties remain to be resolved. These include: (1) automating sensitivity derivative generation to the point that it will become routine with the disciplines, (2) independence of sensitivities within a given discipline, (3) non-linearity of sensitivities and valid limits for linear derivatives, (4) whether higher order derivatives need to be included, and (5) mach number sensitivity of aerodynamic sensitivity derivatives.

Development of the modified design process will consist of the following tasks:

Task A - Background Research/Collect Data,
Task B - Refine Mathematical Models and Resolve Technical Problems,
Task C - Develop/Modify Computer Programs to Generate Aerodynamics and Mass Properties Sensitivities,
Task D - Modify the Integrated Design and Analysis System,
Task E - Generate Sensitivity Derivatives,
Task F - Optimize Test Case Wing Design Variables,
Task G - Optimization - Second Iteration,
Task H- Documentation.

The test case will be the Advanced Technology Multi-role Fighter, an early Rockwell concept for the Air Force Advanced Tactical Fighter. The development effort is estimated to require 17 months and 5120 engineering hours.

APPENDIX A
PARAMETRIC SYNTHESIS MODULE

The parametric Synthesis Module (PSM) has been identified in this report as the computer program which will integrate the wing design variable sensitivities from the aerodynamic and weight specialty disciplines, as well as performing the optimization. The purpose of this appendix is to briefly describe what PSM does and how it works.

PSM is one of the modules of the Rockwell Integrated Design and Analysis System (IDAS). Figure A-1 shows the overall IDAS organization. In addition to PSM, IDAS consists of the Configuration Definition Module (CDM), Configuration Analysis Module (CAM) and the Summary Report Module (SRM). The modules of IDAS are integrated through a combination of library and project data bases, controlled by a file manager.

PSM is an interactive graphic computer program which synthesizes parametric aircraft designs from a known baseline design, and computes a performance analysis of the parametric design. Parametric design parameters include wing loading, thrust-to-weight ratio, payload weight and volume, and gross weight or fuel weight. An internal search routine allows sizing the parametric design to perform a design mission. During the resizing operation, geometry, weights, and aerodynamics of a baseline vehicle are scaled using relations contained in PSM. Variables scaled outside prescribed limits may be flagged and identified on printed and terminal output.

Performance calculations include design and alternate mission profiles; maneuverability characteristics at several speeds, altitudes, and load factors; and take-off and landing calculations for multiple conditions. An option to compute performance analysis only may be selected which permits using a partial input data file.

PSM may be used to generate design, requirement, and sensitivity tradeoffs by systematic variation of design, requirement, and scaling parameters. Selected performance results from each case can be accumulated in a summary output file along with the values of the trade parameters. Graphics output from the summary file may be obtained during module execution, or the summary file may be used as input to the Summary Report Module (SRM). Interfacing routines within the PSM allow input of results produced by other IDAS modules. Baseline geometry, weight and aerodynamic properties produced by the Configuration Analysis Module (CAM), may be used as input to PSM. Alternatively, independently developed baseline analyses may be substituted for the CAM output.

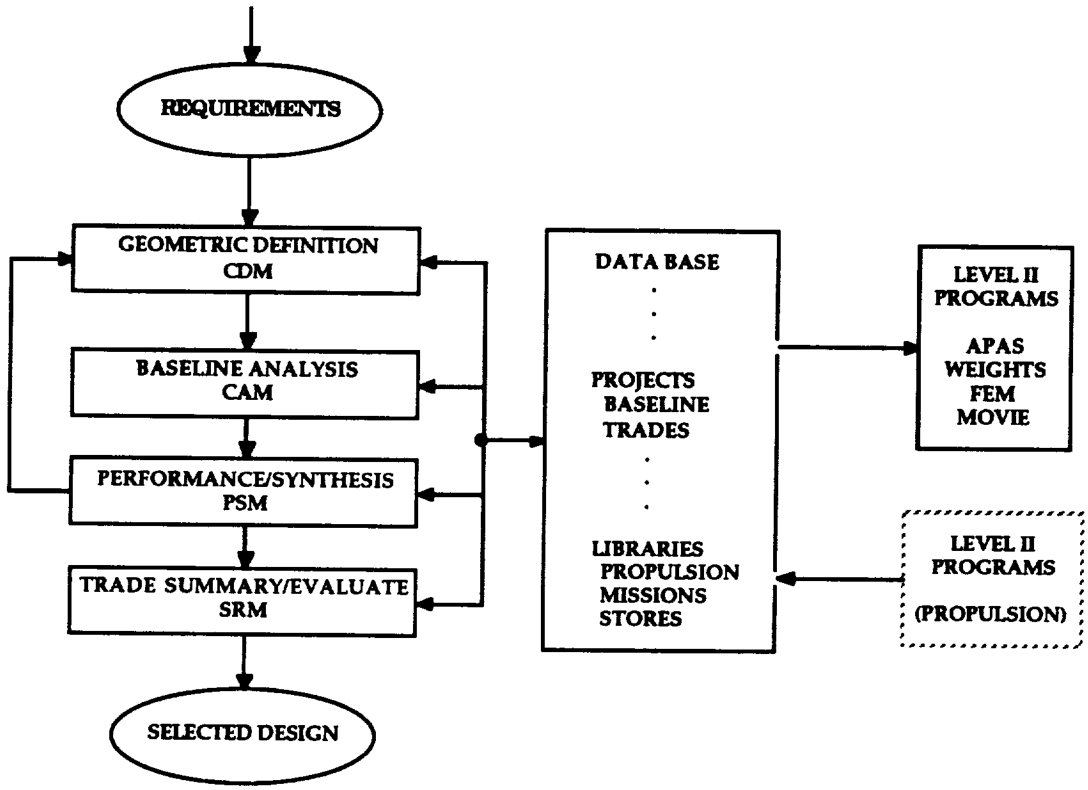


Figure A-1. Overall IDAS Organization

Output interfacing includes the ability to feed trade summaries to the SRM, and geometric scaling parameters to the Configuration Definition Module (CDM). This latter feature may be used to update the baseline geometry during execution of the CDM.

The basic elements of the Parametric Synthesis Module consist of those models required to estimate aircraft characteristics plus the control and search routines needed to tie these models together. Aircraft characteristics may be divided into design characteristics and performance characteristics. Vehicle sizing includes propulsion and airframe subsystem sizing, as well as overall vehicle sizing.

Figure A-2 is a functional flow diagram which shows how the search and control subroutines link individual models together. The design convergence loop is provided to ensure convergence of the design weight, which may depend on the design mission and does depend on the weight and geometry of several vehicle components. The mass/volume convergence loop ensures convergence of fuel weight when gross weight is known or gross weight when fuel weight is known. Fuselage volume convergence is also ensured by this loop. A radius search loop is also included within the mission performance analysis to assure that mission fuel used is equal to fuel available. Overall vehicle sizing to specified mission requirements requires a search, called the requirements convergence loop in figure A-2, which exercises the design and performance analysis modules to converge on a given mission range, radius, or time.

Each pass (i.e. each case) processed through the parametric synthesis and analysis control routine produces a parametric aircraft design and a complete performance analysis of that design. Major performance and design parameters are then saved (as one row) in the summary matrix. Execution of multiple cases produces a set of designs which may be listed, plotted, or saved in an output file by execution of the appropriate utility subprogram. A summary matrix saved on an earlier run may be included as input. New cases will then be added to the previous ones.

Indexing occurs at the start of each job submittal. Baseline data is input from the CAM (or external sources, and includes geometry, aerodynamics, and weights), the propulsion file (library), and the primary synthesis files. The indexing routine then executes internal geometry, weight and aerodynamic routines. Results from these analyses are then indexed to agree with baseline values by calculation of indexing (alpha) factors. This process is applied to cross-section and wetted areas, volumes, component weights, and friction drags. An incremental drag correction curve is also generated to make internally generated drag polars agree with baseline polars.

