

AD-783 307

AIRCRAFT DESIGN INTEGRATION AND
OPTIMIZATION. VOLUME 1

Advisory Group for Aerospace Research
and Development
Paris, France

June 1974

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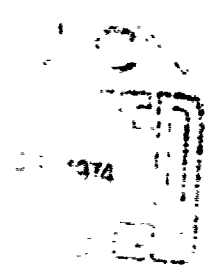
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NORTH ATLANTIC TREATY ORGANIZATION
ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT
(ORGANISATION DU TRAITE DE L'ATLANTIQUE NORD)

AGARD Conference Proceedings No.147
AIRCRAFT DESIGN INTEGRATION AND OPTIMIZATION
Volume i



NATIONAL TECHNICAL
INFORMATION SERVICE

Unclassified papers presented at the 43rd Meeting of the Flight Mechanics Panel of AGARD
held at Scuola di Guerra Aerea, Florence, Italy on 1-4 October 1973.

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- Exchanging of scientific and technical information;
- Continuously stimulating advances in the aerospace sciences relevant to strengthening the common defence posture;
- Improving the co-operation among member nations in aerospace research and development;
- Providing scientific and technical advice and assistance to the North Atlantic Military Committee in the field of aerospace research and development;
- Rendering scientific and technical assistance, as requested, to other NATO bodies and to member nations in connection with research and development problems in the aerospace field;
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Published June 1974

629.73.02:657.474



Printed by Technical Editing and Reproduction Ltd
Harford House, 7-9 Charlotte St, London. W1P 1HD

PREFACE

The Flight Mechanics Panel Symposium on "Aircraft Design Integration and Optimization" was held to share experience and knowledge in aircraft design which will help improve the quality and reduce the cost of military aircraft. The ever-increasing cost of developing and operating modern systems necessary to preserve the security of the NATO nations makes it essential to uncover means of reducing systems costs and, thus, help assure that NATO nations will have the best possible military aircraft that provide the required capabilities within the resources available.

The Symposium emphasis and theme -- impact of the preliminary design process on the development, production, and operational cost of aircraft systems -- were woven into all sessions of the Symposium through discussions of technical approaches, methods and experiences relating to cost reductions, means of improving aircraft quality, and orderly introduction of new technology. Case histories of actual aircraft preliminary design and hardware developments were included to provide retrospective insight into the role and adequacy of the design process. Authors were requested to consider the implications of cost and the extent to which cost was used as a design parameter or as a special objective. A technical approach, with discussion of actual design experiences, analysis methods, and concepts, was emphasized in order to avoid excessive generalization in discussing cost and design process questions and to provide engineering data of value along with insight into the role and impact of the preliminary design process.

The Symposium agenda provided for an introductory paper on the role of the preliminary design process, followed by twenty-seven technical papers, and a panel discussion of key issues as the final session. The sessions were as follows:

SESSION I -- The Preliminary Design Process and Its Impact on Cost -- included papers on designing for reduced development, production, and operational costs through "design-to-cost" and other techniques; prototyping; and design methods and techniques to improve effectiveness of the preliminary design process and ensuing aircraft quality.

SESSION II -- Design Integration -- included papers summarizing basic approaches, methods and trades in balancing requirements, capabilities, and cost; the impact of design interactions and techniques for coping with them; and case histories of the design evolution of recent aircraft to show means used to reduce cost and improve quality.

SESSION III -- Analysis, Optimization and Validation Testing Techniques -- included papers on engineering techniques and methods to improve quality and reduce cost by reducing uncertainties and optimizing tradeoffs and engineering designs.

SESSION IV -- Integration of Subsystems and New Technologies -- included papers to point out lessons learned in subsystem integration and the application of new technology; the impact of design criteria; new technical advancements and means for assessing their value and estimating costs involved in their application.

SESSION V -- Panel Discussion on Best Approach to Reduced Aircraft Costs -- planned to discuss either preplanned issues relating to the impact of preliminary design on aircraft life cycle cost or further pursue issues of much interest which arose during the course of the Symposium. Vigorous discussion among Panel members, with further detailed questions and comments from the floor ensued on the following two questions generated as a result of the Symposium:

1. We design civil aircraft to clearly defined operating cost requirements. In military weapon systems, does the total operational concept require much more detailed definition to the designer to allow system cost to be used as a proper design parameter?
2. Does "multi-role", or at least commonality at major subsystem level, offer major prospects for cost saving with the wide range of future defense systems required?

The Symposium provided many examples of the impact of the preliminary design process on total system cost. Numerous approaches, methods, procedures, technical findings, and case history experiences provided specific examples of how the process can be used to improve quality, apply new technologies, and reduce cost of developing, producing, and operating aircraft systems.

The design process is highly iterative and requires numerous tradeoffs and compromises to synthesize and evolve an integrated design which provides the required military capabilities within time and cost constraints. The numerous decisions throughout this process are based on the designers' ingenuity, innovativeness, and judgment, coupled with the best data available from engineering analyses and tests.

The design approaches, methods, techniques, and specific procedures summarized in the papers point out numerous ways of handling specific problems, and should spark new ideas for improving quality and reducing costs.

While discussions with the authors would be desirable to appreciate the subtleties involved and better understand what the designer does differently under design-to-cost methods, or how he uses the preliminary design process to further reduce cost, an appreciable insight will be gained by review of the papers presented in this Conference Proceedings. Representative things the designer can do to reduce cost include the following:

1. think cost reduction and consider cost as a more dominant parameter in the numerous tradeoffs that are made which affect performance, time, and cost;
2. fully understand the basic mission objectives in order to evaluate, prioritize, and properly balance specific mission and design requirements with full consideration of total force budgets and both current and new technologies;
3. consider requirements to be negotiable, and continually examine the trades between achieving design requirements versus cost involved;
4. consider the complete system problem, including the operational environment and cost implications in deciding design requirements, subsystems to be used, and means of reducing operational costs;
5. design for adaptability to a wide range of missions where feasible without compromising the initial program;
6. provide every required function in the simplest possible manner, and strive for simplification of design and reduction in numbers of parts;
7. use available, fully qualified components to the extent possible,
8. utilize the prototype concept, especially where innovative technologies are involved;
9. use a new technology where it is essential to meet the performance goal or reduce cost, but do not excessively increase risks by using combinations of innovative technologies at the same time, except perhaps in prototype programs;
10. do a more adequate job in the preliminary design and test phases to reduce problems in later detail design, tooling, and manufacturing phases;
11. establish meaningful, usable, and validated design criteria early in the program to avoid concurrency with the system development;
12. assure availability of design methods and data fully capable of coping with the design problem in order to speed the design and development work and reduce engineering costs and design risks; and
13. automate retrieval and data handling aspects of the design process to permit rapid analysis and optimization.

It is clear from the excellent papers presented and active participation of the audience that the preliminary design process is a topic of much interest and worthy of continued attention. The lively discussions following presentation of the papers and the Panel session give hope that increased attention will be given to the preliminary design and development process as a means of reducing cost of future aircraft systems.

The formal papers presented at the Symposium are contained in this Conference Proceedings. Volume I contains the unclassified papers and Volume II contains the classified papers.

Appreciation is due to the well organized response of Professor Hafer, of FRG, who helped assure we would have sufficient papers for the meeting at a time when it appeared that competitive pressures might preclude some of the kinds of discussions necessary for an effective meeting. Diligent efforts by R.F. Creasey, of the UK, and J. Forestier, of France further assured a good spectrum of papers, and the final program closely approximated that planned.

The thorough and well worked out arrangements by R.J. Wasicko, FMP Executive, the excellent accommodations of the Air War College of the Italian Air Force, with the superb efforts of General Lucertini, General Monti, and General Scafafia, plus their able staff, and the gracious hospitality of Avv Luciano Bausi, Mayor of the City of Florence, did much to assure open and lasting discussions.



WILLIAM E. LAMAR
Member
Flight Mechanics Panel

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THE ROLE OF PRELIMINARY DESIGN IN REDUCING DEVELOPMENT,
PRODUCTION AND OPERATIONAL COSTS OF AIRCRAFT SYSTEMS

by

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INTRODUCTION

Good morning, fellow delegates. I would like to add my welcome to that of our host. It is certainly a pleasure being here in Florence and having the opportunity to introduce this meeting on "Aircraft Design Integration and Optimization" to such an eminent group of aircraft designers, engineers, managers and operators.

Selection of Florence for the meeting was most fitting, for the revival of art and design started here during the Renaissance exerted much influence on the world. It is hoped that this meeting will result in greater use of the preliminary design process to spark a new renaissance in improving aircraft quality and reducing cost.

My purpose in this introduction is to discuss the meeting objectives, reasons why we decided to have it, the preliminary design process, and issues related to the impact of the process on cost reduction, design quality, and application of advanced technologies.

MEETING OBJECTIVES

The basic objective of our meeting is to share experiences and knowledge in aircraft design, methods, and technologies that will help reduce aircraft system cost. We need to do this to make our resources go further and assure that NATO nations have the best possible military aircraft within the resources available. Emphasis in this meeting is on the impact of the preliminary design process on the development, production, life cycle cost, and on the orderly introduction of new technologies which can reduce mission accomplishment costs.

We would like to probe a number of questions at this meeting. What can aircraft development designers and managers do to arrest the ever-increasing cost of aircraft weapon systems? What do they do different under "design-to-cost," "prototyping" and other new management methodologies design to accomplish these objectives, compared to what they did before? What is the role of the preliminary design process in improving the quality and reducing the cost of new aircraft? How can we further exploit the potential of the design process in accomplishing these objectives?

I hope the technical sessions, plus the wrap-up Panel discussion, during the four-day meeting will provide a sharper insight into the resolution of such questions as well as much useful technical information.

MEETING RATIONALE

There are many reasons for having this meeting. Much concern has been expressed by high-level defense officials about the ever-increasing sophistication and cost of new aircraft systems. High costs have already forced sizable reductions in the affordable numbers of modern aircraft. Despite improvements in the combat capability of individual aircraft, serious questions have been raised whether total force capability will remain adequate if the trend continues. Assuring both the quality and numbers for adequate defense poses a continual challenge. The number of new aircraft starts has also been significantly reduced, thus reducing options and the experience level of design teams. Some projections even have indicated that if the current trend of exponentially increasing costs continues, in forty years the entire Air Force budget will be required for a single aircraft system!

What can be done about this alarming situation? A number of effective actions have been taken over the past several years in the United States. Defense weapon system acquisition policies have been revised and new methodologies and strategies for development, such as "design-to-cost" and "prototyping", have been implemented.

While emphasis on cost reductions and the use of prototypes to reduce development risk are not new concepts, the current methodologies have unique features. Further, their implementation has been accompanied by a vigorous campaign to assure that aircraft system developers are fully aware of the concepts, objectives, specific details, and the urgency of achieving meaningful reductions in cost while producing a system that provides required capabilities.

However, little has been said about the tremendous impact of the preliminary design process on the cost of developing, producing, and operating aircraft systems, or its value in defining needed technology and assessing the payoff of emerging new technologies in improving quality and reducing cost of future systems. Some excellent broad guidelines for reducing costs have been offered in articles by high level officials and industry leaders, and several interesting studies and case histories of unique design team operations offer insight for improved development. However, much more needs to be learned about just how the design process can be used to make both significant improvements in quality and reductions in cost. Former United States Deputy Secretary of Defense, David Packard, in a July 1972 article on prototyping and the basic problems facing the aircraft industry, reiterated his strong belief that the new policies and improvements in method will be effective only to the extent they are accepted and implemented by

people throughout the Defense Department and throughout industry." Those "responsible for making decisions . . . on specific programs and specific projects, are the ones -- in fact, the only ones -- who can bring about the improvements we must have." This points out the need to improve communications; and since we are talking about aircraft systems, we need to fully involve aircraft designers, engineers, cost experts and operators that have to do the job. Effective communication to share ideas and knowledge is, of course, a basic reason for AGARD.

The AGARD Flight Mechanics Panel has long been concerned with aircraft design and technology integration. The increasing concerns about aircraft development problems and rising costs are shared by all NATO nations. These concerns, coupled with a strong belief that the preliminary design process offered a powerful, but inadequately exploited means of improving quality and reducing aircraft acquisition and operational costs, led to a proposal in October 1971 that the Flight Mechanics Panel sponsor a meeting to point out the great potential of the design process and probe means of better using it to help solve basic aircraft quality and cost problems. Panel approval for the meeting followed the author's presentation of a "pilot paper" to the Panel in Braunschweig, May 1972. Subsequent planning by the program committee and preparation of papers by an excellent group of twenty-seven authors culminated in this meeting.

PRELIMINARY DESIGN PROCESS

I am using the term preliminary design in the broad sense to include the entire preliminary design and development process up to the point of configuration "freeze" and a decision to proceed with detailed design and preparation of hardware fabrication drawings. Many different terms are used in the industry to label various portions of this process. As used here, the preliminary design and development process includes such phases as conceptual design, advanced design, and configuration or product definition.

The preliminary design/development process includes iterative paper and computer analyses, studies and design layouts to optimize trades between requirements, available technology, timing, and cost. The generic development steps of analyses, preliminary design, detailed design, fabrication, test, and establishment of criteria actually go through many iterative cycles in the development of aircraft, and exist as "inner loops" during the aircraft preliminary design process. For example, during the preliminary design of an aircraft, it is necessary to obtain data from wind tunnel model tests. These tests involve analyses, the preliminary design and detailed design of a wind tunnel model, its fabrication, test, and evaluation of results. The resulting criteria and data are then fed into the next iterative cycle of the aircraft design and the process is repeated as necessary. As the process proceeds, the configuration is eventually defined and validated in sufficient depth to proceed into the design fabrication and test phase for a full scale aircraft. Here again, however, the first full scale aircraft could be a prototype and would thereby require an additional cycle for development of the actual production airplane. The preliminary design phase thus involves wind tunnel tests, subsystem and other developmental tests, and both computer and manned simulations in the process of defining the configuration and resolving uncertainties. Some very detailed design work may also be included to resolve questions and assure adequacies in specific areas.

While accuracy of analysis methods has improved steadily over past years, results of pure "paper analyses," sometimes used for decision-making purposes, must be considered "suspect" unless adequate assurances exist that the assumptions made in preparing the analytical model are fully valid for the new design being analyzed. Since the new design usually includes configuration, flight environment, or operational usage improvements in order to warrant its development, it is likely that the analytical models which were developed and validated from previous design efforts will introduce some errors when applied to the new design. Thus, in contrast to "paper studies," the developmental tests involved in the preliminary design process provide a markedly different degree of confidence and progress toward a real airplane.

In the preliminary design process, it is common practice to establish baseline designs which are tested to validate the analyses used in their formulation. After the baseline design is "anchored" by such tests, reasonable extrapolation can be made for parametric tradeoffs. Careful judgment is required because, if the extrapolation is too great, the errors that creep in may become significant, especially in areas of high sensitivity. In such cases, a new baseline design and accompanying tests are required to anchor a new design point. This iterative process involves the structure, flight control, propulsion, and subsystems of the aircraft as well as to the aerodynamics, and is a critical one in the preliminary design process.

Figure 1, taken from a paper by W. C. Swan, a presenter at this meeting, provides additional insight into some of the preliminary design tasks and their interrelationships. Other perspectives will be provided at the meeting. There are obviously many ways of depicting the nature of the process. A thorough discussion could be the subject of an entire report.

EVOLVING RELATIONSHIPS

The increasing complexity and sophistication of the design process to cope with the need of initial acquisition and, later, life cycle costs are indicated by the next chart (Figure 2) recently prepared by Fred Grazio. This shows that during the 1930-1950 time period, emphasis in design was primarily on the airframe, propulsion and payload. Determination of the resulting performance was considered to be the key portion of the design process. In the 1950-1960 time period, the growing need to consider the entire system problem led to the weapon system concept. This was accompanied by greater consideration of the avionics and research, development, test, and evaluation (RDTE) cost in the design process, and much effort was expended to determine performance and effectiveness. In the past decade, system integration was stressed to an even greater degree and the design of compatible ground support equipment was added to the design process. In addition, substantial efforts during the concept analysis phases to determine the performance and cost effectiveness of the resulting system generated extensive paper workloads and attendant cost increases in the design cost. The current emphasis on totally integrated systems requires consideration of total life cycle cost in the aircraft design.

The preliminary design development phase is shown on Figure 3 in comparison with various steps used in the total aircraft development process as specified by system acquisition procedures, both during the late 1960's and early 1970's, and now. Since terminology and definition of phases not only vary with time, but also with governments and organizations, it is important to carefully understand which particular relationships are being discussed if there is to be a clear communication.