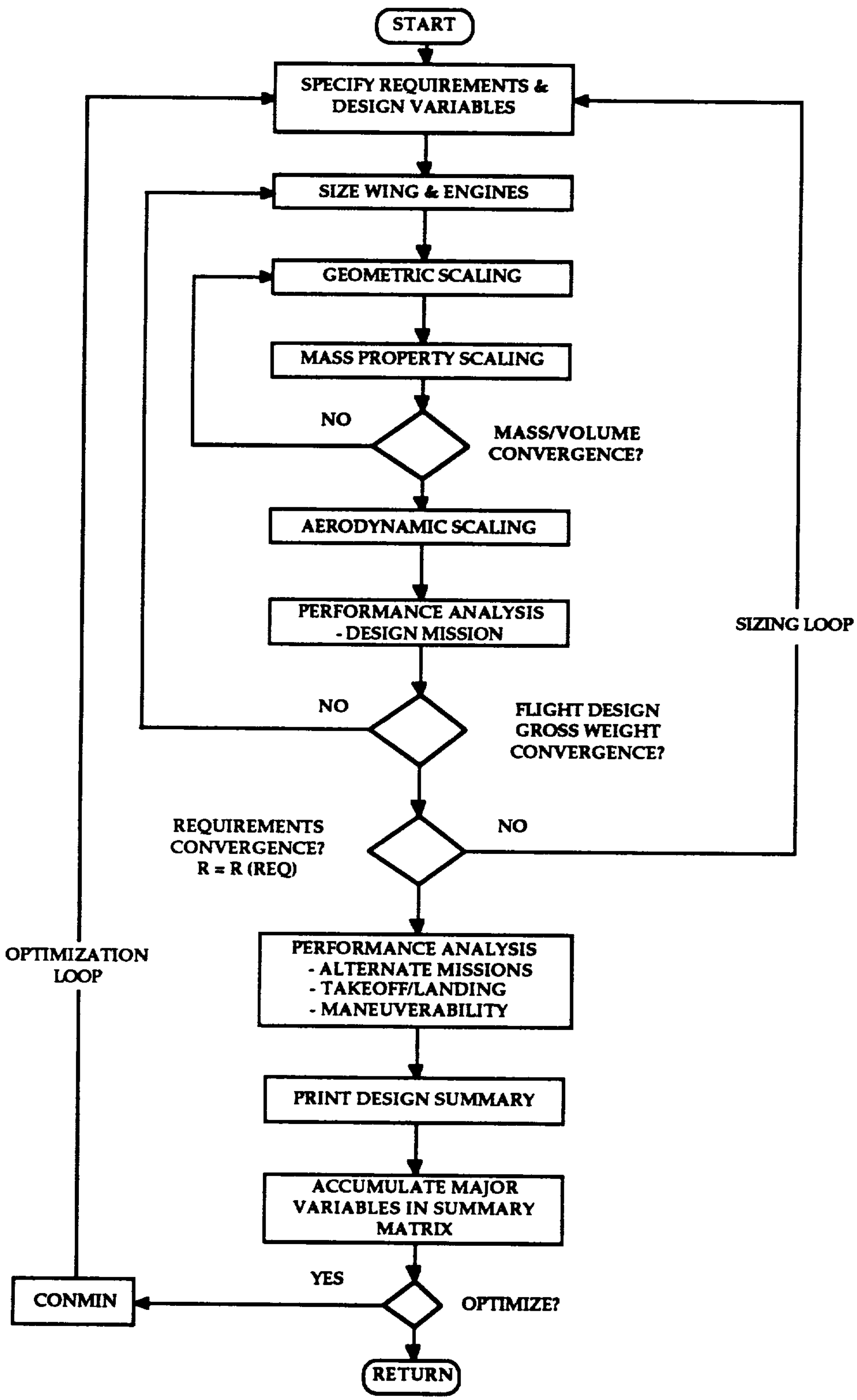
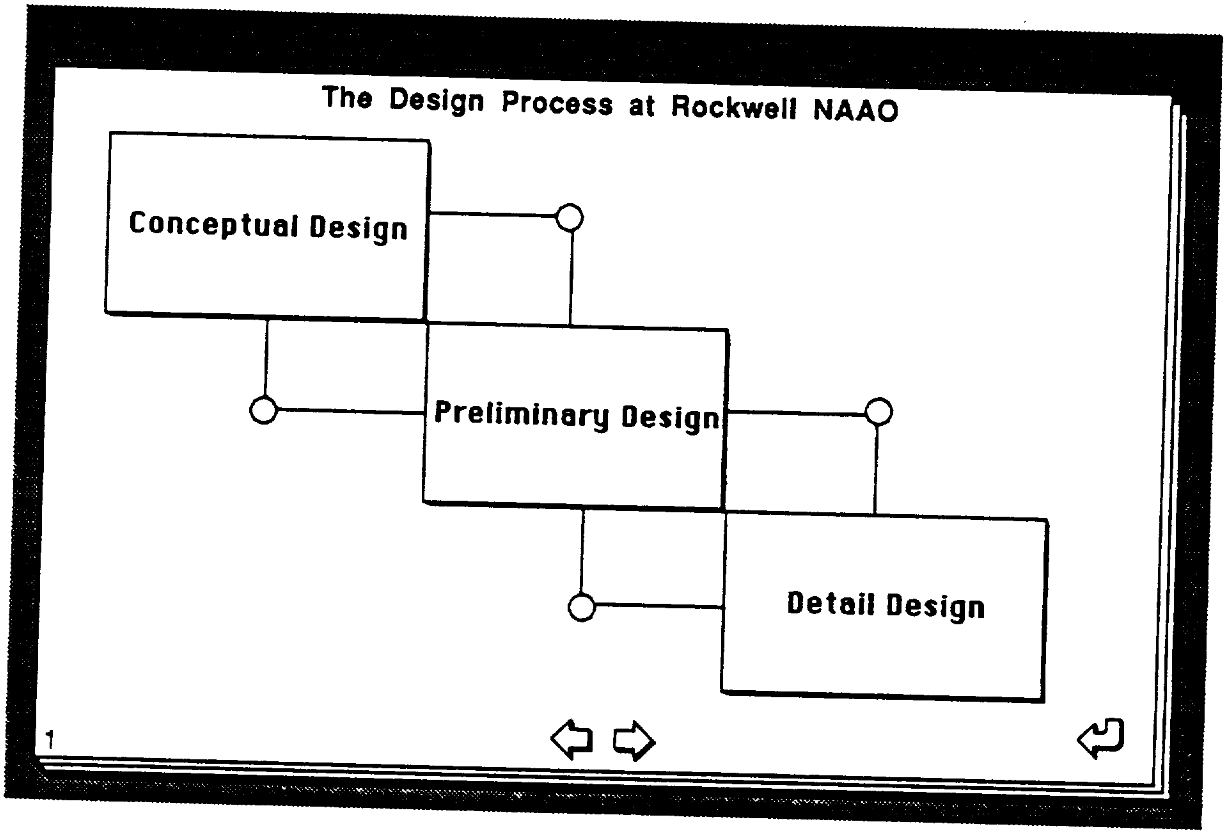
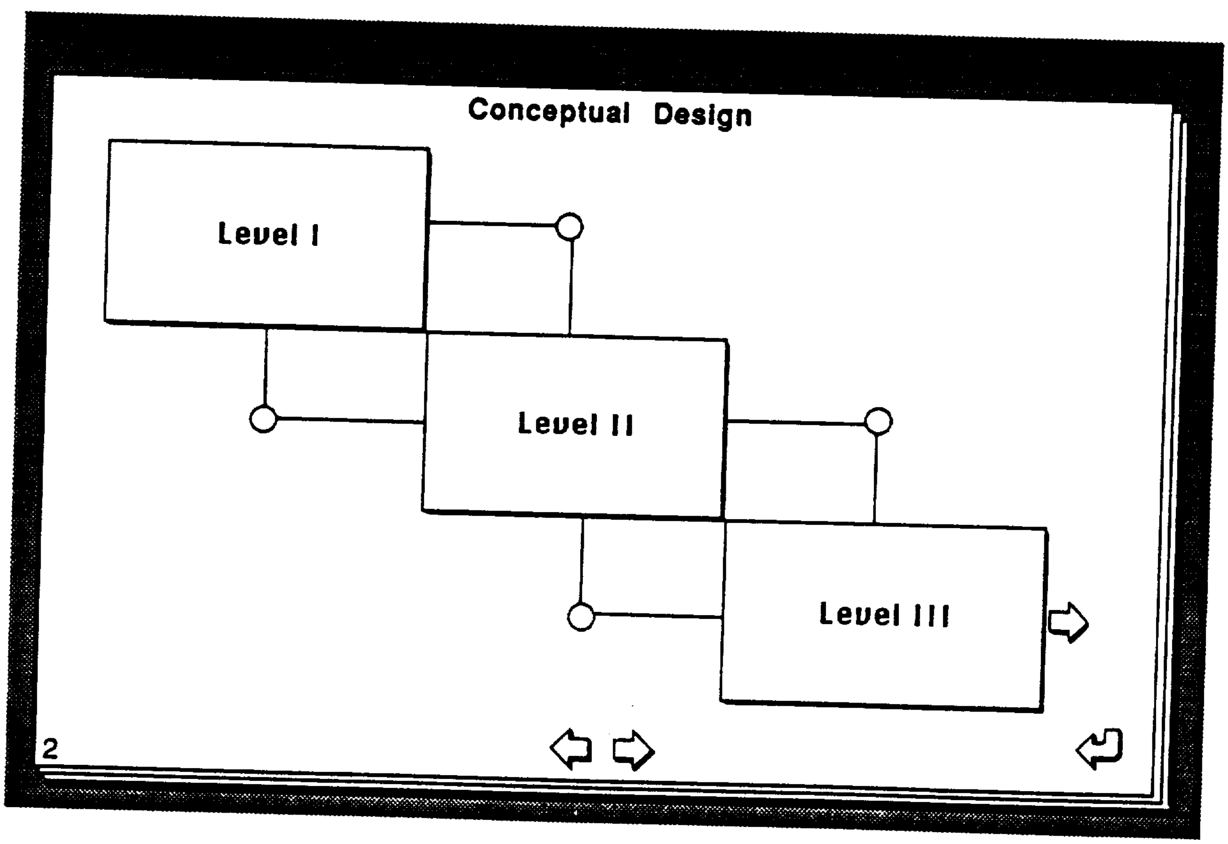


Figure A-2. PSM Functional Flow Diagram

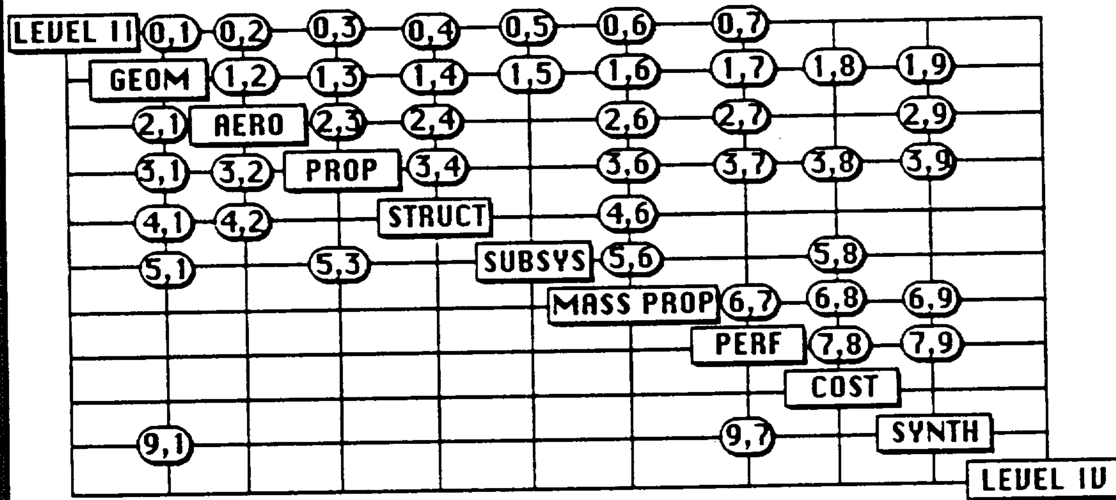
APPENDIX B

HYPERCARD™ PRINTOUT


The Design Process at Rockwell NAAO
Conceptual Design
Preliminary Design
Detail Design
1


Conceptual Design
Level I
Level II
Level III
2

Level III - Configuration Selection
N-Squared Diagram



3

GEOMETRY PROCESS

TASK: (1) Establish design assumptions / goals
(2) Conduct preliminary sizing
(3) Define wing geometry, subject to constraints
(4) Establish major subsystems shape and volume
(5) Define fuselage envelope (external lines), subject to constraints
(6) Establish internal arrangement, subject to constraints
(7) Define control surface geometry / location, subject to constraints
(8) Define landing gear geometry / location, subject to constraints

FIGURE(S) OF MERIT (also goals and constraints:
weight, fuel fraction, L / D, RCS, maneuverability, speed, static margin or CG location, payload weight and volume, range / time, fixed equipment weight and volume, vulnerability level

CONTROL VARIABLES:
all external geometry, locations of subsystems internally, fuel tank geometry, engine performance / size, technology level, inlet and nozzle geometry

4

DATA (0,1)

From (process): DESIGN LEVEL II
To (process): GEOMETRY

AERO GOALS
TAKEOFF GROSS WEIGHT GOAL
THRUST AND FUEL FLOW GOALS
ANTICIPATED FUEL VOLUME / WEIGHT REQUIRED
PAYLOAD TYPE / SIZE
FLIGHT ENVELOPE (MAX SPEED, DESIGN SPEED, ETC.)
ANY KNOWN SUBSYSTEMS / SENSORS
SURVIVABILITY CONSIDERATIONS (ALLOWANCES FOR ARMOR, ETC.)
RCS GOALS
NUMBER OF CREW
PILOT VISIBILITY REQUIREMENTS
RUNWAY REQUIREMENTS
TECHNOLOGY TO BE INCORPORATED (EG ACTIVE FLEXIBLE WING)

5

DATA (3,1)

From (process): PROPULSION
To (process): GEOMETRY

ENGINE SIZE / TYPE (TO MEET THRUST AND FUEL FLOW GOALS)
NEEDED INLET CAPTURE AREA
NEEDED NOZZLE AREA AND BYPASS FLOW AREA

SUGGESTED CHANGES TO INLET / NOZZLE TYPE OR GEOMETRY
SUGGESTED CHANGES TO ENGINE INSTALLATION

6

DATA (5,1)

From (process): SUBSYSTEMS
To (process): GEOMETRY

VOLUMES / DIMENSIONS FOR SUBSYSTEMS

EXTERNAL STORES / PAYLOAD DIMENSIONS

INTERNAL PAYLOAD

SENSOR(S) DIMENSIONS / LOOK ANGLES

CONSTRAINTS
SENSOR LOCATION(S)

7

DATA (2,1)

From (process): AERO
To (process): GEOMETRY

SUGGESTED GEOMETRY CHANGES
WING PLANFORM SIZE OR SHAPE
VOLUME DISTRIBUTION
TWIST AND CAMBER DISTRIBUTION
AIRFOIL TYPE, THICKNESS
HORIZONTAL TAIL / CANARD / VERTICAL TAIL SIZE, SHAPE, LOCATION

CONTROL SURFACE(S) AND HIGH LIFT DEVICE(S) DEFINITION

8

DATA (4,1)

From (process): STRUCTURES
To (process): GEOMETRY

SUGGESTED GEOMETRY CHANGES
- FUEL TANK LOCATIONS
- SUBSYSTEM LOCATIONS
- PLANFORM
- THICKNESS, TWIST, CAMBER
- ENGINE MOUNTING (PYLONS)

STRUCTURAL CONCEPT GEOMETRY
- 3-VIEW DRAWING
- COMPUTER FILE?