IMPORTANCE OF THE PRELIMINARY DESIGN PROCESS

The leverage exerted by the preliminary design phase on the total system cost is indicated by Figure 4, which shows the total life cycle cost for a major weapon system. The preliminary design process in this case included synthesis of a large number of potential configurations and substantial wind tunnel tests, as well as design of the selected configuration. Yet, it constituted only a very small portion of the total life cycle cost of the system -- less than one half of one percent (0.5%). Since this system is still operational, the percentage cost of the design portion is going down even further each year!

We find that typically only one tenth to five tenths of one percent (0.1% to 0.5%) of total life cycle cost is being devoted to the preliminary design phase. Is this enough? I am sure that many designers would think it is not. The problem is to convince the top managers that greater emphasis in this phase will yield rich dividends later, even if the future money savings are fully discounted for interest cost.

Aircraft preliminary design is akin to the architecture of a building. It is the phase in which the future characteristics and capabilities are largely determined and wherein most of the key decisions are made. Thoroughness of the work during this phase exerts a critical impact on the final aircraft system capability and total cost. Because of this and the much higher cost of subsequent phases, as indicated by Figure 4, it should be clear that a major portion of the development, production and operational costs are essentially predetermined at the time when only a small portion of the total life cycle cost has been expended.

The increasingly large effort following the preliminary design obviously involves an increasing investment in financial and technical resources. In addition, the time remaining to complete the development and meet the contractual requirement becomes less and less. The net result of these two pressures -- quickly increasing investment and reduction in time available to complete the job -- is an increasingly strong resistance to design changes. Before long, no basic changes are permitted unless of a critical nature. This is often true even though changes are logically needed to assure adequate confidence in meeting basic required capabilities. While "work around" methods are employed, and some detailed changes and product improvements may have to be made with a large cost penalty, the basic characteristics provided for during the preliminary design often "lock in" or "bound" the system's capabilities throughout its life.

Many critical program decisions are made on the basis of the preliminary design process results -- after either the conceptual or definition phase. These include:

1. selection of the winning contractor, an important factor in determining the quality of the final product;
2. the basic system characteristics and the degree to which they have been optimized and uncertainties eliminated;
3. the level of technology to be used, which greatly affects final aircraft quality and cost; and
4. decision to proceed into expensive full scale development and perhaps production.

All of the above will largely determine the eventual operational capabilities and flexibility of the aircraft, as well as its development, production, and usage costs.

The preliminary design process, when applied to potential future aircraft, offers a means of identifying and quantifying gaps in technical knowledge, methods, processes, and facilities needed for the future design, as well as any technical barriers which might impede a successful development. The means of doing this are straightforward. Technical gaps, barriers, uncertainties, and sensitivities can be determined by preparing and analyzing baseline designs and trade studies with respect to the technologies associated with them. Special attention should be given to areas where both technical uncertainties and high sensitivities exist, since they could result in a large impact to system capabilities and cost.

Further, the preliminary design process offers a powerful means of assessing the payoff of technical innovations and emerging technologies on system capability. Emphasis can be given to development and application of technologies which can reduce cost. Baseline designs and associated analyses, with and without the new technology, provide a clear means of obtaining good comparisons and determining the impact of the new technology on the design characteristics, capabilities, cost, and timing.

Use of the preliminary design process thus offers much potential in determining needed new technologies and assessing the payoff possible from application of emerging technologies. The process can be applied to examine possible modifications to existing systems, as well as potential new aircraft systems. Since the entire system design is considered, this methodology should provide an effective means of avoiding suboptimization in the selection of technical programs which should be given emphasis.

Some of the uses of the preliminary design process which have been discussed above are summarized in Figure 5.

QUESTIONS AND ISSUES

While it is clear that the design process exerts a large influence on all subsequent phases of the aircraft life, and offers much potential in improving quality and thereby reducing costs, a number of questions and issues appear to exist. Some of these are listed below.

1. How can we really do a more effective job in the preliminary design process to reduce life cycle costs in the face of all the practical constraints of resources, time schedules, competition pressures, and requirement changes that actually exist? Will increased resources applied to preliminary design significantly reduce total program costs?
2. How can the preliminary design process be more effectively interacted with the requirements evolution process and thereby eliminate costly unnecessary requirements?
3. Can factors affecting production and operational costs, such as reliability, maintenance and field operations be meaningfully traded and quantitatively assessed in preliminary design, or is it like trying to measure a doughball with a micrometer?
4. Are improvements in preliminary design approaches and methods keeping pace with needs imposed by modern sophisticated aircraft and design-to-cost requirements? Can we reduce the cost of conducting the design process itself?
5. Will extensive use of computerized analyses, optimization programs, and computer graphics (man-in-the-loop) design methods actually reduce cost and improve effectivity of both the design process and resulting aircraft, and how much is practical? Will they stifle innovation and creativity?
6. What depth in the preliminary design process is required to assess the payoffs, risks, and costs involved in use of innovative new technology with sufficient credibility to be acceptable in design-to-cost proposals? What depth of analysis, design, and demonstration of the new technology is required?

COST REDUCTIONS

Consideration of cost is hardly new in aircraft design. A good designer always considers cost directly or indirectly when he examines the knee of a curve to determine the point of diminishing returns when increasing the value of a desired parameter. Value engineering programs have been used for years to reduce cost, especially at the detail design level. However, the continual cost escalations over recent years clearly point out the necessity for major campaigns to reduce cost. There have been many of these in the past, as indicated in Figure 6. This indicates that managers as well as designers have been most active about seeking ways to do something about the problem. Systematic review of these would uncover a number of useful lessons and is recommended for those interested. Without question, we need to do better, and since it is really a tough problem, we will probably continue to need new cost reduction campaigns to keep attention on the problems, and to provide improved policies and methodologies compatible with the changing social, political, economic and military environment.

In addition to the design-cost questions already noted in the previous section, several others are worthy of discussion.

1. The prototype concept clearly provides a means for working out technical problems and proving feasibility and is especially valuable when the design pushes the state of the art or incorporates innovative technologies. On the other hand, a prototype also increases the total time and perhaps cost on paper (i.e., a "plan for success" type program frequently assumes no major problems, and therefore, would normally show a shorter time and less cost by going directly into production aircraft, rather than going through the prototype step). The issue is whether the prototype step will reduce the actual total cost of producing satisfactory operational aircraft. If the cost saved by avoiding high cost changes in production or operational aircraft is greater than the prototype cost, it obviously is a good buy. Unfortunately, an evaluation of this is always subjective. Prototypes offer other advantages in that competition and options can be maintained for a longer period of time. However, the prototype concept can result in large gaps in the design, engineering, and fabrication activities, thus decreasing continuity of the work effort and increasing total costs, unless the program allows some concurrency in initiating development of additional test or production aircraft. This has been worked out in a number of programs by timely initiation of additional preproduction aircraft for engineering, development, service test, or operational test. More widespread use of the prototype concept would permit more new aircraft starts, provide more viable options for decision makers and do much to preserve competent and experienced design teams, a key factor in program success.
2. It is often said that greater use of standardized components and subsystems would greatly reduce life cycle cost. Much evidence indicates that if the proper standardized components and subsystems existed, this might well be true. A number of questions exist here. How do we select items to be standardized and avoid technology and operational obsolescence? Would such a concept result in large numbers of equipments wasted on the shelf because of lack of users? Can commonality at the aircraft systems level and multi-mission capability provide a significant gain?
3. Another important issue relates to the additional money that should be spent during the design and development phase in order to reduce operational cost. Examples exist where the fuel cost savings alone would more than pay for aircraft modification costs within one year. It is

also true that discounting of future cost benefits, when used to compare options on the basis of present value, favors systems with low acquisition costs. How much reliability should be provided? What is the tradeoff between high reliability, which will reduce equipment removals, maintenance, repairs, and replacement cost in the future versus the current cost of developing the component with assured high reliability? Reliability should be designed into the system by careful consideration and use of the best known techniques for attaining reliability during the design process. Relatively little increase in cost is involved in doing this where clear guidelines and experienced designers are involved. However, assured reliability requires much testing and directly increases cost. Concepts such as "life cycle cost" (LCC) emphasize the importance of balancing the design to properly consider operational costs. The fundamental problem appears to be the requirement to spend large dollars now in order to acquire a potential future savings. Use of the preliminary design process to assess the probability of achieving the future savings might be useful in permitting a more balanced judgment.

DESIGN QUALITY

Methods of analysis, optimization, and design integration, and means of reducing uncertainties and risks are of key importance in assuring quality, providing a credible design, and preventing future cost increases to fix problems and omissions. Cost must be considered as an engineering parameter in the optimization process. Most methods involve some degree of uncertainty, depending on their application and degree of extrapolation from test validated data. New configurations often require the concurrent development of new methods, plus considerable test substantiation in order to achieve accurate results.

Reduction of tests to save money and failure to adequately eliminate uncertainties in the design stage can later result in multifold cost increases to fix problems or improve capabilities to meet minimum requirements. An old adage -- "A stitch in time saves nine." -- is very appropriate in this area. A key question regarding analysis and test validations is, "How much is enough?" One answer is the judgment of an experienced design team with a clear understanding of uncertainties and sensitivities.

"KIS" -- Keep It Simple! A good rule to keep in mind. Certainly, emphasis should be given to achieving the most simple system that can effectively accomplish the required military mission. Simplicity in design and hardware not only helps reduce initial development and acquisition costs, but often reduces operational costs as well. System readiness is enhanced and, in many cases, operational success is improved. However, operational simplicity may also demand sophisticated designs and hardware. Many examples exist. A sophisticated control augmentation system greatly simplifies the pilot's task of controlling an aircraft which has poor handling qualities. Complex sensors may be essential to identify targets at night or under all-weather conditions. A complex instrument landing system not only simplifies the adverse weather landing task, but also greatly contributes to safety. Such sophisticated systems can thus greatly improve mission success and reduce aircraft losses, and thereby reduce the overall cost of accomplishing a military task. Too often, emphasis on cost reduction implies that new sophisticated technology is pricing us out of business, without taking into account the value of the increased operational effectiveness provided by the new technology. Careful design and cost trades, in relation to the essential military requirements for both peacetime and wartime operations, are necessary to determine the degree of simplicity or sophistication. Clearly, all aspects of the system design, hardware, support, and operational use must be considered in defining the most effective system. Otherwise, suboptimization may well be the result.

ADVANCED TECHNOLOGY

A number of emerging technologies appear to offer payoff in improving effectiveness and reducing cost of new systems. Figure 7 lists many such technologies frequently discussed in detail at AGARD meetings. Despite the potential improvements which these technologies appear to offer, their application into production or even prototype aircraft often appears excessively slow to the technology developer. Even quantitative assessments of their payoff to specific aircraft appear to lag. Why? There are many reasons for the lack of application; however, I think they can be summed up by two words -- uncertainty and cost. Uncertainties exist as to the extent and credibility of the expected payoff in performance, operational capability, or cost reduction, possible interactions in the design, and best means of applying the new technology. Uncertainties exist as to the impact of including the new technology on the development, cost, and schedule. Application of new technology is also inhibited by the high cost of reducing the uncertainties. Innovative technologies require credible demonstrations to provide adequate confidence that they are ready for application. Since such demonstrations are quite expensive, limited development budgets demand a high degree of selectivity. The problem is complicated by lack of clear-cut criteria for acceptance. What constitutes an adequate demonstration? The answer seems to vary with each specific situation, depending on need, payoff, risk, and cost.

The preliminary design process offers a powerful tool in reaching a decision. Payoff can be determined by comparison of designs with and without the new technology. Risk can be assessed by appropriate subsystem, interaction, and failure analyses, taking into account sensitivities and uncertainties.

In some cases, as for example when aeroelastic interactions are critical, a valid assessment of risk can be quite expensive and involve detailed engineering analysis and test. Nevertheless, it is far less expensive to resolve such questions in the design phase than face the problem in a later production phase. Design is an iterative process and the application of assessment "gates" during both design, prototype development, and later production phases can be used to assure that the payoff, risk, and funds involved with application of the new technology are in harmony. Since analytical processes are not perfect, a gradual evolutionary approach to the application of new technology has sometimes been the only acceptable answer; as an example, use of graphite composite materials in small noncritical structural applications before application to primary load carrying structure.

While it is the belief of many that new technologies invariably increase complexity, sophistication, and cost of new systems, new technology can either directly or indirectly reduce the life cycle cost of

the system, as well as the cost of accomplishing a military function. Unfortunately, assessments of the operational and equivalent dollar value of improved capabilities provided by new technologies versus the cost involved are complicated by inadequacies in analysis, evaluation, and costing methods. Credibility of results, however, depends on the credibility of the inputs -- many of which depend on the preliminary design process and the tests associated with it.

The presentations and discussions in the technical sessions of the Symposium will provide insight into these and many other questions. I believe it is clear that the preliminary design and development process offers an effective tool in increasing design credibility and reducing costs. How much is enough varies widely with the particular design problem. The meeting papers will provide further insight into the questions relevant to improving quality and reducing cost, and suggest means of doing better.

PRELIMINARY DESIGN FLOW CHART

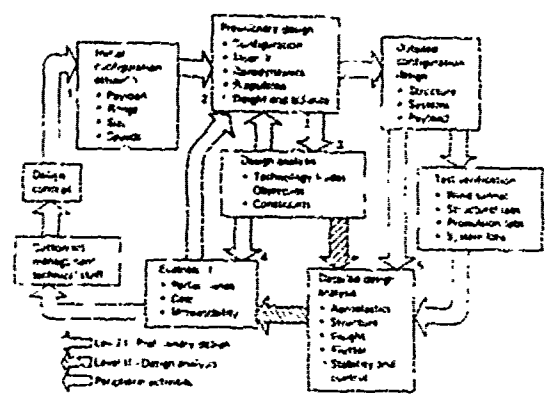


FIGURE 1

PRELIMINARY DESIGN SOPHISTICATION WITH YEARS

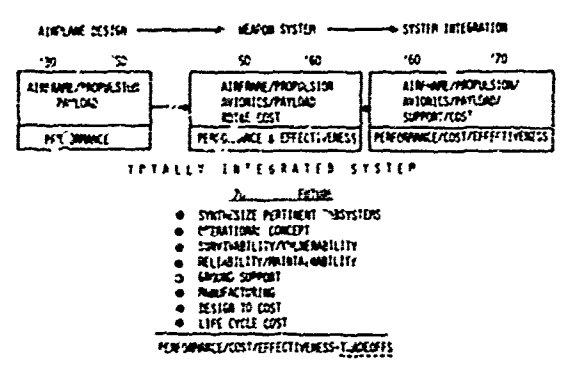


FIGURE 2

SYSTEM PROGRAM PHASES PRELIMINARY DESIGN DEVELOPMENT COMPARISON

PHASE	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
DESIGN	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
ANALYSIS	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
TESTING	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION

PRELIMINARY DESIGN DEVELOPMENT

PHASE	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
DESIGN	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
ANALYSIS	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION
TESTING	CONCEPTUAL DESIGN	PRELIMINARY DESIGN	Detailed Design	MANUFACTURE	OPERATION

FIGURE 3

PRELIMINARY DESIGN PROCESS USES

- IDENTIFY NEW RECENT AND FUTURE TRENDS & CAPABILITIES
- DEVELOP A PLAN OF WORK CAPABLE OF MEETING NEW NEEDS
- UNDERSTAND & RESOLVE INTERDEPENDENCIES
- IDENTIFY KEY DESIGNING FACTORS, CONSTRAINTS & SENSITIVITIES
- DETERMINE & ASSESS IMPACT OF INTERDEPENDENCIES
- DETERMINE TECHNICAL RISKS & BARRIERS & RATE OF NEW TECHNICAL STRATEGIC ADVANCEMENTS
- EXPLORE USE & ASSESS PAYOFF OF EXISTING TECHNOLOGIES
- FOCUSSES AND AIMS
 - BETTER USE OF RESOURCES
 - IMPROVED PERFORMANCE
 - EARLY IDENTIFICATION OF PROBLEMS
 - EARLY IDENTIFICATION OF TECHNICAL RISKS
 - ESTABLISH TEST CAPABILITIES
 - MEANS OF REDUCING LIFE CYCLE COSTS

FIGURE 5

SOME NEW AND PAST COST REDUCTION METHODOLOGIES

- HIGH YIELD COST
- DESIGN TO COST
- PRICE ENGINEERING
- SHARED COST
- FLY BY WIRE
- FLY BY WIRE II
- CONSOLIDATED
- ELITE
- E-FIELD PACKAGE
- FASTER INTEGRATION
- MULTISTAGE
- FLYING WINGS
- ADAPTED MANAGEMENT
- INTEGRATED DEVELOPMENT

FIGURE 6

LIFE CYCLE COST

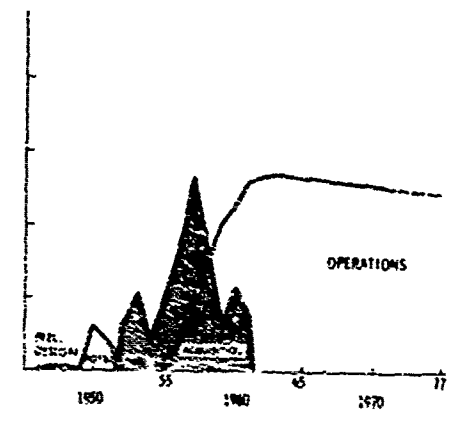


FIGURE 4

HIGH PAYOFF TECHNOLOGIES

- CONTROL CONFIGURED VEHICLES
- ADVANCED WING LIFT-DRAG REDUCTION/AERODYNAMICS
- THRUST VECTORS/REVERSAL
- APPLICATION OF COMPOSITES & ALUMINUM METALLIC CONCEPTS
- HIGH G COCEPT
- CONFERRAL CARRIAGE
- INDEPENDENT WEAPON AIRING
- INTEGRATED FLIGHT CONTROL/FIRE CONTROL
- MULTI MODE DIGITAL FLIGHT CONTROL
- REDUCED DETECTABILITY/VULNERABILITY
- SCALE/SPIN SOLUTIONS
- VARIABLE CYCLE ENGINES
- DIGITAL AVIONICS

FIGURE 7

PRELIMINARY DESIGN ASPECTS OF DESIGN-TO-COST
FOR THE YF-16 PROTOTYPE FIGHTER

by

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SUMMARY

The YF-16 prototype aircraft was conceived and is being developed as a low-cost, exceptionally high-maneuvering-performance fighter aircraft. To meet the cost/performance objectives, a number of advanced technology features, including vortex lift, variable wing camber, wing/body blending, relaxed static stability/fly-by-wire, and high-performance normal-shock inlet, were optimized and integrated during the preliminary design phase. The basic design concept was to apply these advanced technologies in a way, first, to produce a small-size aircraft and, second to achieve simplicity - both of these design objectives having a direct beneficial effect on the development, acquisition, and life-cycle cost. The resulting configuration is predicted to meet all program cost/performance objectives; it will be flight demonstrated starting in early 1974.

1. INTRODUCTION

There is increasing concern with the problem of escalating costs of modern weapon systems, including of course aircraft, which in the U.S. represents a large percentage of the Defense Department procurement budget. Without entering into all of the ramifications of this problem, it must be admitted that, in the past, cost has not been given the same consideration as performance or schedule, and that design decisions have been generally made in favor of performance. Although cost has not been ignored in the past, and cost-effectiveness studies and value engineering programs have been of benefit in making procurement and design decisions, they have not in themselves produced the desired results. Cost-effectiveness studies that were intended to provide a quantitative assessment of capability versus cost for alternative systems or design did not in many instances take into full account the development risks or true operating cost that were incurred, and these analysis tended toward justification of more exotic and complex designs. Value engineering programs have proved effective in reducing costs, but they have generally been concerned with cost reduction on a detail level and have not affected overall aircraft configurations.

In view of these continuing cost problems, there is increasing effort to interject cost as a parameter co-equal with performance in the design decision process and to set specific cost targets against which to design. When these targets are established at the time a preliminary design is started, they can have a profound effect on the procedures used and on the design concepts and the configuration selection process. There are at least three fundamental aspects of the preliminary design process that have a direct impact on program and air vehicle cost:

1. Design concept and configuration chosen.
2. Degree of risk involved in meeting a given set of requirements.
3. Adequacy and thoroughness of the preliminary design effort in establishing a firm configuration from which to proceed into subsequent phases of the program.

The first and second aspects are self evident. It is possible for a manufacturing operation to expensively produce an inherently low-cost design, but it is not possible for a manufacturing operation to inexpensively produce an inherently high-cost design. Historically, high-risk designs have also been responsible in a number of cases for high costs, particularly when development is coupled with concurrent production.

The third fundamental aspect is also generally recognized. As noted in the program theme, major accelerating expenditures of money occur after completion of the preliminary design, and a thorough and complete preliminary design is the key to permitting an orderly and economical transition into the detailed design, tooling, and manufacturing phases of an aircraft. Errors, miscalculations, or wrong engineering judgement during the preliminary design process can cause a project or program to fail or at best flounder,

with consequent inordinate high cost, schedule slippage, or both. The probable successful completion of an aircraft design project through the flight test phase is a direct function of the soundness of the engineering data available, the adequacy of the analysts of this data, and the judgement used in arriving at and making design configuration decisions based on this information during preliminary design. This is a readily recognized tenet, and there is a constant striving to add greater credibility to the design effort by bolstering the preliminary design data base through improved analytical procedures and methods, simulators and syntheses, and wind tunnel, laboratory, and flight tests.

The preliminary design of an aircraft can above all else be characterized as an iterative process in which a very large number of variables must be evaluated and considered. The evolving design concepts must be integrated, hopefully in a synergistic or complimentary way but more usually in a compromise. The process involves innovation and inventiveness. The preliminary design configuration that results from this process is a reflection of the relative weighing given the various requirements, the effectiveness of the integration process, the inventiveness of the designers, and the judgement applied to the compromises made. Walter Diehl in his book, "Engineering Aerodynamics", first published in 1928, makes the following statement in his opening paragraph:

The designer of an airplane is confronted with an endless series of compromises. At each stage in the design he must decide just how far a loss in one characteristic is justified by a gain in some other characteristic. The degree of success finally attained depends largely on the soundness of the judgement exercised in the designer's decisions.

We have come a long way since 1928 in our preliminary design methods and procedures; however, compromise and exercise of sound engineering judgement are still two very essential ingredients of the preliminary design process. The extensive efforts that have been put forth in developing more precise procedures and methods for design purposes (particularly computer analysis and synthesis, and improved wind tunnel and flight simulation techniques) provide a great deal of information not available even a relatively few years ago. Present computerized design programs make it possible to test and evaluate many more alternatives and options than heretofore. These improved methods and procedures have materially aided in developing aircraft configurations at the preliminary design stage so that, when rendered into hardware, risk is reduced and there is greater assurance of meeting the design objectives. Nevertheless, there are still some obvious problems in structuring a preliminary design today that prevents completely computerized preliminary designs. Some of the more significant of these are concerned with understanding all of the interactions involved in the integration process, and quantification of the variables, such as the factors involved in life-cycle cost. It is these areas, and there are no doubt others, that require application of judgement if a successful design is to be evolved.

There is an additional point that needs to be made that has a direct bearing on the cost of an aircraft and development program. That is the imposition of overly restrictive requirements that leave little room for compromises that may need to be made to achieve the primary performance objectives. It appears that this situation is universally understood but inconsistently practiced. It requires rigorous discipline on the part of the designer and customer to assure that priorities are properly placed and that the important performance goals or objectives are not eroded away by secondary considerations.

2. YF-16 LIGHTWEIGHT FIGHTER PROGRAM

To get more specifically to the theme of this session, "The Preliminary Design Process and Its Impact on Cost", the preliminary design approach and concepts and resulting configuration characteristics of the US Air Force/General Dynamics YF-16 Lightweight Fighter will be discussed, since this program has specific cost objectives attached to it.

The YF-16 Lightweight Fighter is a current program in which cost is a first-order consideration in the prototype design, tooling, manufacturing, and test. Also, design to achieve the lowest possible production unit cost is paramount. A brief description of the project and the aircraft is appropriate for an understanding of the subsequent discussion concerning the specific actions taken during preliminary design to meet these cost objectives.

The program has several specific objectives, which were established at the time the Request for Proposal was made to industry by the U.S. Air Force:

1. To fully explore the advantages of emerging technology.
2. To reduce the risk and uncertainties of full-scale development and production.

3. To provide the Department of Defense with a variety of options, readily available for application to military hardware needs.

The contract requirements to accomplish these objectives for the prototype program are very simple and straightforward:

1. Design, develop, and fabricate two prototype aircraft.
2. Assess and certify aircraft safety-of-flight.
3. Conduct a joint Contractor/Air Force flight test program.
4. Train Air Force test pilots.
5. Provide total contractor support during the flight test program.
6. Provide a data accession list.
7. Prepare a final report.

Further, under Item 1 above, complete latitude is provided in making trades; it is only required that the Contractor "Design, develop, and fabricate two prototype aircraft substantially in accordance with Contractor Technical, Management, and Cost Proposal FZP-1401, dated 18 February 1972" and acquiesce with an agreement letter covering a few miscellaneous revisions desired. There are no contractually obligated Statements of Work or detail specification requirements; the design responsibility rests solely with the Contractor. The program is planned for the first flight to occur 21 months after go-ahead, followed by a one-year period of flight testing of the two aircraft.

There is also a cost stipulation in the contract awarded. The total program cost committed by the Air Force to General Dynamics for two prototype aircraft, including one year of flight test, could not exceed \$37.9 million, and the average unit fly away cost goal for 300 aircraft was \$3 million in FY-72 dollars.

A three-view of the aircraft is shown in Figure 1. The design is a single-seat, single-engine configuration with a relatively low wing loading at combat weight (60 pounds per square foot) and a very high thrust-to-weight ratio (1.4). The structural design gross weight is 16,500 pounds; maximum design takeoff weight is 27,000 pounds. The YF-16 is a relatively small aircraft with a wing span of 30 feet and a length of 46.5 feet. The engine is the Pratt & Whitney F-100-PW-100, identical in configuration to that used in the F-15. In accordance with the objectives of the program, a number of advanced technology features for flight evaluation have been incorporated. (These are elaborated below in Section 4.)

The basic performance goals are relatively few and unencumbered. High maneuvering performance in a day visual air combat environment, with maximizing of the useable maneuverability and agility in the combat arena is desired. There is a single design combat mission that permits use of external tanks to fly to the combat zone, perform a prescribed set of combat maneuvers including a number of sustained turns at both subsonic and supersonic speeds at maximum power, and cruise back to the takeoff point. Intercontinental range performance with external tanks is also desired.

To achieve the maximum in maneuverability and agility it was necessary to assure that no flow anomalies exist at high angles-of-attack or yaw that would affect the flying qualities. To achieve the minimum size and weight aircraft, it is obvious high sustained turn rates are desirable which are directly a function of the drag at lift and the thrust available. The combat maneuver load factor is only required at the combat fuel weight which is added incentive to reduce combat weight and thus in turn structural weight. It is worthy of note that no specific high supersonic speed is required, nor is the design compromised for high speed, but a natural fallout of the high thrust to weight ratio places the airplane in the Mach 2 class.

In recognition of the exploratory technology concept of the program, a modular design approach was taken, and convenient manufacturing breakpoints are provided for flexibility in the test program or for evaluation of additional experimental configuration concepts, if so warranted, at completion of the present program. This concept is illustrated in Figure 2. Another feature of the design is the additional structural margins of safety of 25 percent incorporated in all primary flight structure. This design concept was adopted to permit flight testing up to limit design conditions without exceeding 80 percent of the limit design strength and without extensive static strength testing.

3. DESIGN CONCEPT AND APPROACH

The basic design concept was to apply advanced and emerging technology in a simple way to achieve the performance objectives in a minimum-size aircraft, with simplicity being the keynote. As previously pointed out, there were two cost goals established: the cost of the prototype aircraft and program, and the unit cost for a production quantity of aircraft. There is, of course, an additional cost objective that must be considered, although not specifically quantified, and that is the total life-cycle cost. While there are obviously trades that can be made between development, acquisition, and operating costs, small size and simplicity directly affect all elements of the total life-cycle cost in a beneficial way.

The importance of achieving a small-size aircraft from a cost standpoint is readily apparent. In most instances airframe costs can be considered a first order effect of eight other considerations such as material, detail design, and construction being equal. For example, on the design mission of the F-16, the growth factor is about 2.5. This means that for every pound of dry weight saved by new technologies, the aircraft gross mission weight decreases by 2.5 pounds - thus reducing design weight and spiraling dry weight down further. Similarly, for every count of drag saved by the new technologies, mission weight was reduced by 5 pounds - again, spiraling down the dry weight. The importance of advanced technologies in reducing aircraft size, and cost, is therefore of major concern in our preliminary design efforts.

The integration of advanced technologies with the objective of small size and simplicity involved a concerted and concentrated effort in the wind tunnel along with complementary parametric sizing and optimization studies, including single-versus-twin-engine configuration. The extent of this program is illustrated in Figure 3, where it is noted that 78 significant variations were tested, covering several different wing planforms and airfoil sections, fixed-and variable-camber wings, bifurcated and single inlets, twin and single vertical tails, and 50 variations in forebody strake design. Major emphasis was placed on achieving flight and configuration characteristics that would contribute directly to the air-to-air kill potential in the combat arena. These include the maximization of maneuver/energy potential and the elimination of aerodynamic anomalies up to the maneuver angle-of-attack limits.

In addition to the application of advanced technology to achieve small size and simplicity, the following principles were adopted to minimize costs:

1. Emphasize the new and novel technology features of the design that are significant in meeting performance requirements. Where new technology is not required, use proven systems and components wherever possible, particularly where only marginal benefits will accrue by redesign.
2. Establish specific cost goals and be willing to compromise or trade performance or operational capability to meet them.
3. Design for low manufacturing cost by making detail and component assemblies simple to manufacture, using low-cost materials and processes, standardizing hardware, and designing for multiple-use parts and assemblies.

While certain of these principles are usually associated with detail design, they do affect the basic design of the aircraft and therefore must be considered in the preliminary design phase.

It was recognized from the onset of the program that a somewhat anomalous situation existed, i.e., the design for a minimum-cost two-airplane prototype program would not necessarily be the same as that for an aircraft to be produced in quantity. However, most basic design and program criteria that significantly affect cost are applicable to both and, particularly to assure that the requirement for a viable production option was retained, there was no departure made in the basic configuration in the prototype because only two aircraft were to be manufactured. In fact, the aircraft was first configured in an operational form and then departures were made in detail to meet the specific objectives of cost and schedule required by the two-airplane prototype concept. The flight test aircraft will provide truly valid performance data for an operational aircraft.

4. APPLICATION AND INTEGRATION OF ADVANCED TECHNOLOGY AND INNOVATIVE DESIGN FEATURES

In consonance with the desires of the Air Force to "fully explore the advantages of emerging technology" as applied to a fighter aircraft to achieve exceptional maneuvering performance, a number of potential advanced aerodynamic and flight control concepts were considered. The criteria for selection were established as follows:

1. Must contribute directly to the performance/design goals.
2. Must be sufficiently advanced to warrant prototyping.
3. Must individually not be of such high risk as to jeopardize the total program.
4. Must fall within imposed constraints of cost, complexity, and utility.

There was an additional obvious consideration: compatibility between technologies was essential. The program is not structured to test individual advancements in the state of the art, but to test and evaluate a complete aircraft having the potential of development into an effective weapon system. It also is pointed out that the actual preliminary design procedure did not involve shopping through a list of advanced technologies and then attempting to structure the configuration. A more conventional design than that of the current YF-16 was evolved, and various advanced configuration features were analyzed and tested and then incorporated if proven that they did in fact meet the selection criteria. This effort extended over a period of two years before submission of a firm design proposal. The advanced technologies incorporated into the design are illustrated in Figure 4. The combined effect of incorporating these features in the design was to reduce the weight empty by 1300 pounds and the combat weight by 2200 pounds. In addition, other performance benefits were accrued and simplification was achieved that otherwise would not have been possible.

It is difficult to discuss the benefits of individual advanced technologies as applied to the YF-16 since they are completely integrated into the design - not only with each other but with other innovative design features not truly representative of "advanced" technologies. Therefore, for this discussion, the advanced technologies are categorized as those related to achieving high-maneuvering performance, as follows:

- . Variable-camber wing
- . Wing-body blending
- . Vertex lift
- . Relaxed static stability/fly-by-wire
- . Bottom inlet location
- . Composite materials

The innovative (and in some cases, "advanced") design features are separated into two categories: (1) those incorporated to improve pilot effectiveness, and (2) those incorporated to reduce cost. In the category of pilot effectiveness, the features are

- . 30-degree seat-back angle
- . Side-stick controller
- . Clear-view-forward canopy with 360-degree vision.