9

DATA (9,1)

From (process): SYNTHESIS
To (process): GEOMETRY

FUEL VOLUME REQUIRED (MAYBE FUEL WEIGHT)

SUGGESTED GEOMETRY CHANGES
- FUSELAGE LENGTH OR FUSELAGE LENGTH, WIDTH, HEIGHT
- WING AREA AND LOCATION
- CONTROL SURFACE AREA
- ENGINE SIZE AND LOCATION

10

AERODYNAMICS PROCESS

TASK: Determine the vehicle external shape that yields the best aerodynamics, within the given constraints

FIGURE(S) OF MERIT: L/D, zero lift drag, stability, control effectiveness, flutter divergence velocities, trim drag, aero heating rates, maximum temperature, boundary layer transition

CONTROL VARIABLES: - - early conceptual design phase - -
Planform (shape, whether canard, tailess, etc), fineness ratio (volume distribution),
control philosophy (surfaces, control power, thrust vectoring, static margin)
- - late conceptual design and preliminary design phases - -
Lifting surface thickness, refined volume distribution
twist, camber, deformed shape

11

DATA (0,2)

From (process): DESIGN LEVEL II
To (process): AERO

FLIGHT ENVELOPE LIMITS

CONSTRAINTS
EXPECTED MISSION (SPEED, ALTITUDE, TURN PERFORMANCE, ETC.)
TECHNOLOGY BASE
HISTORICAL DATA
IR&D TECHNOLOGY STUDIES
THEORETICAL METHODS

12

DATA (1,2)

From (process): GEOMETRY
To (process): AERO

3-VIEW DRAWING - EXTERNAL LINES
OR CDM GEOMETRY FILE
OR APAS GEOMETRY FILE

FOR CAM AERO NEED CDM.OUTPUT FILE
THE FOLLOWING GEOMETRY COMPONENTS NEED TO BE DEFINED:
- WING PLANFORM
- AIRFOIL
- FUSELAGE
- NACELLE(S)
- CANOPY
- HORIZONTAL TAIL/CANARD
- VERTICAL TAIL
- PYLON(S)
- INLET(S)/CAPTURE AREA
- NOZZLE(S)

Continue

13

DATA (1,4) -card 2-

From (process): GEOMETRY
To (process): AERO

-BOUNDRY LAYER DIVERTER

CONSTRAINTS:
- EXTERNAL STORES
- VOLUMES FOR SYSTEMS/PAYLOAD/FUEL

14

DATA (3,2)

From (process): PROPULSION
To (process): AERO

THRUST DEPENDENT LIFT AND DRAG
REAL GAS EFFECTS
NOZZLE BASE DRAG AND PLUME EFFECTS
FLOWFIELD CHANGES

15

DATA (4,2)

From (process): STRUCTURES
To (process): AERO

CONSTRAINTS
  FLEXIBLE (LOADED) SHAPE LIMITS (TWIST / CAMBER)
  MINIMUM T / C (AND OTHER PLANFORM PARAMETERS)

16

PROPULSION PROCESS

TASK(S): (1) Determine inlet pressure recovery and drag throughout flight regime
(2) Determine engine cycle characteristics and engine airflow, thrust, fuel consumption, weight
(3) Determine nozzle internal thrust coefficient and external drags
(4) Determine installed thrust, fuel flow, thermal loads, acoustic loads and thrust cycle loads throughout operating regime

FIGURE(S) OF MERIT: Weight, performance, observables, distortion, thrust, fuel consumption, life, maintainability, life cycle cost

CONTROL VARIABLES: Compression surface position, terminal shock position, throttle angle, fuel flow, stator angles, nozzle areas, nozzle vector angle, variable cycle features, throat area, exit area, vector angle, speed, altitude, power setting, angle of attack, bypass ratio, compression ratio, efficiencies, burner temperatures

CONSTRAINTS: Momentum, energy, mass, temperature, thermo-dynamic properties

17

DATA (0,3)

From (process): DESIGN LEVEL II
To (process): PROPULSION

FLIGHT ENVELOPE (speed, altitude, angle of attack, power setting)
THRUST GOALS AND DESIGN POINTS
FUEL FLOW GOALS AND DESIGN POINTS

CONSTRAINTS
- TECHNOLOGY BASE
  - HISTORICAL DATA
  - ENGINE MANUFACTURER IR&D TECHNOLOGY STUDIES
  - THEORETICAL METHODS (INLETS, NOZZLES, ENGINE CYCLE)
  - ENGINE DECKS AVAILABLE
  - WHETHER EXISTING OR PARAMETRIC ENGINE

18

DATA (1,3)

From (process): GEOMETRY
To (process): PROPULSION

INLET GEOMETRY (TYPE / SIZE)
NOZZLE GEOMETRY (TYPE / SIZE)
ENGINE INSTALLATION PARAMETERS (DUCT LENGTH, ETC.)

CONSTRAINTS
  ENGINE DIMENSIONS (DEPENDS ON THE STUDY)
  EXISTING OR PARAMETRIC ENGINE

19

DATA (2,3)

From (process): AERO
To (process): PROPULSION

ANGLE OF ATTACK AT DESIGN SPEED(S) AND ALTITUDE(S)
ENGINE INLET AIR CONDITIONS
RAM DRAG

20

DATA (5,3)

From (process): SUBSYSTEMS
To (process): PROPULSION

CONSTRAINTS: BLEED AND POWER EXTRACTION REQUIREMENTS

21

STRUCTURES PROCESS

TASK: Optimize existing finite element model, subject to defined physical constraints (to include buckling analysis for static loading, vibration analysis for dynamic loading, generating aero loads (AIC matrix), identifying flutter and divergence modes, maximizing flutter and divergence speeds, thermal analysis, panel buckling, general buckling)

FIGURE(S) OF MERIT: Structural weight, stiffness / flexibility, flutter speed, stress concentration
(note: writeup from structures did not identify figures of merit)

CONTROL VARIABLES: Stress, strain, gauge sizing, laminate material properties, laminate strains, laminate geometric orientation

22

DATA (0,4)

From (process): DESIGN LEVEL II
To (process): STRUCTURES

MANEUVER LOADS
MISSION PROFILE (TRAJECTORY)
FLIGHT ENVELOPE
TEMPERATURES / HEATING RATES
DYNAMIC PRESSURE
PAYLOAD
DESIGN GROSS WEIGHT GOALS
SURVIVABILITY CONCEPT (BATTLE DAMAGE TOLERANCE)

CONSTRAINTS
STRUCTURAL LOAD MARGINS (DESIGN VERSUS ULTIMATE)

23

DATA (1,4)

From (process): GEOMETRY
To (process): STRUCTURES

EXTERNAL SHAPE IN THE FORM OF SURFACE POINTS

MASS DISTRIBUTION IN THE FORM OF INTERNAL SUBSYSTEM AND EXTERNAL STORES LOCATIONS

CONSTRAINTS
INTERNAL SUBSYSTEM LOCATIONS WHICH WOULD AFFECT THE STRUCTURAL ARRANGEMENT (EG FUEL TANK AND ENGINE IMPACT ON WING CARRYTHROUGH STRUCTURE)

24

DATA (2,4)

From (process): AERO
To (process): STRUCTURES

GOALS FOR LOADED STRUCTURAL SHAPE
(EG TWIST AND CAMBER)

STATIC AERO LOADS

DYNAMIC AERO LOADS

AERO-HEATING RATES

25

DATA (3,4)

From (process): PROPULSION
To (process): STRUCTURES

ACOUSTIC VIBRATIONS - PANEL FLUTTER (dB / LOAD)