In the category of cost reduction, the features are

- . Single engine
- . Normal-shock inlet
- . System simplification
- . Multiple-part usage
- . Standardization
- . Materials selection.

These advanced technologies and innovative design features and the benefits accruing from their utilization are discussed below according to the category and the order given above.

A. High Maneuver-Performance Design Features

So that the desired performance objectives could be achieved, major emphasis was placed in the areas of aerodynamics, stability and control, and propulsion. Specifically, the requirement was to achieve at high angles of attack and sideslip an efficient L/D, low buffet intensities, good control characteristics, spin resistance, high thrust for sustained turn capability, and low inlet distortion.

Variable-Camber Wing. Variable wing camber was selected early in the preliminary design phase after a parametric wing study of wing airfoil sections, areas, and planforms. The effectiveness of this feature in increasing the sustained-maneuver L/D is shown in Figure 5. The chord of the leading-edge flaps is 30 percent at the wing tip and 18 percent at the root. This variation in chord along the span effectively provides a greater camber at the tip than at the root when the flap is deflected. The flap rate is 35 degrees per second, which is compatible with the airplane response in pitch and is

automatically programmed as a function of Mach number and angle of attack. Pitch rate is also supplied to the controller to provide more accurate tracking of the flap position with airplane attitude. In addition to the direct maneuver improvement gained from the leading-edge flap, it also proved beneficial in linearizing the pitching-moment curve and reducing angle-of-attack effects on directional stability. Although the leading-edge flap adds some complexity and cost to the design, it permits a wing having a smaller area and a lower t/c , with good performance in both the transonic and supersonic area of the flight envelope.

Wing-Body Blending. As shown on Figure 6, wing-body blending was adopted in an effort to reduce weight and achieve high volumetric efficiency, thus reducing overall size and cost. This approach resulted in a lower fineness ratio (3.5) than desired for the supersonic flight regime but provided a better wetted-area-to-volume ratio, partially offsetting the wave drag penalty and permitting a good distribution of fuel about the center of gravity. The calculated weight reduction and associated volume increase are 570 pounds and 9%. A wetted area reduction of 2% was realized. The use of blending also permits greater flexibility in tailoring the area distribution curve.

Vortex Lift. The initial effort in blending of the body lines forward of the wing leading edge resulted in poor pitching moment and less-than-desirable directional stability characteristics at high angles of attack. It was suggested by the NASA/Langley Research Center that sharpening of the forebody wing fillet lines from the relatively blunt leading edges that had been tested, which would deliberately strengthen the forebody vortex, could improve the air flow over the wing with an attendant increase in lift and improved stability. This possibility was investigated through a series of wind tunnel investigations wherein 46 forebody strake variations were tested to optimize the design and achieve the proper balance between aerodynamic performance and stability and control. This controlled vortex principle is shown in Figure 7; the effect on lift, stability derivatives, and buffet is shown in Figure 8. This concept is a direct result of prior investigations of highly swept double-delta wing configurations that indicate the benefits which could be derived from a strong inboard vortex that reattached the flow over the inboard wing area. Delaying the separation from the fuselage and inboard wing also provided an improvement in stability characteristics at high lift coefficients. Since there was concern over the effect of Reynolds number on this phenomenon, model tests were run at varying Reynolds numbers approaching those for the full-scale aircraft.

Relaxed Static Stability/Fly-by-Wire. Two major concepts integrated into the YF-16 design are the relaxed static stability (RSS) aspect of control-configured-vehicle (CCV) technology and a fly-by-wire (FBW) flight control system. RSS was adopted to enhance the maneuvering performance of the aircraft, and FBW was adopted to provide those desired handling qualities necessary to exploit the maneuvering capability of the aircraft. The current state of the art relative to flight control system designs incorporating high reliability design features and redundancy has permitted the integration of the two major concepts in today's aircraft designs.

In order to minimize the maneuvering trim drag, the aircraft was balanced so that the desired maneuvering performance is achieved supersonically. This approach to aircraft balance results in an aircraft that exhibits a longitudinal instability (negative static margin) at low C_L 's in subsonic flight of 6 percent MAC for the combat configuration, extending as high as 10 percent for large external store configurations. The aircraft balance, horizontal tail trim requirements, and the resulting effect on the trimmed drag polar are shown in Figure 9. The resulting effect on performance and weight is shown in Figure 10.

The amount of allowable instability at subsonic speeds is governed by the amount of control power required to maintain fully controllable flight, including gust effects, for all possible flight conditions. Other major considerations encompass short-period dynamics, aerodynamic and inertial coupling, dynamic directional stability, and lateral control-alone divergence.

A major benefit resulting from the adoption of relaxed static stability at subsonic speeds is the attendant reduction in static margin at supersonic speeds. The reduction in supersonic static margin results in the aircraft having the capability to maneuver at supersonic speeds to its design load factor without being limited by longitudinal control power.

The acceptance of the total FBW flight control system for aircraft today provides the tool through which the necessary aircraft flying qualities can be achieved to utilize the advances in aircraft aerodynamics and propulsion systems. The aircraft's flight control has no mechanical connection between the pilot control and the control surfaces. Pilot commands are made through a side-stick controller in the form of force inputs to command

a blend of normal acceleration and pitch rate longitudinally and roll-rate laterally. Directional commands are made through toe inputs at the rudder pedals.

All of the electronic components in the system are quadrupally redundant, including the power supplies. A backup power source is provided through batteries. A built-in-test capability is provided to check the total system prior to flight. The wiring between various components of the system has been separated to reduce vulnerability. Many of the system's design features have been adopted from the various technology programs sponsored by the Air Force and NASA.

Functional implementation of the flight control system to achieve desired flying qualities has been enhanced through FBW technology. The inescapable conflict between associated mechanical and electronic implementation has been eliminated. The designer is now able to shape electronically the desired pilot feel and aircraft response. Although the normal basic mechanical linkage has been eliminated from the aircraft, no major reductions in system weight has been achieved because of additional design requirements in the interfacing systems such as the hydraulic and electrical systems. However, one major benefit that materialized to reduce weight was the simplification of basic aircraft structure, i.e., bulkheads, frames, etc., that normally have to be compromised to support, permit passage through, etc., the mechanical linkage.

Bottom Inlet Location. The inlet location was specifically selected to meet the high-maneuver-performance requirements. The aft position was selected to take advantage of the shielding provided by the fuselage forebody, to provide a good area distribution, and to avoid excessive weight penalty in inlet duct lengths. Considerable effort was expended to wind tunnel testing to assure that flow conditions were satisfactory and that low distortion and turbulence were present at the compressor face. Particular attention was paid to diverting the boundary layer. A 0.15-scale inlet model was used in the testing, conducted in both General Dynamics and NASA wind tunnels. The model was instrumented with a 40-probe high-response rake to obtain distortion data for use by the engine manufacturer in assessing the inlet performance relative to engine stall tolerance.

The normal-shock design, selected specifically for simplicity and low cost, represents a compromise to the maximum speed performance. It is, however, completely in consonance with the requirement for the aircraft to achieve maximum performance in the combat regime. The inlet design features are shown in Figure 11.

Foreign object damage (FOD) was also considered in the design. The nose gear was located aft of the inlet with the lip 1.2 equivalent diameters above the ground to avoid ingestion of ground debris.

The fixed-geometry design incorporated on the prototypes is calculated to be 400 pounds lighter than a complete variable-geometry inlet designed for optimum performance over the structural design Mach/altitude envelope of the airframe. The forward inlet structure has been designed with two production breaks to permit possible future incorporation of inlets designed for high Mach number performance and also, if necessary, to permit flexibility in the test program. Several variations of inlet geometry and type have been investigated, and wind tunnel testing has been performed on several configurations ranging in concept from simple aerodynamically actuated two-position inlets to completely actuated variable-geometry designs.

Composite Materials. Though composite structural materials have not yet become competitive with metal structural from a cost standpoint, extensive use is made of graphite-epoxy composites in both the vertical and horizontal tails. These materials were selected specifically in recognition of the structural dynamic and aeroelastic requirements, with approximately 30 percent weight saving being achieved. The alternative would have been to increase the t/c, with an attendant drag penalty in the transonic and supersonic regions. It is expected that as more wide-spread useage is made of these materials the cost will become competitive with comparable metal structures.

B. Improved Pilot Effectiveness Design Features

In recognition of the human factors aspects associated with the unusually high maneuver potential of the YF-16 aircraft, several somewhat unconventional design features have been incorporated in the cockpit design to improve the man-machine interface. Since these features of the design (shown in Figure 12) have no particular implication relative to cost, they will not be discussed in detail, but they do affect the basic design and configuration of the total airframe and systems and are worthy of mention.

30-degree Seat-Back Angle. So that the pilot 'g' tolerance would be increased, the seat-back angle was increased to 30 degrees from the usual 13 degrees and the heel line

was raised. It has not been quantitatively proved that this configuration will appreciably improve the tolerable sustained "g" limits, but this seat geometry permits easier aft viewing and it is believed will aid in tracking in the high-"g" environment. The structural arrangement of the prototype aircraft will permit testing alternate seat-back angles if desired.

Side-Stick Controller. Along with the revised seat geometry, a side-stick controller has been provided with appropriate arm rest, which will assist in executing more precise combat maneuvers. This type of installation was made possible by incorporation of the fly-by-wire control system discussed previously. If desired, relocation of the control stick to a center location for test can be simply accomplished because of the fly-by-wire technology.

Bubble Canopy. The clear-view-forward bubble canopy permits unobstructed forward and up vision in the most important vision area. Good vision is also provided over the side and to the rear. A significant supersonic drag penalty is associated with the aft vision, but this compromise to performance is considered essential if the pilot is to operate at maximum efficiency. In view of these somewhat unconventional design features, a crew escape system sled test program is being run to prove their adequacy prior to flight. Also, special consideration was given to the canopy latch system because of the elimination of a forward fixed windshield. A set of safety latches is provided that is capable of withstanding the total imposed loads with both the normal and safety latch systems.

C. Low-Cost Design Features

In addition to the application of advanced technology directed specifically to achieve small size and overall design simplicity, a number of features (listed previously) have been incorporated into the YF-16 specifically to reduce cost.

Single Engine. The single-engine design was chosen after study of both single- and twin-engine configurations. In recognition of the questions surrounding the safety aspects of single-versus-twin-engine designs, an extensive review of accident and aircraft loss statistics was conducted. This study did not conclusively indicate a significant difference in the two choices, and the single engine design was selected on the basis of an approximately 15 percent lower design gross weight, simplicity, and lower cost. Controls, instruments, and other installation provisions are, of course, also one half that required for a twin-engine configuration. The fact that the F-100 engine is also used in the F-15 was an important factor in that all logistic support for the engine would be available in the Air Force inventory.

Normal-Shock Inlet. As mentioned previously in the discussion on high-maneuver performance, the normal-shock design of the inlet was selected specifically for simplicity and low cost. If, however, a variable-geometry inlet is desired for future growth, the design (production breaks) will easily accommodate such a change.

System Simplification. Existing system configurations were used where new technology was not required or where only marginal benefits would accrue by use of new design concepts. This approach permitted the use of a large number of existing, fully developed components or only slightly modified components used on other aircraft. In addition to the initial procurement cost saving, use can be made of existing spares, repair tools, and data already in the Air Force inventory. A specific cost saving has not been established for adopting this concept, however, it is obvious that the ultimate saving in total life-cycle cost would be very significant should the airplane be procured for the Air Force inventory. It is also obvious that some penalty is associated with this design concept; it is not apparent, however, that the overall configuration has in any way been compromised.

Multiple-Part Usage, Standardization, and Materials Selection. In order to minimize tooling and reduce cost, a number of components in the airplane have been designed for multiple use, as shown in Figure 13. These include the horizontal tail, which may be used on either the left- or right-hand side; the wing trailing edge was modified slightly from the theoretical contour so that the flaperons are flat straight wedges that may be used on either the left or right side of the airplane; 80 percent of the landing gear parts, used on either the left- or right-hand main landing gear; a single electro-hydraulic flight control servo used in five places for actuation of the two flaperons, horizontal tail, and the rudder; a single hydraulic valve, used on all five control surfaces; hydraulic rams for surface actuation, identical for the flaperons and horizontal tail; the leading-edge drive system, employing four rotary actuators per side and requiring only two different actuator assemblies. In addition to these more significant items, a large number of smaller detail parts have been designed for multiple usage.

In addition to the airplane peculiar parts mentioned above, the number of different types and kinds of standard parts were considerably restricted (Figure 13). A special standards book, containing a minimum number of selected standard parts having the most universal application and low cost, was provided to the designers. A typical example is the reduction in the number of fastener types used. The YF-16 uses 55 as compared to 232 on the F-111.

A somewhat similar approach was taken on the use of raw materials. A separate restricted list was provided to the designers, and the application of low-cost materials was encouraged. In the case of the structural design, the principal material used is aluminum (Figure 13).

5. PROTOTYPE PROGRAM

In conclusion, a few comments on the conduct of the prototype program for the YF-16 seem appropriate.

The extensive effort put forth in the preliminary design phase of the program has had a very beneficial effect on the subsequent design, tooling, and manufacturing phases of the prototype program. Only a small number of refinement changes have been required, and the airplane has been produced with only minor variation from the proposal design. The wind tunnel program conducted after authorization to proceed has been a data-gathering process for the generation of design information, and it has not been necessary to test a large number of configurations to resolve problems or to achieve additional optimization. The schedule and cost budget have been rigorously adhered to.

As was noted earlier, a simple systems approach has permitted extensive use of off-the-shelf fully qualified components. Of the 432 purchased components used in the airplane, 332 are used as is or with only minor modification. This has appreciably reduced the test program, has resulted in ready availability of parts, and will produce a high degree of reliability in the flight test program.

As mentioned earlier, there were no specific detail specification requirements, and the design responsibility was placed with the contractor. In actual fact, the airplane is being produced essentially to the usual military specifications with departures only as necessary and when good engineering judgement so dictates. The Air Force monitoring of the program, while informal, has been complete and thorough. The Air Force has been provided all contractual generated data at their specific request. In each case where significant departure has been made from usual requirements, the Air Force Project Office was advised and, even though they are not required to formally approve, agreement was sought. These procedures have obviously expedited the program, minimized the cost, and been a major contribution to the rapid progress (with minimum problems) evident today.