THERMAL LOADS (HEATING RATES)

STATIC AND CYCLIC (THRUST) LOADS

26

SUBSYSTEMS PROCESS

TASK(S):
(1) Translate general system operational requirements into specific subsystems performance reqs
(2) Allocate specific performance requirements to various subsystems of the proposed A/C design
(3) Define candidate subsystems of the proposed aircraft design
(4) Determine performance capability of each candidate subsystem identified in (3)
(5) Select integrated sensor system which will achieve the mission and performance requirements, subject to constraints

FIGURE(S) OF MERIT:
(1) Measures of effectiveness (MOEs) for each subsystem (mean detection range, etc)
(2) Maintainability (MTTR), reliability (MTBF)
(3) Cost (DTLCC), schedule risk, technical risk
(4) Cost, DTLCC, error budget, CEP, MTBF, MTTR, survivability
(5) Probability of acquisition

CONTROL VARIABLES:
(1) Performance of each subsystem (eg radar range, A/C speed, power, scan vol, etc.)
(2) Bandwidths, data rates, other interface requirements
(3) none identified
(4) A/C trajectory, subsystem variables (INS-position drift, drift rate, Radar-ave power)
(5) A/C trajectory, selection criteria (for cost, weight, volume, power, etc.)

DATA (0,5)

From (process): DESIGN LEVEL II
To (process): SUBSYSTEMS

PERFORMANCE REQUIREMENTS FOR EACH SUBSYSTEM / WEAPON
WEIGHT / COST / RELIABILITY GOALS
AIRCRAFT FLIGHT ENVELOPE
RESULTS OF OPERATIONS ANALYSIS
CUSTOMER REQUIREMENTS
OPERATIONAL REQUIREMENTS
AIRCRAFT TRAJECTORY (FLIGHT PATH, ATTITUDE, POSITION, ALTITUDE, VELOCITY, LOAD FACTOR, RANGE TO TARGET)
SURVIVABILITY CONCEPT
APPROPRIATE MEASURES OF EFFECTIVENESS (IF NOT DERRIVED INTERNALLY)

CONSTRAINTS
POWER AVAILABLE
COST / WEIGHT UPPER LIMITS

DATA (1,5)

From (process): GEOMETRY
To (process): SUBSYSYSTEMS

CONSTRAINTS
- INTERNAL VOLUME AVAILABLE
- UNAVAILABLE LOCATIONS
- INTERFERENCES FROM OTHER COMPONENTS

29

MASS PROPERTIES PROCESS

TASK: Determine vehicle component and total system weight, balance and inertia

FIGURE(S) OF MERIT: Weight, balance within aerodynamic limits

CONTROL VARIABLES: Balance limits, fuel tank arrangement, payload internal / external location, gear retraction, emergency / safety / redundancy, growth, performance environment

30

DATA (0,6)

From (process): DESIGN LEVEL II
To (process): MASS PROP

VEHICLE LIMITS (MANEUVER, Q, ALTITUDE, TEMPERATURE
OPERATION / MISSION (PAYLOAD, CREW, USAF / USN, BASING)
THREATS
SURVIVABILITY CONCEPT (OBSERVABLES, ARMOR, EMP, ECM, FUEL PROTECTION)

CONSTRAINTS
ANY OR ALL OF THE ABOVE

31

DATA (1,6)

From (process): GEOMETRY
To (process): MASS PROP

EXTERNAL SHAPE IN THE FORM OF RELEVANT GEOMETRY PARAMETERS
(EG PLANFORM AREA, ASPECT RATIO, ETC.)
-- THERE ARE LOTS OF THESE --

FUEL TANK SIZE / LOCATION / VOLUME

PROPULSION INTEGRATION CONCEPT

INTERNAL SYSTEM ARRANGEMENT

CONSTRAINTS
SOME OR ALL OF THE ABOVE GEOMETRY INPUTS (DEPENDS ON THE STUDY)

32

DATA (2,6)

From (process): AERO
To (process): MASS PROP

AERODYNAMIC STABILITY LIMITS (ACCEPTIBLE CG LIMITS)

CONTROL SURFACE DESIGN
ASSUMPTIONS FOR MAX LIFT COEFFICIENT
CONCEPT FOR HIGH LIFT DEVICES (SINGLE / DOUBLE SLOTTED FLAPS, ETC.)

33

DATA (3,6)

From (process): PROPULSION
To (process): MASS PROP

ENGINE SIZE OR ALTERNATELY ENGINE WEIGHT
ENGINE SCALE FACTOR
FUEL CONSUMPTION
ENGINE TECHNOLOGY
ENGINE SCALING RELATIONSHIPS (IF UNIQUE)

34

DATA (4,6)

From (process): STRUCTURES
To (process): MASS PROP

STRUCTURAL CONCEPT
MATERIAL TYPES / MIX
THICKNESSES
PRELIMINARY STRUCTURE SIZING

35

DATA (5,6)

From (process): SUBSYSTEMS
To (process): MASS PROP

SUBSYSTEM DIMENSIONS OR OTHER FIGURES OF MERIT
(EG POWER OUTPUT) THAT CAN BE RELATED EMPIRICALLY TO WEIGHT
[or]
SUBSYSTEM WEIGHTS

SECONDARY POWER CONCEPTS

36

PERFORMANCE PROCESS

TASK(S): Determine the performance capability of a proposed aircraft design when operating over a required mission profile (to include mission, maneuverability, takeoff and landing)

FIGURE(S) OF MERIT: Range, radius or endurance, specific excess power, max sustained g's, takeoff distance, landing distance (these can be constraints)

CONTROL VARIABLES: Speed, altitude, power setting, angle of attack, bank angle

37

DATA (0,7) --page 1--

From (process): DESIGN LEVEL II
To (process): PERFORMANCE

MANEUVER PERFORMANCE, DESIGN POINTS, PERFORMANCE GOALS
TAKEOFF AND LANDING DISTANCE GOALS
MAX MACH NUMBER
MAX DYNAMIC PRESSURE
RANGE / RADIUS GOALS

DETAILED MISSION DEFINITION:
- MAX ALLOWABLE CLIMB (CUTOFF) DISTANCE
- MIN ALLOWABLE VALUE OF PRIMARY MISSION VARIABLE
- MIN ESTIMATED VALUE OF PRIMARY MISSION VARIABLE
- DELTA FIXED USEFUL LOAD
- DELTA FUEL LOAD
- LEG NUMBER OF LAST LEG
- RESERVE FUEL (FRACTION OF INITIAL FUEL)
- PRIMARY SEQUENCE MISSION CONTROL VARIABLE

Continue

38

DATA (0,7) --page 2--

From (process): DESIGN LEVEL II
To (process): PERFORMANCE

FIXED SEQUENCE #3 DISTANCE (NM OR TIME)
FIXED SEQUENCE #4 DISTANCE (NM OR TIME)
LET TYPE CONTROL FOR EACH LEG

FOR EACH RESERVE / WARMUP / TAXI / TAKEOFF / COMBAT / DROP / DROP / DUMMY LEG
LEG TYPE
WEIGHT FLAG
WEIGHT (OPTIONAL)
MACH, ALTITUDE FLAG
MACH AT END OF LEG
ALT AT END OF LEG
DROP WEIGHT
PAX CODE
SV CODE
DISTANCE OR TIME (PICK ONLY ONE)

Continue

39

DATA (0,7) --page 3--

From (process): DESIGN LEVEL II
To (process): PERFORMANCE

FOR EACH CLIMB LEG
LEG TYPE (CLIMB OR DESCENT)
FLIGHT PATH OR SPECIAL FLIGHT LIMITS
TYPE OF CLIMB / DESCENT (MIN TIME, FUEL, CONSTANT THROTTLE)
INITIAL SPEED / ALTITUDE
INTEGRATION INTERVAL
FINAL MACH
FINAL ALTITUDE
PAX CODE
SV CODE
DISTANCE OR TIME (PICK ONLY ONE)
FOR EACH CRUISE / LOITER / TURN LEG
LEG TYPE
WEIGHT FLAG
ALTITUDE FLAG