STRUCTURAL DESIGN GROSS WEIGHT 10,500 LBS
 MAXIMUM T.O. GROSS WEIGHT 27,000 LBS
 FUEL CAPACITY 12,185 LBS
 • INTERNAL 6,400 LBS
 • EXTERNAL (2) 370 GAL + (1) 150 GAL 5,785 LBS
 ENGINE P&WA F-100PW-100
 W/S (60 TSF (DESIGN MISSION WEIGHT)) T/W = 1.4

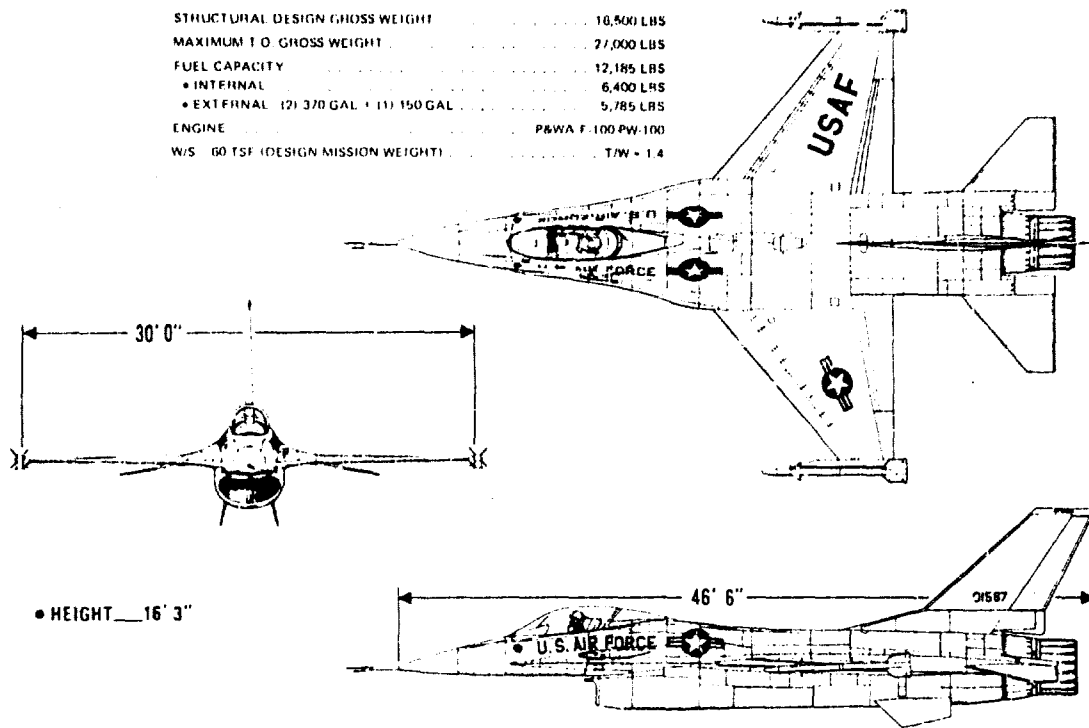


Figure 1 General Arrangement YF-16

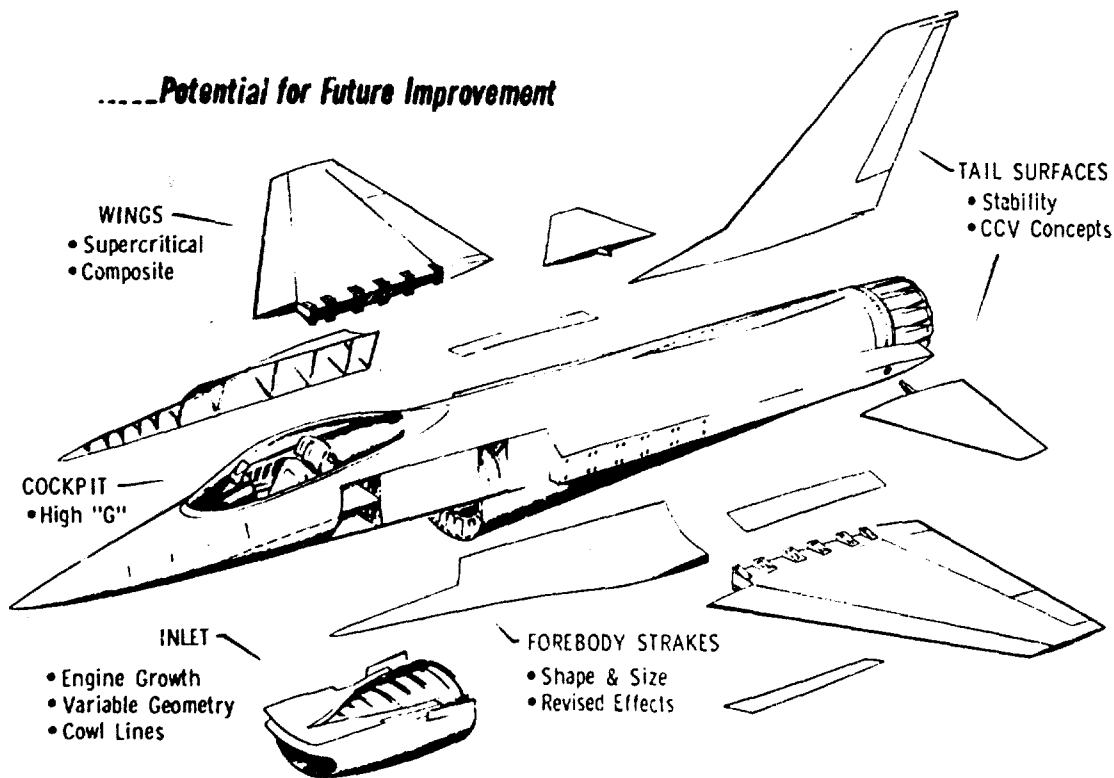


Figure 2 Prototype Versatility

• 78 SIGNIFICANT VARIATIONS • $M = .2 - 2.2$ • $\alpha = 28^\circ$ • $\beta = 12^\circ$

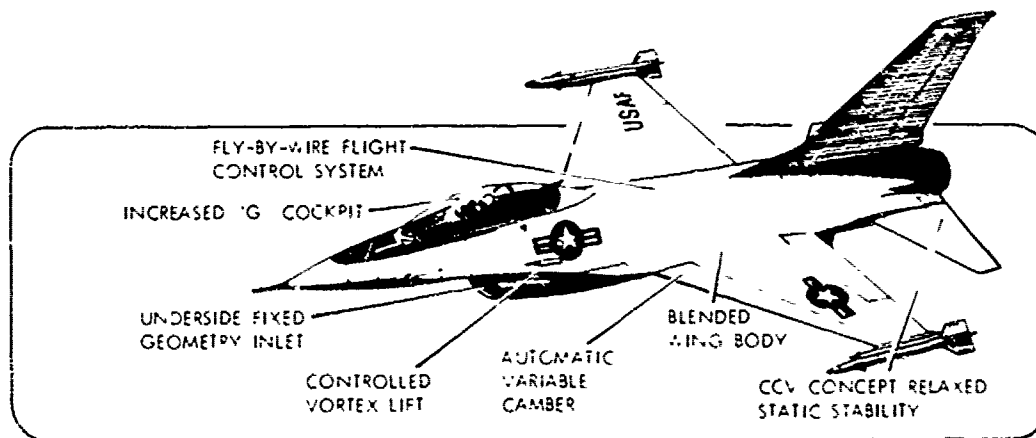
Configurations Tested	A _{1/4}	WINGS		AIRFOIL ID	SLEETS		VERTICAL TAILS		VORTEX LIFT (# of body tested)	WIND TUNNEL TEST HOURS
		TRAIL CAMBER	INBOARD CAMBER		LOW	HIGH	TRIM	SHOULDER		
783	40°	✓		84A200 & 84A403	✓	✓	✓	✓	48	
786	35°	✓		84A200 & 84A403	✓	✓	✓	✓	28	
786	40°	✓		84A200 & 84A403	✓	✓	✓	✓	48	
40W-0	35°	✓	✓	4.1 INCORPORATED	✓	✓	✓	✓	107	
40W-2	40°	✓	✓	84A200	✓	✓	✓	✓	91	
40W-3	25°	✓	✓	84A200 & 84A403	✓	✓	✓	✓	28	
40W-4	40°	✓	✓	84A200	✓	✓	✓	✓	29	
40W-5	40°	✓	✓	84A200	✓	✓	✓	✓	29	
40W-6	40°	✓	✓	84A200	✓	✓	✓	✓	138	
40W-7	40°	✓	✓	84A200	✓	✓	✓	✓	30	
40W-8	40°	✓	✓	84A200	✓	✓	✓	✓	36	
40W-9A	40°	✓	✓	1.1A200	✓	✓	✓	✓	32	
40W-10	40°	✓	✓	84A200	✓	✓	✓	✓	442	
40W-11	45°	✓	✓	CORICAL CAMBER SPACE & BLENDED	✓	✓	✓	✓	126	
40W-12	40°	✓	✓	84A200	✓	✓	✓	✓		

Wing Moved Forward 1/4chords

TOTAL WIND TUNNEL HOURS 1272

Figure 3 LWF Force Models Tested

ADVANCED TECHNOLOGY – DESIGN INNOVATIONS



- INTEGRATED APPLICATION OF VERY LATEST FIGHTER TECHNOLOGY
 - Equivalent to a 2200 lb Reduction in Mission Wt.
 - Increases Maximum Usable Lift
 - Eliminates all Flow Anomalies at High Angles-of-Attack
 - Maneuver to Aerodynamic (Lift) or Load Limit

Figure 4 Advanced Technology • Design Innovations

VARIABLE CAMBER MAXIMIZES L/D

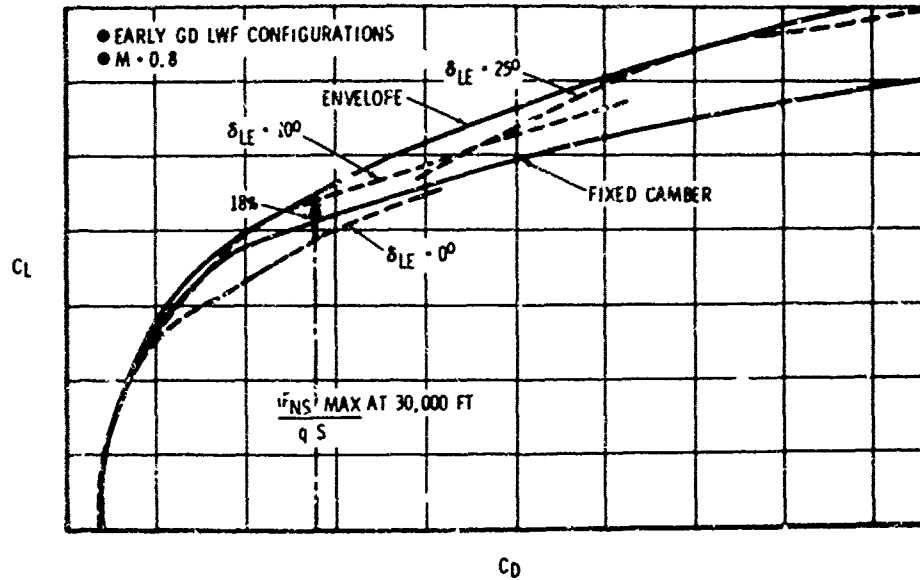
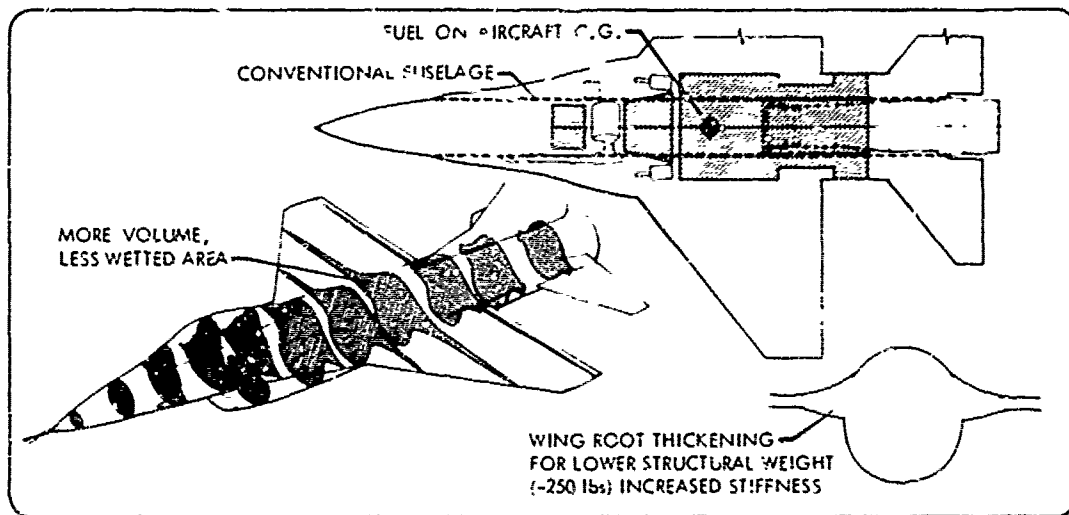


Figure 5 Variable Camber Maximizes L/D

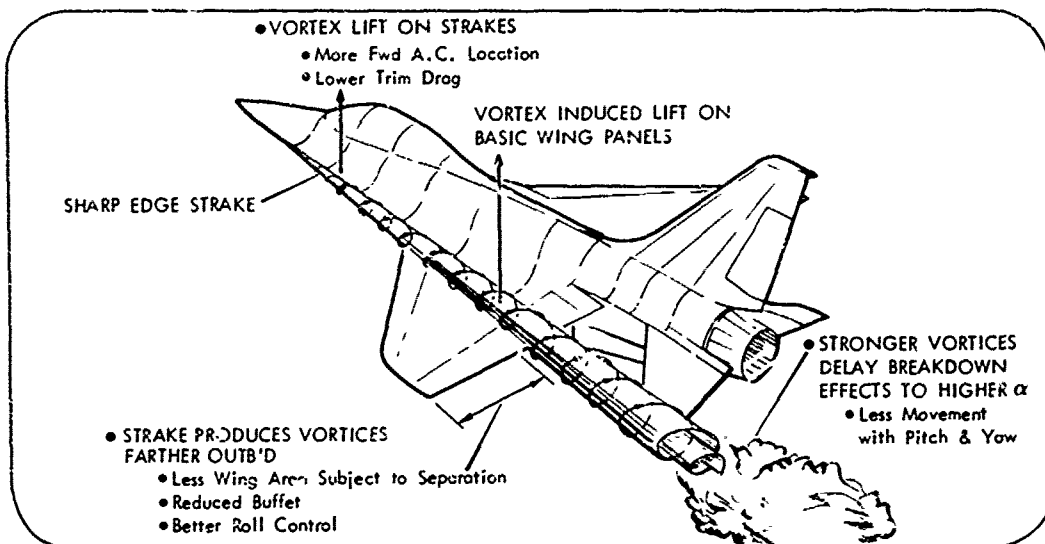
WING-BODY BLENDING



- INCREASED BODY LIFT
- IMPROVED AREA DISTRIBUTION - SMOOTHER CROSS-SECTION AREA CURVE, REDUCED TRANSONIC DRAG
- BETTER WETTED AREA-TO-VOLUME RATIO
- BETTER VOLUME/FUEL DISTRIBUTION - CONSTANT C.G.
- SHORTER FUSELAGE (5.5 ft) ≈ 320 LBS

Figure 6 Wing-Body Blending

CONTROLLED VORTEX LIFT (FOREBODY STRAKES)



- HIGHER LIFT PER UNIT OF EXPOSED WING AREA
 - Effective W/S = 52 at $M = .9$ and 41 at $M = 1.2$ (Geom. = 60)
 - Equivalent Wing would Weigh +490 lbs
- GREATLY IMPROVED DIRECTIONAL STABILITY
- REDUCES TRIM DRAG
 - Straightens Pitching Moment Curve

Figure 7 Controlled Vortex Lift (Forebody Strakes)