Continue

40

DATA (0,7) --page 4--

From (process): DESIGN LEVEL II
To (process): PERFORMANCE

RATE OF CLIMB MINIMUM
MACH MINIMUM
MACH MAXIMUM
TYPE OF CRUISE / LOITER / TURN
LOAD FACTOR
PAX CODE
SV CODE
DISTANCE OR TIME (PICK ONLY ONE)

Continue

41

DATA (1,7)

From (process): GEOMETRY
To (process): PERFORMANCE

NUMBER OF ENGINES (OF EACH TYPE)
REFERENCE WING AREA
THRUST INCIDENCE (FOR EACH ENGINE)
FUSELAGE LENGTH
TOTAL AIRPLANE WETTED AREA

42

DATA (2,7)

From (process): AERO
To (process): PERFORMANCE

CDL vs CL vs MACH NUMBER vs AIRCRAFT CONFIGURATION (AC)
CD0 vs MACH vs AC
[or]
CD0 vs MACH vs AC
CDK vs MACH vs AC
CLK vs MACH vs AC
K vs MACH vs CL vs AC
[or]
CD-wave vs MACH vs AC
MACH-drag divergence vs CL vs AC
CD-boundry layer diverter vs MACH vs AC
CL0 vs MACH vs AC
CL-alpha vs MACH vs AC
ALPHA-CL-max vs MACH vs AC
CL-max vs MACH vs ALTITUDE vs AC

Continue

43

DATA (2,7) -card 2-

From (process): AERO
To (process): PERFORMANCE

CD-total vs CL vs MACH vs AC
CD-landing gear vs ALPHA vs AC
D/Q vs MACH vs AC (FOR EXTERNAL STORES)
CD-friction @ REFERENCE MACH, ALTITUDE, AC

ALSO TRIMMING LIFT, DRAG, PITCHING MOMENT (OPTIONAL)

44

DATA (3,7)

From (process): PROPULSION
To (process): PERFORMANCE

ENGINE SCALE FACTOR (OR ENGINE SIZE)
INSTALLED THRUST (fixed power) vs MACH NUMBER vs PRESSURE (ALT) (FOR 1, 100% ENGINE)
INSTALLED FUEL FLOW (FIXED POWER) vs MACH NUMBER vs PRESSURE (ALT) (FOR 1, 100% ENGINE)
INSTALLED FUEL FLOW (PART POWER) vs POWER SETTING vs PRESSURE (ALT) vs MACH NUMBER (FOR 1, 100% ENGINE)
RAM DRAG vs MACH NUMBER vs PRESSURE (ALT) (FOR 1, 100% ENGINE)

NOTE: FOR HYPERSONIC AIR-BREATHING CONFIGURATIONS NEED INLET FORCE, INLET FORCE ANGLE, NOZZLE FORCE AND NOZZLE FORCE ANGLE vs MACH NUMBER vs PRESSURE (ALT) vs EQUIVELENCE RATIO vs ANGLE OF ATTACK (FOR 1, 100% ENGINE)

45

DATA (6,7)

From (process): MASS PROP
To (process): PERFORMANCE

FUEL WEIGHT
PAYLOAD WEIGHT
TAKEOFF GROSS WEIGHT
OTHER EQUIPMENT WEIGHT (= TAKEOFF GROSS WT - FUEL WT - PAYLOAD WT)

46

DATA (9,7)

From (process): SYNTHESIS
To (process): PERFORMANCE

NEW TOGW
NEW FUEL WEIGHT
NEW AERODYNAMICS
NEW MASS PROPERTIES
NEW ENGINE SCALE FACTOR

47

COST PROCESS

TASK: (1) MINIMIZE THE COSTS OF PROCURING A NEW AIRCRAFT SYSTEM
(2) DETERMINE THE COST OF AN AIRCRAFT CONCEPT, SUBJECT TO A SPECIFIED OPERATIONS CONCEPT AND PRODUCTION CONCEPT

FIGURE(S) OF MERIT: LIFE CYCLE COST
FLY AWAY COST
ACQUISITION COST

CONTROL VARIABLES:

LABOR RATES
OVERHEAD RATES
MATERIAL RATES
RECURRING O&S COSTS
RDT&E COSTS
PRODUCTION RATES / QUANTITIES
MANUFACTURING COMPLEXITY

48

DATA (1,8)

From (process): GEOMETRY
To (process): COST

WING AREA
EMPENNAGE AREA
TOTAL WETTED AREA
HORIZONTAL (CANARD) SPAN
LENGTH OF AIRCRAFT
NUMBER OF ENGINES
THRUST PER ENGINE (INCLUDING A / B)
NOZZLE TYPE
VARIABLE WING-SWEEP FACTOR
AIRCRAFT TYPE

49

DATA (3,8)

From (process): PROPULSION
To (process): COST

FUEL FLOW FACTOR
THRUST PER ENGINE
PROPULSION SYSTEM PRODUCTION

50

DATA (5,8)

From (process): SUBSYS
To (process): COST

AVIONICS PRODUCTION COSTS
AUXILIARY POWER SYSTEM RDT&E HOURS
CREW SYSTEM RDT&E HOURS
ARMAMENT SYSTEM RDT&E HOURS

51

DATA (6,8)

From (process): MASS PROP
To (process): COST

FUSELAGE WEIGHT
WING WEIGHT
EMPENNAGE WEIGHT
NACELLE WEIGHT
LANDING GEAR WEIGHT
FUEL SYSTEM WEIGHT
ELECTRICAL SYSTEM WEIGHT
AUXILIARY POWER SYSTEM WEIGHT
HYDRAULIC AND PENUMATIC SYSTEM WEIGHT
CREW ACCOMMODATIONS WEIGHT
INSTRUMENT WEIGHT
FLIGHT CONTROLS WEIGHT
ARMAMENT SYSTEM WEIGHT
AIR INDUCTION CONTROL SYSTEM WEIGHT
TAKEOFF GROSS WEIGHT

CONTINUE

52

DATA (5,8) --PAGE 2--

From (process): MASS PROP
To (process): COST

TAKEOFF GROSS WEIGHT
EMPTY WEIGHT
AMPR WEIGHT
STRUCTURES WEIGHT
FUEL WEIGHT
EQUIPMENT GROUP WEIGHT
ENGINE WEIGHT
AVIONICS WEIGHT
RDT&E MATERIAL MIX (FUSELAGE, WING, TAIL, NACELLE)
PRODUCTION MATERIAL MIX (FUSELAGE, WING, TAIL, NACELLE)

53

DATA (7,8)

From (process): PERFORMANCE
To (process): COST

MAX G'S
MAX DYNAMIC PRESSURE
MAX MACH NUMBER
MISSION TYPE
- CLOSE AIR SUPPORT
- BOMBER
- AIR SUPERIORITY
- INTERDICTION
- INTERCEPTOR
- MULTIPLE

54

SYNTHESIS PROCESS

TASK(S): (1) Size to design mission requirements (determine vehicle size - minimum gross weight - such that mission performance requirements are satisfied

(2) Requirement trade studies

(3) Design trade studies

(4) Size to performance requirements

FIGURE(S) OF MERIT: (PRIMARY) Design takeoff gross weight

SECONDARY - - (also constraints): Range / Radius, Specific Excess Power, Sustained G's, Takeoff Distance Landing Distance, Max G's

CONTROL VARIABLES: PRIMARY: T / W, W / S

SECONDARY: Engine scale factor, reference wing area , wetted area, fuselage fineness ratio, propulsion system volume, total airplane frontal area, tail volume coefficient, fuselage length, width, height, frontal area, volume, wetted area, length of center section, wing volume, nacelle volume

55

DATA (1,9)

From (process): GEOMETRY

To (process): SYNTHESIS

BASELINE VALUES FOR:

- WING AREA
- WING FUEL VOLUME
- NACELLE VOLUME
- NACELLE MAX FRONTAL AREA
- NACELLE TOTAL WETTED AREA
- NACELLE LENGTH
- FUSELAGE VOLUME REQUIRED
  - FIXED LOAD VOLUME
  - NACELLE VOLUME
  - LANDING GEAR BAY VOLUME
  - FUEL VOLUME
  - ENGINE VOLUME (IF NOT IN NACELLE)
- FUSELAGE FINENESS RATIO
- FUSELAGE MAX FRONTAL AREA
- FUSELAGE TOTAL WETTED AREA
- FUSELAGE LENGTH

- - PLUS ALL GEOMETRY INPUTS NEEDED BY MASS PROPERTIES - - THERE ARE LOTS OF THESE!

56

DATA (2,9)

From (process): AERO
To (process): SYNTHESIS

BASELINE:
MAX LIFT COEFFICIENT VERSUS MACH NUMBER
LIFT COEFFICIENT AT ZERO ALPHA VERSUS MACH NUMBER
C - L - ALPHA VERSUS MACH NUMBER
MAX ALPHA VERSUS MACH NUMBER
LANDING GEAR DRAG COEFFICIENT VERSUS ALPHA
DRAG DIVERGENCE MACH NUMBER
BOUNDRY LAYER DIVERTER DRAG COEFFICIENT VERSUS MACH NUMBER
WAVE DRAG COEFFICIENT VERSUS MACH NUMBER
TOTAL DRAG COEFFICIENT VERSUS MACH NUMBER
STORE D/q VERSUS MACH NUMBER
FRICTION DRAG COEFFICIENT AT A REFERENCE MACH NUMBER AND ALTITUDE

57

DATA (3,9)

From (process): PROPULSION
To (process): SYNTHESIS

BASELINE THRUST PER ENGINE
BASELINE FUEL FLOW PER ENGINE
ENGINE SCALING RELATIONS, IF UNIQUE
BASELINE RAM DRAG + INLET DRAG PER ENGINE

58

DATA (6,9)

From (process): MASS PROP
To (process): SYNTH

BASELINE GROUP WEIGHTS
WEIGHT SCALING MATRIX / ALGORITHMS

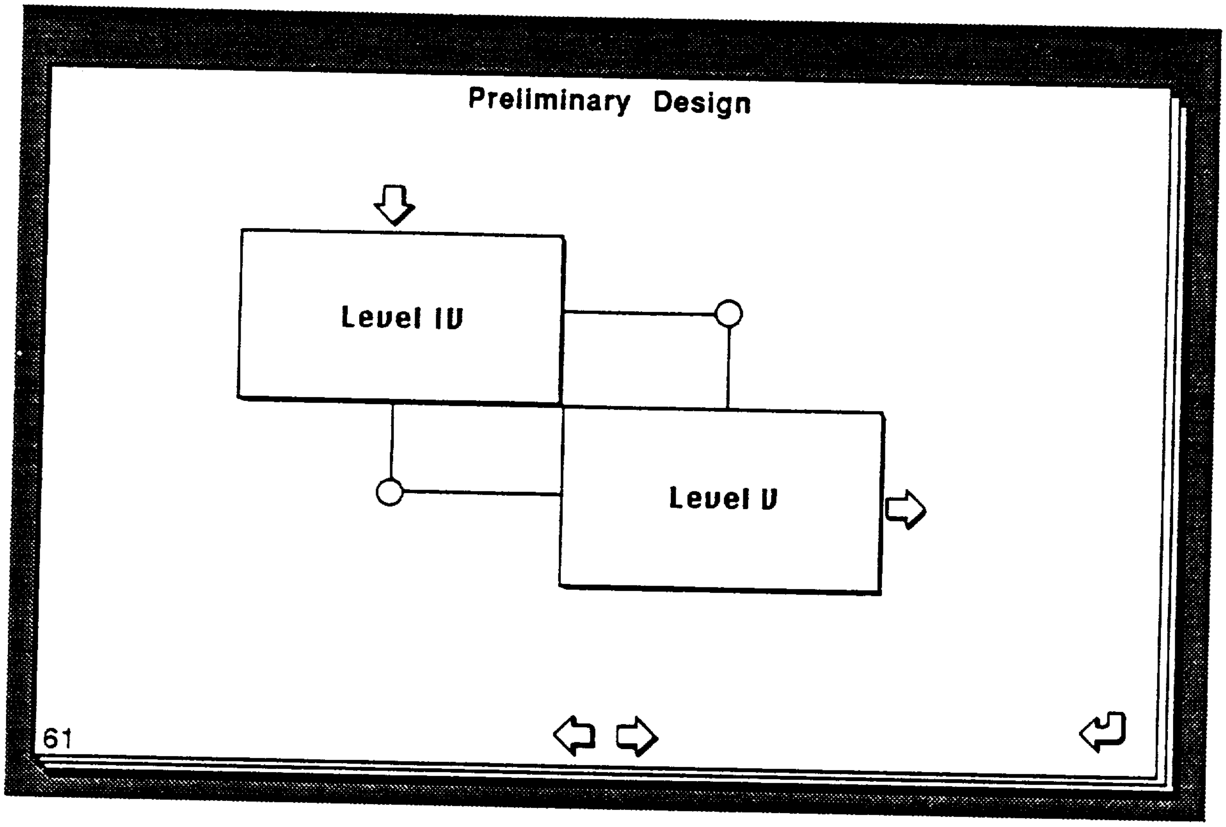
59

DATA (7,9)

From (process): PERFORMANCE
To (process): SYNTHESIS

FUEL WEIGHT REQUIRED TO FLY MISSION
RANGE / RADIUS ACHIEVED
SPECIFIC EXCESS POWER
MAX G'S
TAKEOFF AND LANDING DISTANCE
GROSS WEIGHT

60


Preliminary Design
Level IV
Level V
61

APPENDIX C

OPIMIZATION PROBLEM SUMMARY

This appendix summarizes the selected optimization approach, using the simplified mass properties model. No local sensitivity derivatives have been shown for the performance model because the selected approach will have the performance model send a new value of the objective function to the optimizer in response to a variation in one of the design variables. Also shown in this appendix is a graphical depiction of the wing design variables.

Table C -I. OPTIMIZATION PROBLEM SUMMARY

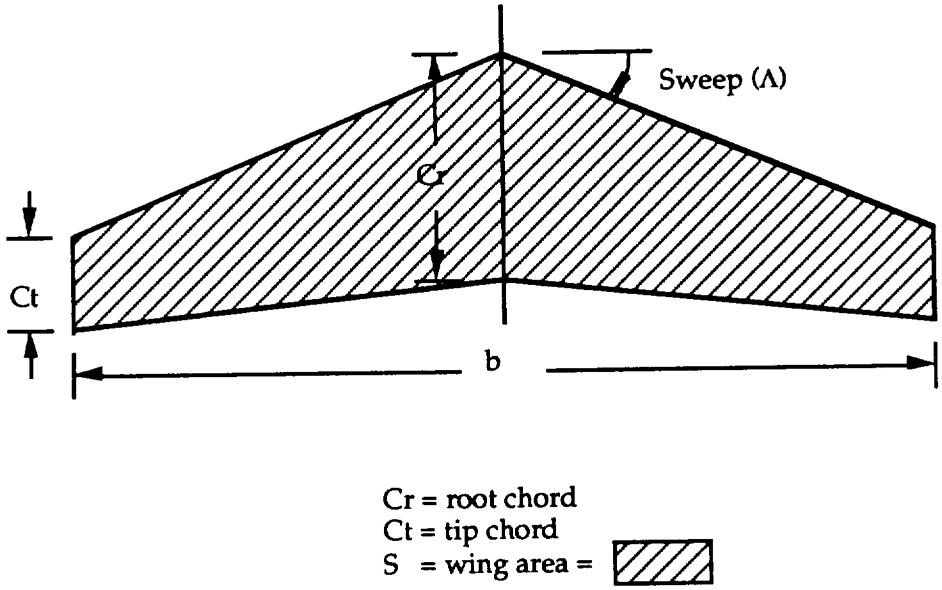
OBJECTIVE FUNCTION		
Minimize take-off gross weight $F(\psi)$ for a fixed "design" mission		
DESIGN VARIABLES		
area (S),	aspect ratio (AR),	camber.
sweep (Λ),	taper ratio (λ),	
thickness ratio (t/c),	twist,	
CONSTRAINTS		
Alternate mission radius $\geq$ required radius		G1(y)
Take-off distance $\leq$ required distance		G2(y)
Landing distance $\leq$ required distance		G3(y)
Max sustained load factor $\geq$ required load factor		G4(y)
Specific excess power (SEP) $\geq$ required SEP		G5(y)
SEP at specified load factor ≥ 0		G6(y)