STRAKES IMPROVE USABLE LIFT

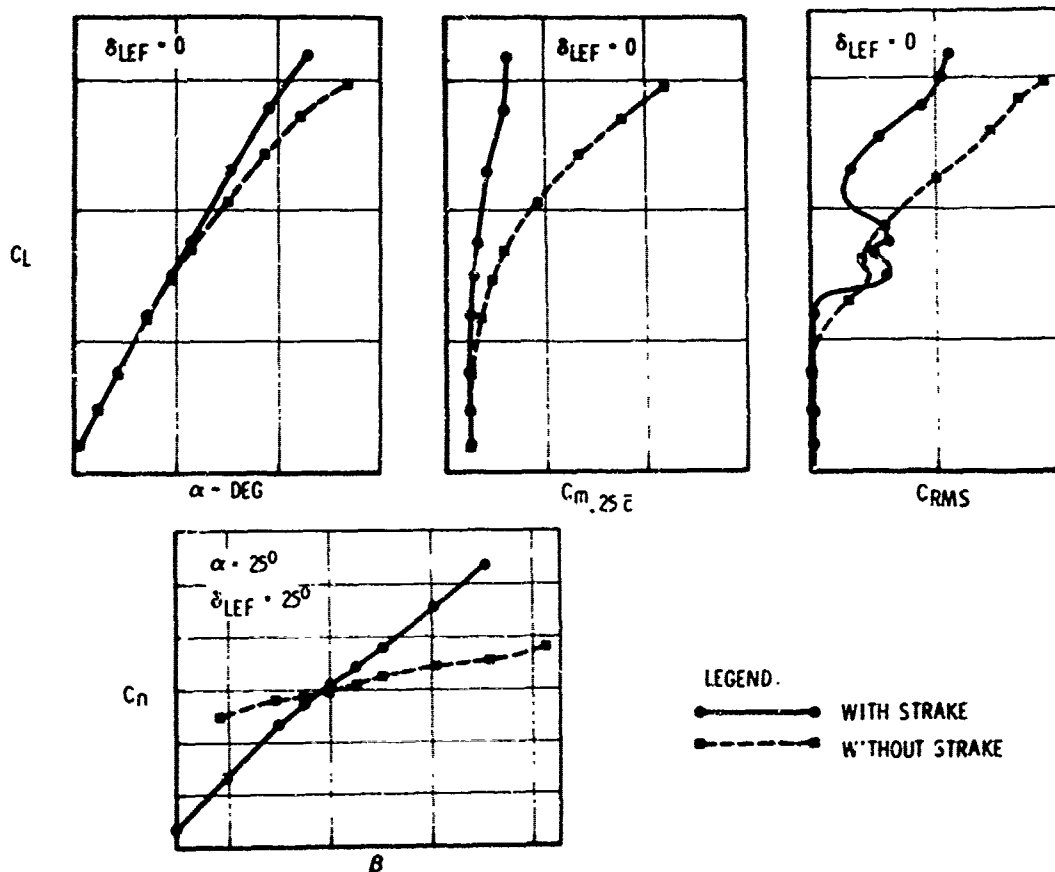


Figure 8 Strakes Improve Usable Lift

CCV TRIM AND MANEUVER IMPROVEMENTS

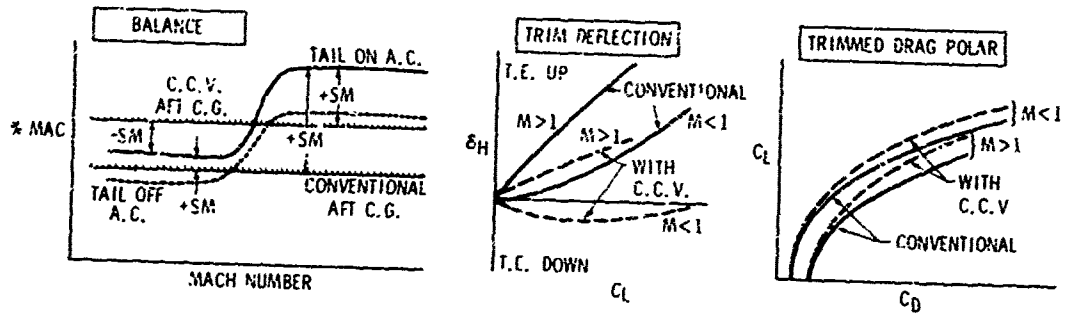


Figure 9 CCV Trim and Maneuver Improvements

CCV PERFORMANCE IMPROVEMENT

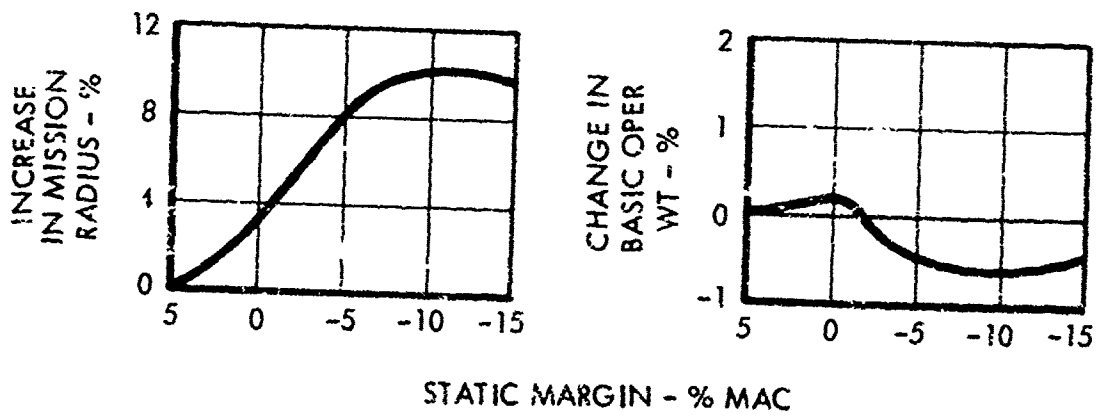


Figure 10 CCV Performance Improvement

INLET DESIGN FEATURES

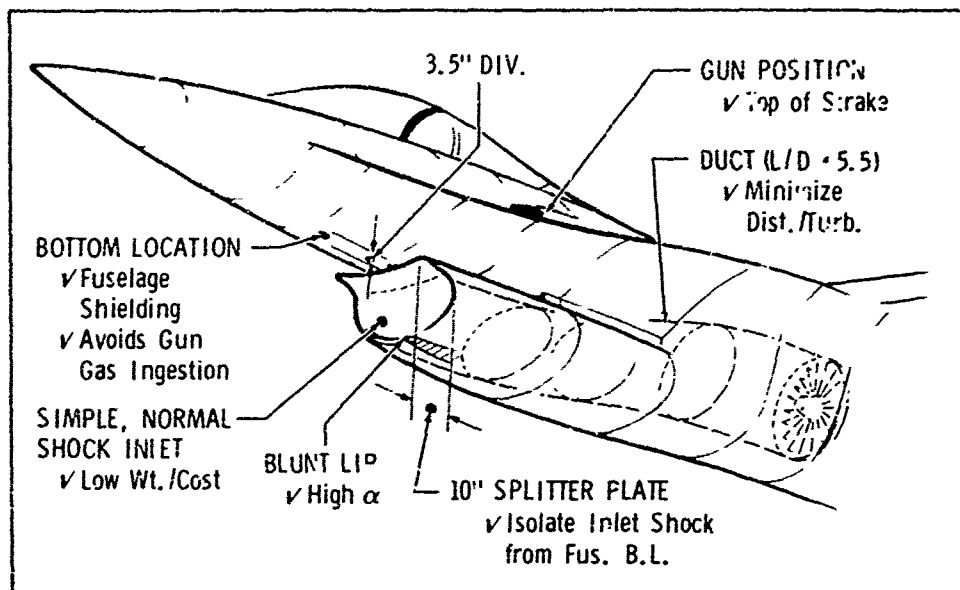
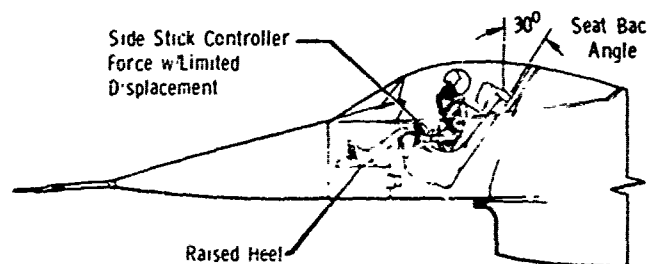


Figure 11 Inlet Design Features

FEATURES FOR IMPROVED PILOT PERFORMANCE

IMPROVED PILOT PERFORMANCE IN ACCELERATED FLIGHT



UNOBSTRUCTED VIEW IN PRINCIPAL SEARCH AREA

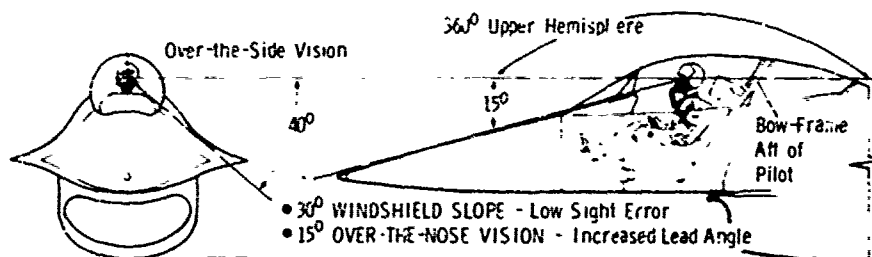


Figure 12 Features for Improved Pilot Performance

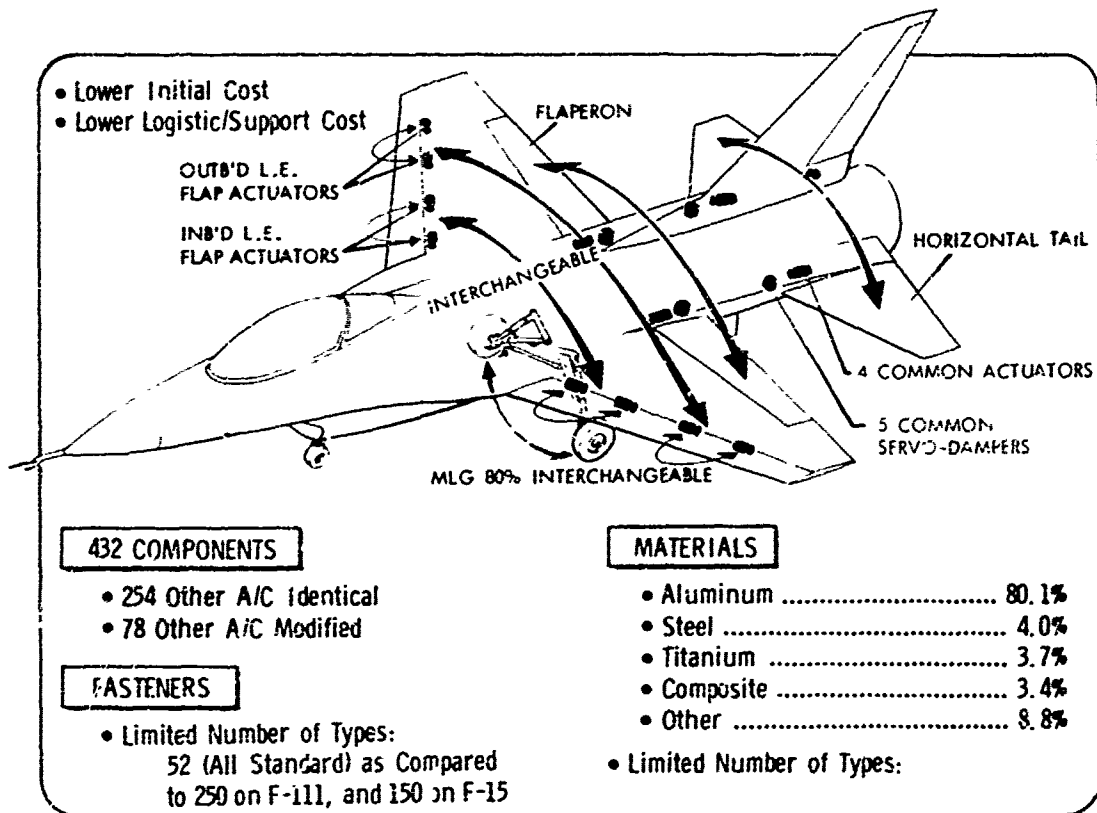


Figure 13 Configure/Design for Cost

ECONOMIC ASPECTS OF PROTOTYPING

by

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INTRODUCTION

The R & D process exhibits features of uncertainty in many of its phases. Government personnel draw up specifications in great detail and contractors try to respond in even more voluminous detail, to demonstrate (on paper) their capabilities to meet them. (The three competitors for the C-5A contract, for example, submitted a total of 240,000 pages of basic documents.)

The flood of paperwork continues through the entire acquisition process as long as the hopeless struggle of giving a final estimate on cost lingers on. Early prototyping appears to be an economic development concept to reduce the inherent risk in the transition of design advancements into operational hardware. The simplicity of this method can bring substantial weapon system cost saving and could help to keep good design teams together in times of few acquisition programs.

This paper attempts to analyse the cost aspects and schedule implications of a prototype development concept in comparison with present day development philosophies.

TIMESCALE

RDT & E or total system development programs begin with a conceptual phase. Contractors introduce their design concepts for a weapon system, their way of responding to the performance requirements of an RFP. The submittal of a proposal for design, manufacture and evaluation, as well as prices and operating cost usually terminates competition, at least in Europe (fig. 1).

The pacing item for first flight is normally the availability of an advanced engine, its program has to be launched in parallel.

If the program behaves in a way which one might call "good natured", the evaluation and demonstration of required weapon system performance of a frame, engine, avionics, armament etc. will proceed according to plan. An operationally sound aircraft would be expected 7 to 8 years after initial go-ahead. But there is the "if"; the unforeseeable can jeopardise the whole system of program milestones, set to interlock like gears. History is full of examples.

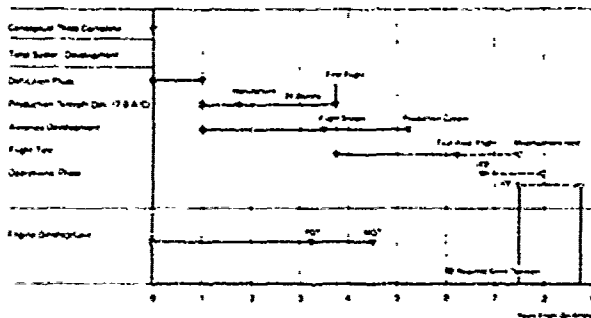


Fig. 1 Total System Development Program

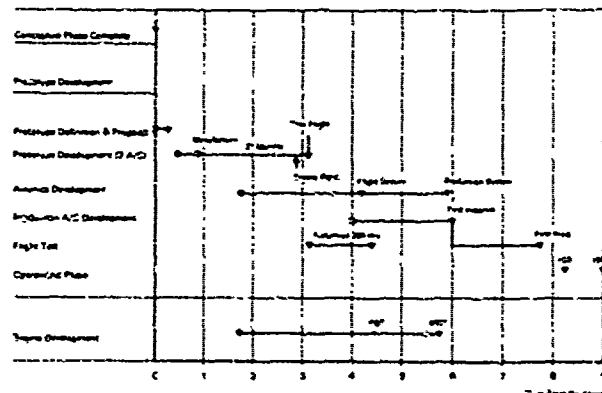


Fig. 2 Prototype Development Program

The prototype concept (fig. 2) is similar to the experimental production concept used in the US until the 1950's and becoming more attractive again for newer projects (AX, Light Weight Fighter, Mirage-family, technology prototypes). The customer gives contractors only a rough outline of the kind of weapon system required, leaving the detail to the designer. An RFP could be as short as two pages. Hardware powered by an available engine is put into the air as early as possible. Program relevant and critical characteristics can be tested and evaluated in a realistic environment (flight testing). Adequate time is left for the development of advanced avionics and engines and their

eventual modification of requirements due to the findings of early flight testing. The prototypes could serve as flying testbeds. Decision for further approval will be based on hardware experience not on paper analysis. A pessimistic estimate of this concept delays the I.O.D. (initial operational delivery) date by 12 - 15 months.

COST

The cost of total system development programs is difficult to analyse. Prices tagged on such a program must be estimated and negotiated from Contract Definition Phase information only.

Cost for a production aircraft development program based on the prototype program will be more realistic and reliable, most of the estimates will be based on hardware experience.

But what is the extra cost of monetary risk improvement? Is it too much in relation to the cost of programs which do without those precautions, considering that a successful prototype still requires full operational development, utilising another 4 prototypes?

Fig. 3 through 6 outline the major differences of activities and cost, identifying what it entails to design "Skunk work" (or "knife and fork") prototypes.

Engineering applies less refined analysis methods, aircraft are manufactured from red line (or preliminary) drawings, system analysis is shifted to a later phase, engineering work is transferred to special prototype shop organisations, taken out of company overheads. Column 2 of fig. 3 indicates the reduction potential, but 54 % of the total engineering effort planned is still required, not considering production engineering.

	Fraction of Cost Total System Development %	Reduction of Cost Prototype Development %	Fraction of Cost Prototype Development %
Structural Design	33.1	30.0	35.8
Subsystem Design	12.1	30.0	0.2
Design Analysis	16.8	30.0	12.1
Engine Integration	3.5	30.0	7.8
Qualitative Analysis	22.3	100.0	8.8
Systems Engineering	8.2	80.0	3.7
Engineering	100.0	48.7	84.2

Fig. 3 Initial Engineering

	Fraction of Cost Total System Development %	Reduction of Cost Prototype Development %	Fraction of Cost Prototype Development %
Windtunnel Tests	25.3	91.0	2.3
Airframe Groundtests	17.2	88.8	0.8
Airframe Subsystem Tests	15.8	88.0	3.0
Subsystem Tests	32.0	84.5	5.1
Armament, Stores	4.0	100.0	0.0
Escape System	0.7	100.0	0.0
Ground Tests	100.0	88.0	11.0

Fig. 4 Prototype Ground Testing

- Analysis of required ground testing applies to four groups of the various test areas
- Basic test, necessary for the functional integrity and verification of basic analysis (for example: low speed and high speed wind tunnel tests with a complete model to verify basic aerodynamic characteristics, particularly stability and control derivatives)
- Go-no-go tests (e.g. landing gear drop tests, thrust reverser re-ingestion tests)
- Performance verification tests (e.g. half model high speed test to verify usable lift boundaries, load distribution tests)
- Refinement tests (e.g. auxiliary intake test to improve intake distortion).