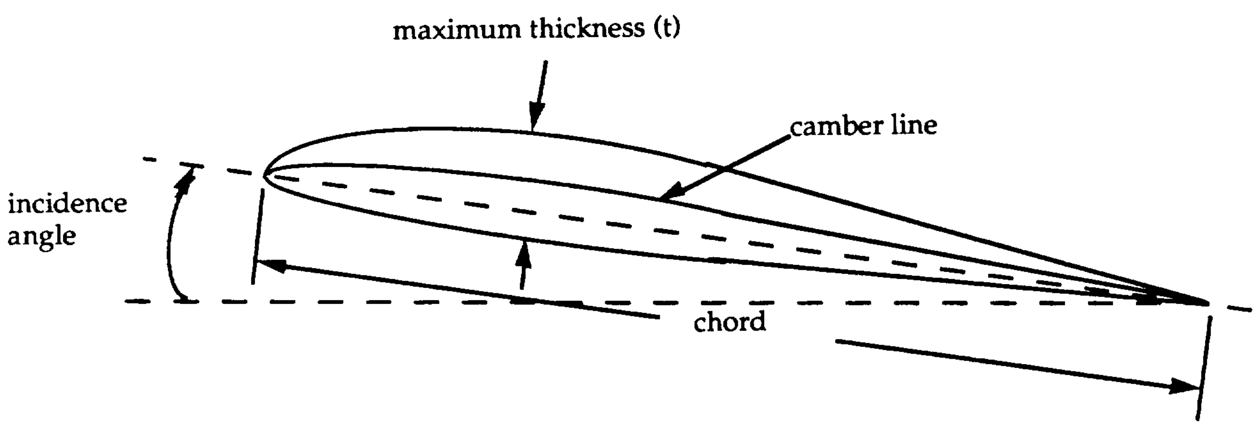
LOCAL SENSITIVITY DERIVATIVES

Mass Properties	Aerodynamics
$\frac{\partial WW}{\partial \psi}$, $\frac{\partial WW}{\partial LOAD}$,	$\frac{\partial CDWAVE(WING)}{\partial \psi^*}$, $\frac{\partial MDD}{\partial \psi^*}$,
$\frac{\partial WW}{\partial X_{cp}}$, $\frac{\partial WW}{\partial Y_{cp}}$	$\frac{\partial (dMDD/dCL)}{\partial \psi^*}$, $\frac{\partial K}{\partial \psi^*}$,
$\frac{\partial WW}{\partial NEW\ MATERIAL}$,	$\frac{\partial CLK}{\partial \psi^*}$, $\frac{\partial CDK}{\partial \psi^*}$, $\frac{\partial CDSF}{\partial \psi^*}$
	$\frac{\partial CL0}{\partial \psi^*}$, $\frac{\partial CL\alpha}{\partial \psi^*}$, $\frac{\partial CLMAX}{\partial \psi^*}$
	$\frac{\partial \alpha CLMAX}{\partial \psi^*}$, $\frac{\partial Xcp}{\partial \psi^*}$, $\frac{\partial Ycp}{\partial \psi^*}$
Total of 11 sensitivity derivatives	Total of 78 sensitivity derivatives

Legend: ψ = design variables

ψ^* = design variables except wing area





WING DESIGN VARIABLES

Wing Area (S)

Sweep (Λ)

Aspect Ratio (AR) = b^2/S

Taper Ratio (λ) = Ct/Cr

Thickness Ratio (t/c) = maximum thickness/chord

Twist = root incidence - tip incidence

Camber = max height of camber line/chord

Figure C - 1. Wing Planform and Airfoil Design Parameters

APPENDIX D

REFERENCES

1. Sobieszczanski-Sobieski, J.; Barthelemy, J. F.; and Giles, G.L., "Aerospace Engineering Design by Systematic Decomposition and Multilevel Optimization", ICAS-84-4.7.3, September 1984.

2. Sobieszczanski-Sobieski, J.; and Barthelemy, J. F., "Improving Engineering System Design by Formal Decomposition, Sensitivity Analysis, and Optimization", NASA TM-86377, February 1985.

3. Sobieszczanski-Sobieski, J.; James, B.; and Dovi, A., "Structural Optimization by Multilevel Decomposition", AIAA Paper 83-0832-CP, AIAA/ASME/ASCE/AHS 24th Structures, Structural Dynamics and Materials Conference, Lake Tahoe, Nevada, May 2-4, 1983.

4. McCullers, L.A., "Aircraft Configuration Optimization Including Optimized Flight Profiles", Presented at NASA Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, Langley Research Center, Hampton, Virginia, April 24-26, 1984.

5. Giles, G.L.; and Wrenn, G.A., "Multidisciplinary Optimization Applied to a Transport Aircraft", Presented at NASA Symposium on Recent Experiences in Multidisciplinary Analysis and Optimization, Langley Research Center, Hampton, Virginia, April 24-26, 1984.

6. Owens, H. G., "Military Aircraft Conceptual Design Trade Studies", NA-85-1543, Rockwell International, North American Aircraft Operations, 14 October 1985.

7. Owens, H. G., "Integrated Aircraft Design (IAID) Development Plan", NA-76-595-1, Rockwell International, Los Angeles Aircraft Division, 30 November 1976.

8. Axtell, H. W. ; and Mairs, R. Y., "Aircraft Optimization with the Integrated Design and Analysis System (IDAS)", TFD-85-1453, Rockwell International, North American Aircraft Operations, 21 May 1985.

9. Rogan, J. E.; and Kolb, M. A., "Application of Decomposition Techniques to the Preliminary Design of a Transport Aircraft", NASA CR-178239, February 1987.

10. Sobieszczanski-Sobieski, J., "Structural Optimization: Challenges and Opportunities", International Journal of Vehicle Design, vol. 7, nos 3/4, 1986.

11. Nicolai, L. M., Fundamentals of Aircraft Design; METS, Inc., San Jose, CA, 1975.

12. Catalano, R. F.; and Fabis, J. M., "Integrated Design Analysis System Test Report", AFWAL-TR-85-3068, Rockwell International, North American Aircraft Operations, Los Angeles, CA, Sept 1985. (permission has been obtained to use this information)

13. Taguchi, Genichi and Fhadke, Mahdev S. "Quality Engineering Through Design Optimization", IEEE GLOBECOM Conference Record, Atlanta, GA, 1984, 1106-1113.

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16. Abstract

A study has been conducted to investigate the feasibility of applying multi-level functional decomposition and optimization techniques to conceptual design of advanced fighter aircraft. Applying the functional decomposition techniques to the conceptual design phase appears to be feasible. The initial implementation of the modified design process will optimize wing design variables. A hybrid approach; combining functional decomposition techniques for generation of aerodynamic and mass properties linear sensitivity derivatives with existing techniques for sizing mission performance and optimization; is proposed.

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