	Fraction of Cost Total System Development %	Reduction of Cost Prototype Development %	Fraction of Cost Prototype Development %
Labour	87.0	80.0	77.4
Material	13.0	80.0	2.6
Tooling	100.0	80.0	20.0
Airframe	90.8	83.0	8.8
Equipment	8.8	80.0	1.4
Armament	11.8	80.0	1.2
Propulsion	16.1	11.0	4.7
Engineering	8.8	83.0	7.7
Prototype	100.0	82.4	17.6
Stores	71.3	80.0	7.1
ACE	71.8	100.0	0
Data	1.7	100.0	3.0
Support	100.0	82.8	7.1

Fig. 5 Prototype Tooling, Manufacturing & Support

	Fraction of Cost Total System Development %	Reduction of Cost Prototype Development %	Fraction of Cost Prototype Development %
Handling	73.7	90.0	8.8
Performance	2.7	90.0	1.8
Flaps	4.8	80.0	1.0
Landing	7.8	80.0	0.4
Engine/Inlets	10.2	90.0	5.1
Aircraft Systems	4.1	80.0	2.8
AFC/CSAS	3.4	30.0	7.4
Air Data System	1.4	80.0	0.3
Communication	2.8	100.0	0.2
Navigation	7.8	100.0	0.0
Air Traffic	4.1	100.0	0.8
Target Following	8.8	100.0	0.0
Weapon Delivery System	27.7	100.0	0.3
Maintenance	2.4	100.0	3.8
Basic Aircraft F...	48.8	77.3	18.8
Armament, Operation	88.0	100.0	0.0
Flight Test	100.0	80.0	10.0

Fig. 6 Flight Testing

All basic tests are retained for the prototype approach or only reduced in scope. Go-no-go and performance verification testing are either eliminated (most of fatigue test) or retained in total (thrust reverser re-ingestion) according to their criticality to the prototype program and/or the following production. Thus a reduction of cost down to 11 % is achievable, although many of the eliminated tests will have to be conducted later, if the program continues as a production order (fig. 4).

Soft tools for only two prototypes are assumed to the extent possible, considering the metals in the airframe. Simpler and less formal manufacturing methods are applied, with weight penalties accepted, only off-the-shelf equipment and flight necessary avionics are used and few spare parts are planned to maintain the flight program (fig. 5).

Roughly 200 hrs flight testing are required for the essential "basic" flight tests. All flights devoted to testing of operational equipment and capabilities are disregarded, thus only 10 % of the cost of a full size test series of a total system development program remain (fig. 6).

Fig. 7 summarizes the initial cost for two prototypes. The achieved figure of 27 % applies only to a specific program but indicates an effective way of hardware acquisition. The remaining 73 % is devoted to the development of operational capabilities (31, 5 %) and to aircraft production in quantities (fig. 8).

Development Activity	Fraction of Cost Total System Development %	Reduction of Cost Prototype Comparison %	Fraction of Cost Prototype Development %
Engineering	32.5	46.7	17.8
Ground Test	11.9	58.0	7.0
Flight Test	14.7	80.0	1.8
Tooling	12.8	96.0	2.7
Prototype	18.2	87.5	2.4
Support	8.8	82.8	0.8
	100.0	72.8	27.0

Fig. 7 Summary: Reduction of Initial Cost

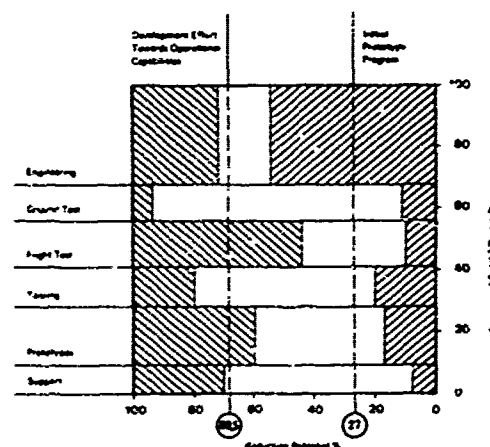


Fig. 8 Initial Prototyping - Reduction Potential

CASH FLOW

The financial requirements of the two philosophies are characterized by the relatively low expenditure at the first assurance mark (first flight) of the prototype program. Both curves of fig. 9 represent almost ideal progression and the accumulated cost for the first batches of operational aircraft should be somewhat less in the case of a complete RDT & E program. (Ref. (1) indicates that the costs accrued at the 100th unit is about the same for both types of plans).

Major aircraft modifications are often required and frequently necessary to incorporate changes resulting from flight tests. At the time such changes are defined and authorized (or not, when design goals are sacrificed) a defined quantity of production aircraft has already been launched in the manufacturing process. Aircraft modifications prior to I.O.D. may cause an appreciable increase in cost. Delays of 1 year (~ 15 %) and more are not unusual and amount to additional development work (redesign) of about 11 % (this is an estimate for a specific program) totalling in an increase of 10 % of program cost at the 100th unit over the comparable prototype plan. (The F-15 program consumed at a comparable stage 1.5 M \$ per day (reference 2)).

Fig. 10 compares cost at possible program termination. The motivation of cancellation can be manifold: changes in the tactical environment or technical reorientation.

TECHNICAL AND MONETARY RISK

The question of "which plan offers the highest probability to develop the most successful weapon system?" still remains. "A probabilistic approach to aeronautical research and development" (Reference 3) attempts to quantify probabilities of setbacks, but leaves the final answer to a great part to subjective elements such as the degree of advancement or actual cost. Fig. 11 indicates the capability of predicting program goals at the end of a Contract Definition Phase. Engineers must cope with 20 % and more deviation; perhaps these uncertainties can be removed by early flight testing. (The simulation included only the uncertainties of airframe development; weights, volumes and performances of engine and avionics were assumed as fixed inputs).

Those deficiencies of course result in similar unreliability of program cost estimates and time scale prediction. Again the spread of the cost estimate tends to be optimistic, uncertainties of engine and avionics prices were removed by using off-the-shelf equipment.

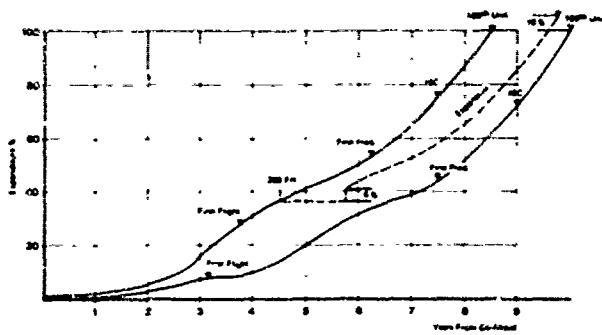


Fig. 9 Budget Requirements

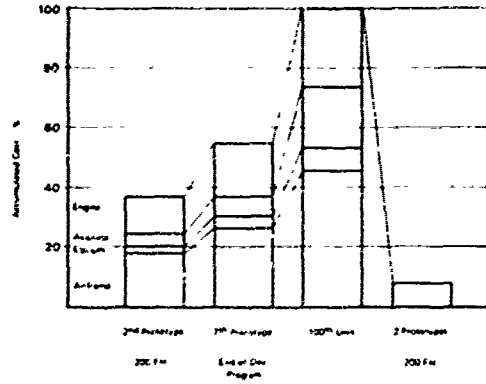


Fig. 10 Accumulated Cost at Program Milestones

Fig. 12 further indicates that the eventual additional cost of an initial prototype program is well within the uncertainty boundaries of a full scale program (~ 3% at a quantity of 300 A/C). Ever redundancy would be justified, going through a fly-off competition and thus resulting in a product of improved technical quality. An initial prototype assures contractors sufficiently that they could agree to contract forms with minimum or no risk to the government.

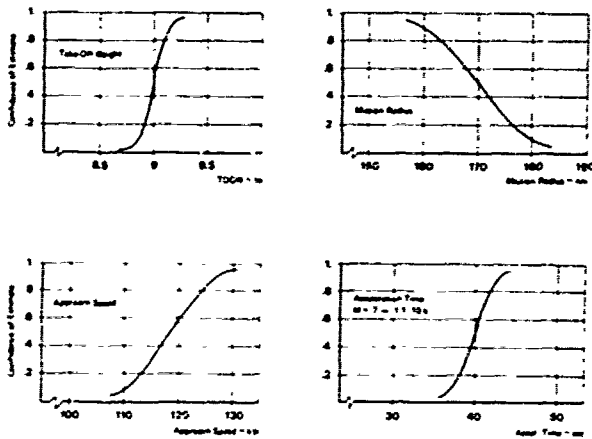


Fig. 11 Specifications Uncertainties

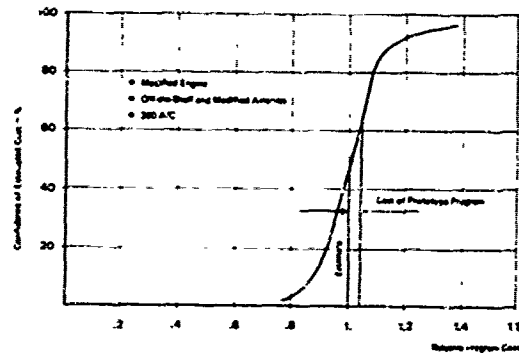


Fig. 12 Confidence & Program Cost

ORGANISATION

Future European military development will probably continue as multi-national ventures. Most recent programs have proven the feasibility of multilateral management systems. Fig. 13 describes the management structure of possible international collaboration. A multi-national decision making governmental organisation with its executing agencies, which handle the project, is "opposed"

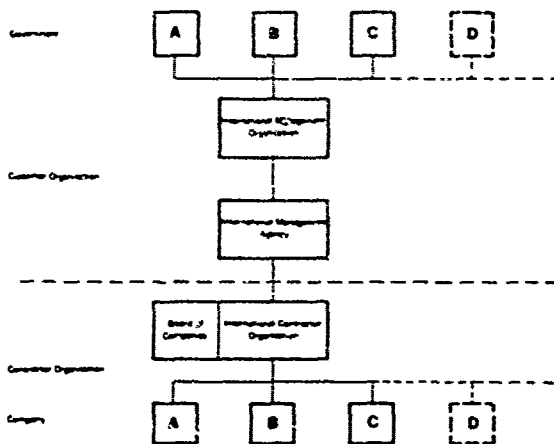


Fig. 13 Management Organisation - Total System Development

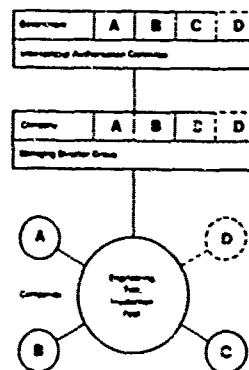


Fig. 14 Management Organisation - Prototype Development

by an international contractor organisation, which is responsible for the design, development and production effort of the various participating companies. An initial prototype program, on a somewhat lower effort level, should function sufficiently well, using a "stripped-down"-version of a highly complex management structure (fig. 14). A group of government representatives, one from each participating country, form an authorisation committee to approve required budgets and to set up a loose framework of specifications. A group of representatives from participating companies is solely responsible to that committee. They head an engineering, test and production pool, supplied by the participating companies and operating by its own economic regulations. A most effective product (in terms of time, cost and performance) would be achieved, if all activities could be placed in one location. A follow-on production development program should of course return to the large scale management structure.

SUMMARY

A final answer to which approach ultimately is cheaper and yields the better product cannot be given. The degree of advancement determined by engineers, the military situation and political considerations, related to their subjective importance must lead to the overall decision. But many positive prospects of initial prototyping, should attract decision making organisation:

- o an advanced aircraft would favour an initial prototype approach,
- o the relative small sum of money required would even justify a prototype fly-off-competition,
- o initial commitment at the time of least information would be kept to a minimum,
- o the cost of the "hardware" approach is well within the predicting boundaries of total program cost,
- o technical excellency will be improved to a degree as competition and redundancy increases,
- o prototyping gives to the engineer the opportunity more often to work hardware oriented, thus keeping good design teams together,
- o an eventual delay of the production program could be reduced by moving the go-ahead to an earlier date or by accelerating the production delivery schedule at a modest increment in total cost.

REFERENCES

- 1 Planning Research Corporation, "Methods of Estimating Fixed-Wing Aircraft Costs", PRC R-547, February 1, 1965
- 2 Aviation Week, June 26, 1972
- 3 R. A. Harvey, "A Probabilistic Approach to Aeronautical Research and Development", The Aeronautical Journal of the Royal Aeronautical Society, May 1970

CREATIVE ADVANCED DESIGN -- A KEY TO REDUCED LIFE-CYCLE COSTS

By

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INTRODUCTION -- The Need for Creative Advanced Design

Advanced design decisions have a very significant impact upon the life cycle costs of aircraft systems. The impact of Advanced Design decisions commences during the detail design and development phases, continues into the production phase and becomes a dominant factor during the operational life of the aircraft system.

Even for a well defined mission there are many factors bearing on costs that must be considered during the Advanced Design phase. Several of these have been selected for examination:

- The number of engines to be used
- Selection of equipment and systems
- The simplification of design
- The materials to be used
- The level of avionics sophistication

These Advanced Design decisions cannot be made solely from a technical or cost standpoint, but must consider the operational environment in which the aircraft system will function as well as the demands of the market place.

There are strong indications that the more intense the Advanced Design effort the lower will be the detailed design, manufacturing and development costs as well as life cycle costs, Figure 1. During the Advanced Design phase, it is extremely important that the critical wind-tunnel tests be performed to avoid costly redesign after the detailed design drawings have been released. The contribution that creative Advanced Design makes to the development of effective aircraft systems has greatly increased as the result of rising costs. Errors in judgment are now more costly than ever before.

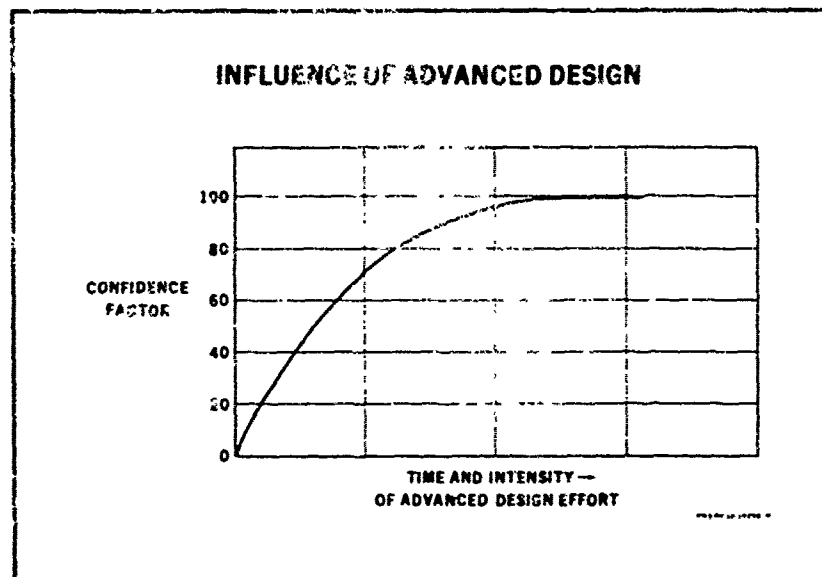


FIGURE 1.

NUMBER OF ENGINES TO BE USED -- A Fundamental Decision

One of the most important Advanced Design decisions is the number of engines to be used to meet a given design requirement. This determination is affected by airplane range, operating cost, and whether existing engines are used or new engines are to be developed to satisfy the requirements. Before these Advanced Design judgments can be made, a technical analysis must be completed to determine the optimum number of engines to best accomplish the design mission at minimum cost, assuming the availability of engines. In the real world, factors other than the optimum number of engines to meet a specific design requirement are considered. Flight safety is of paramount importance even though an accurate assessment of the cost/benefits of increasing safety is usually difficult. Growth potential and market characteristics must also be factors in the determination of the configuration.

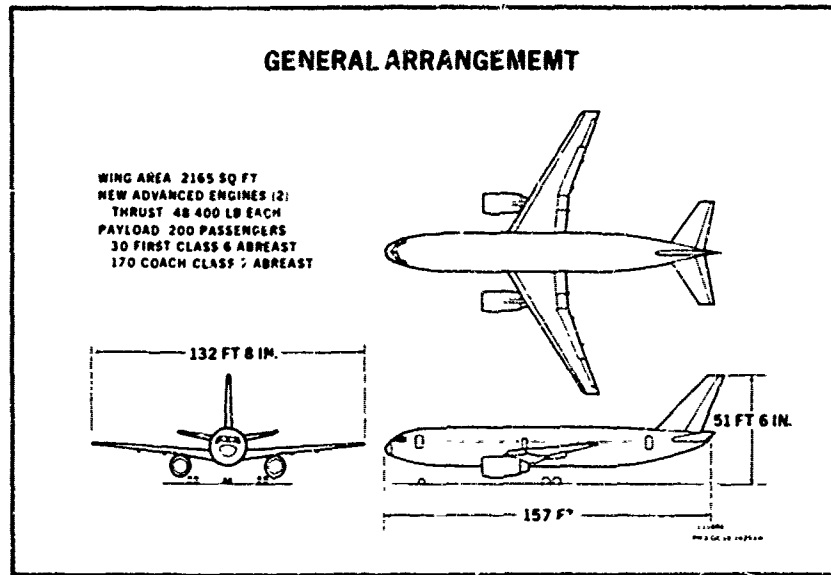


FIGURE 2.

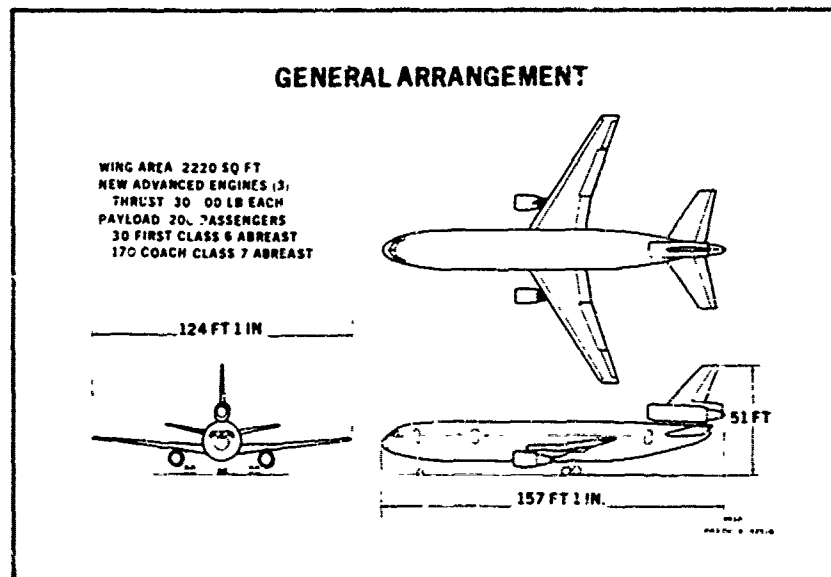


FIGURE 3.

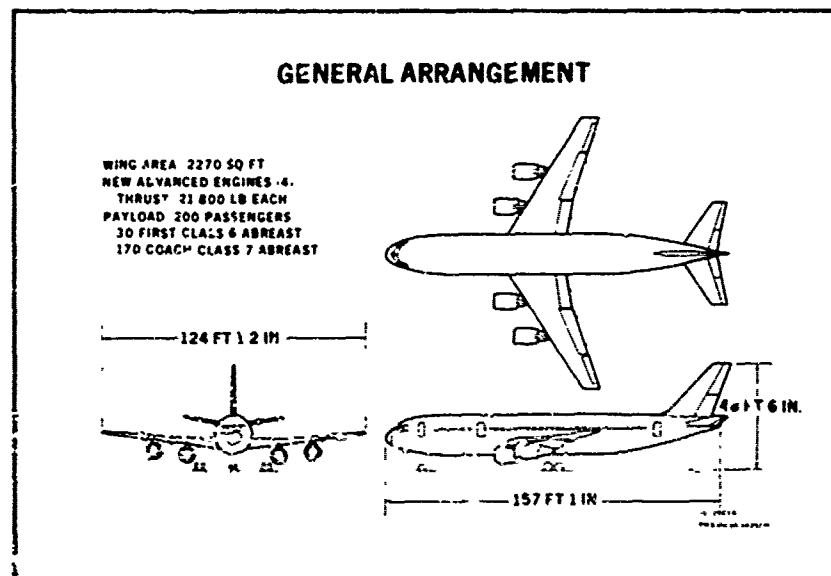


FIGURE 4.

A recent Douglas study to determine the number of engines that should be used on a new airplane to replace the present narrow-body twin and tri-jets in the 1980 time period is of interest. Market studies indicate that this aircraft should have a seating capacity of 200 passengers, a design range of 2250 nautical miles and a cruise speed of 0.85 Mach. Properly sized configurations having two, three and four engines meeting this requirement were developed, Figures 2, 3 and 4. New advanced technology engines optimized for the 2250-nautical-mile mission were used.

The characteristics of these three airplanes are shown in Figure 5. A market size of 250 airplanes was used in pricing each configuration. The prices of the all new engines are based on the assumption of similar engine production quantities. The baseline twin engine airplane has the lowest direct operating costs, a four-engine airplane has 3.4 percent higher direct operating cost and a tri-jet has direct operating costs 4.1 percent higher than the baseline twin. These calculations assume that complete engine development costs would have to be undertaken for each case. In this study the twin engine design has an engine size close to an existing engine. The saving in development costs reflected in a lower engine price would, therefore, significantly lower the direct operating cost of the twin even further if the existing engine were used.

COMPARATIVE CHARACTERISTICS			
ENGINES NUMBER	2	3	4
• TYPE	NEW	NEW	NEW
• SLS THRUST, ENGINE (LB)	48,400	30,700	21,800
NUMBER OF MIXED CLASS PASSENGERS	200	200	200
DESIGN RANGE (N MI)	2250	2250	2250
CRUISE MACH NUMBER	0.85	0.85	0.85
MAXIMUM TAKEOFF DISTANCE, SL 84°F (FT)	7300	6860	6550
WING AREA (SQ FT)	2165	2220	2270
MAXIMUM TAKEOFF WEIGHT (LB)	273,300	277,500	267,400
NOISE			
• TAKEOFF (EPNdB)	88.3	92.1	92.1
• SIDELINE (EPNdB)	94.4	94.7	94.4
• APPROACH (EPNdB)	93.4	96.5	96.3
STUDY PRICE (1975 \$ MILLION)	14.89	15.49	15.29
RELATIVE DIRECT OPERATING COST	0	+4.1%	+3.4%

FIGURE 5

The impact of engine price on the direct operating costs of these three configurations is shown in Figure 6. The engine price increases from a minimum for the twin engine configuration reaching a maximum for the four-engine configuration because the cost per pound of thrust decreases with engine size.

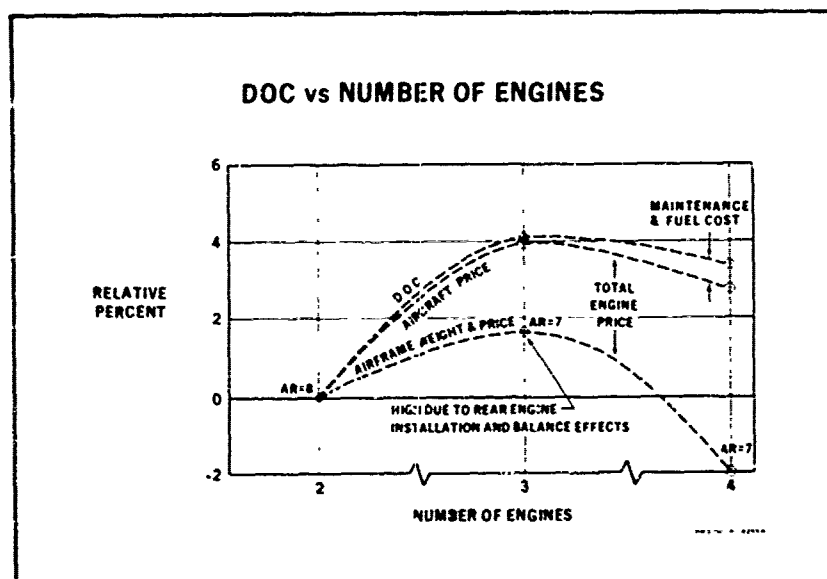


FIGURE 6

As the number of engines increases from the baseline the maintenance costs also increase. The fuel cost increases from two to three engines and decreases slightly for the four-engine case. If the cost of fuel continues to rise this difference can well become greater. The higher maintenance costs for airplanes with a larger number of engines reflect the increased complexity. Also the probability of engine failure is greater as the number of engines increases. This is not to say that airplanes with multiple engines are less safe, only that the probability of a shutdown of a single engine is greater as the number of engines increases.

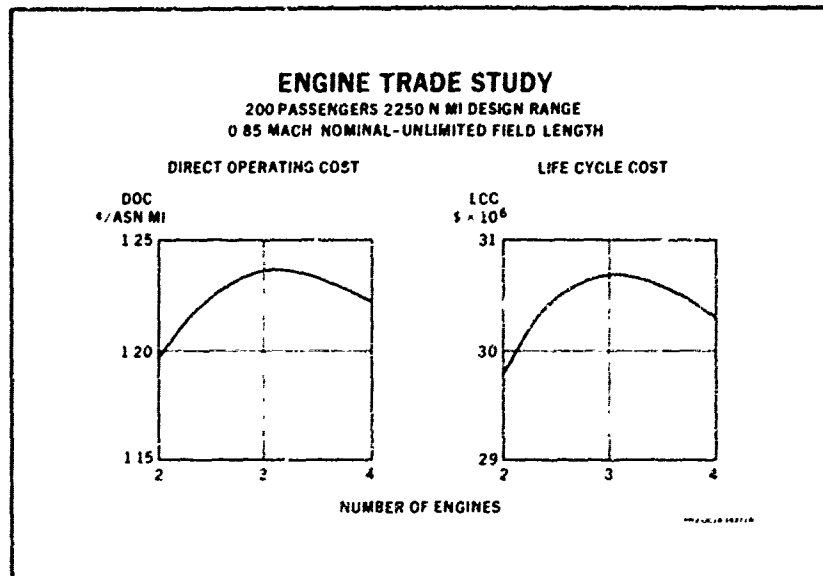


FIGURE 7.

Figure 7 shows the direct and life cycle costs for 2-, 3- and 4-engine airplanes. It should not be assumed solely from the life cycle costs that the two-engine configuration is the proper one to replace the narrow-body twin and tri-jets. Other factors must be considered during Advanced Design, such as engine availability, aircraft market size, range and the development of derivative aircraft. It may be more desirable to use engines developed from those now being used on the wide-body jets rather than commencing a new engine development program just to satisfy this requirement. Figure 8 clearly illustrates the influence of market size on engine costs. An increase in engine market size due to using an available engine can have a significant influence upon both the direct and life cycle airplane costs.

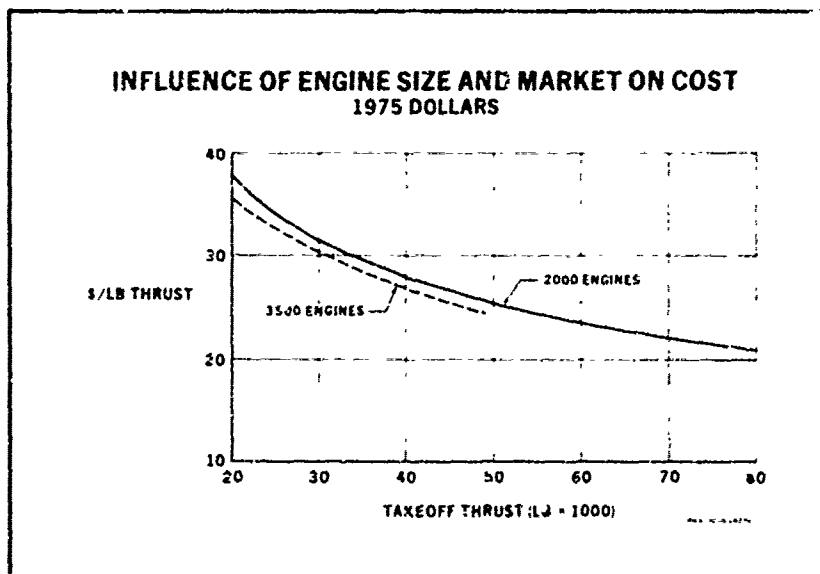


FIGURE 8.

Present U.S. civil regulations restrict two-engine aircraft from making long overwater flights. Thus three- and four-engine aircraft enjoy greater utilization due to their greater range flexibility.

In striving for reduced life cycle costs, Advanced Design must never lose sight of the possibility of developing derivative airplanes, for example, adding or removing an engine, or more commonly stretching the fuselage length. These airplanes are usually the most successful, not only from a life cycle cost standpoint but also from a sales standpoint.

SELECTION OF MATURE EQUIPMENT AND SYSTEMS - An Element of Reduced Life Cycle Cost

The airlines are increasingly concerned about productivity and lower unit operating cost. Expressed in different terms both the military and the commercial airlines require lower life cycle costs.

One way of meeting the challenge of lower life cycle costs is by recognizing the fact that successful airplanes use mature equipment and systems. Maturity can be defined as being fully developed or perfected. In engineering terms, maturity can be translated to mean such things as

- Debugged and service proven
- Accepted and understood by maintenance personnel
- Achieved successful levels of performance, reliability and safety
- Competitive; not outdated or obsolete.

The use of mature equipment and systems with their high degree of reliability reduces development costs for the manufacturer and also maintenance costs for the operator

On the maintenance side it has been estimated that between 20 and 25 percent of all failures are caused by maintenance actions incorrectly performed on the flight line or in repair facilities. This factor combined with the less frequent failure and lower maintenance required by mature equipment and systems will significantly reduce life cycle costs.

STRUCTURAL SIMPLIFICATION – Reduces Life Cycle Costs

The complexity of the aircraft structural system can be reduced. Today's aircraft consist of a myriad of individual parts, much riveting and large numbers of costly machined parts, Figure 9. Emphasis on reducing the number of parts may reduce life cycle costs. A recent Douglas Aircraft Company study of possible gains in structural simplification is of interest, Figure 10. A comparison was made using a typical fuselage section incorporating current wide-body aircraft structural technology as a baseline. Three increasingly more radical methods of reducing the numbers of parts were studied:

- Body Longeron Reduction
- Flattened Longerons
- Isogrid Panels

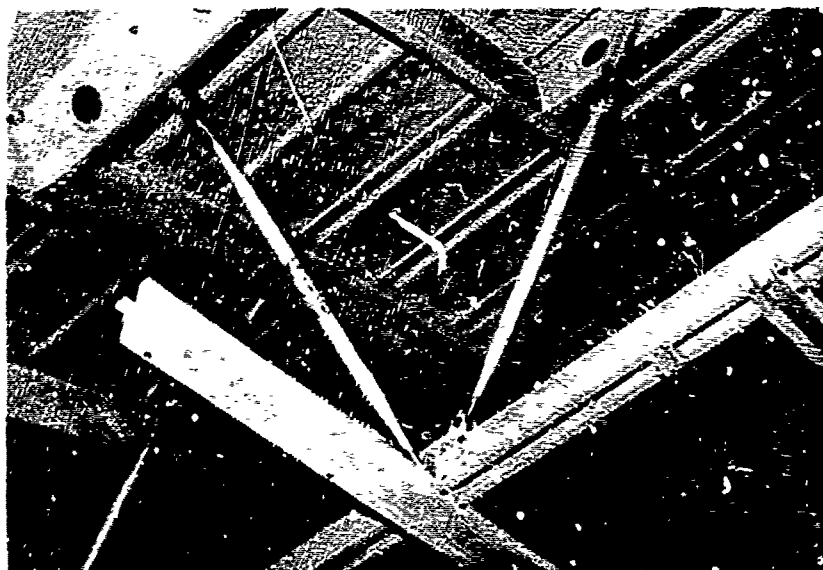


FIGURE 9. CURRENT FUSELAGE STRUCTURE

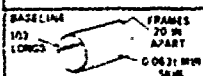

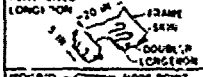

FUSELAGE WEIGHTS, PARTS, COSTS 340 INCH SECTION						
STRUCTURAL CONCEPT	NO. OF SHELL PARTS	RELATIVE COSTS/LB			RELATIVE SECTION WT.	RELATIVE SECTION COST
		MAT'L	LABOR AND DEV.	TOTAL		
BASELINE 	1660	0.03	0.97	1.00	1.00	1.00
LONGERON REDUCTION 	1460	0.03	0.95	0.98	0.88	0.86
FLATTENED LONGERONS 	1270	0.03	0.93	0.96	0.90	0.86
ISOGRID PANELS 	51	0.07	0.65	0.72	0.84	0.61

FIGURE 10.

The use of a reduced number of longerons and of flattened longerons reduced the number of components by about 12 percent and 24 percent, respectively. Both concepts reduce the weight by about 8 percent. A reduced number of parts yields a saving in fabrication and assembly, as well as in maintenance costs.

The use of a new structural technology concept called "isogrid panels," Figure 11, resulted in a radical reduction in parts, and an 11-percent reduction in fuselage weight. Isogrid panels have already been used successfully in space vehicles. It is particularly attractive since all members resist all loads and the shell is the entire structure with the members forming integral longerons and frames. The use of this concept markedly reduces the fuselage wall thickness as well as the overall fuselage diameter for the same inside volume and width. The result is a reduction of both weight and drag.

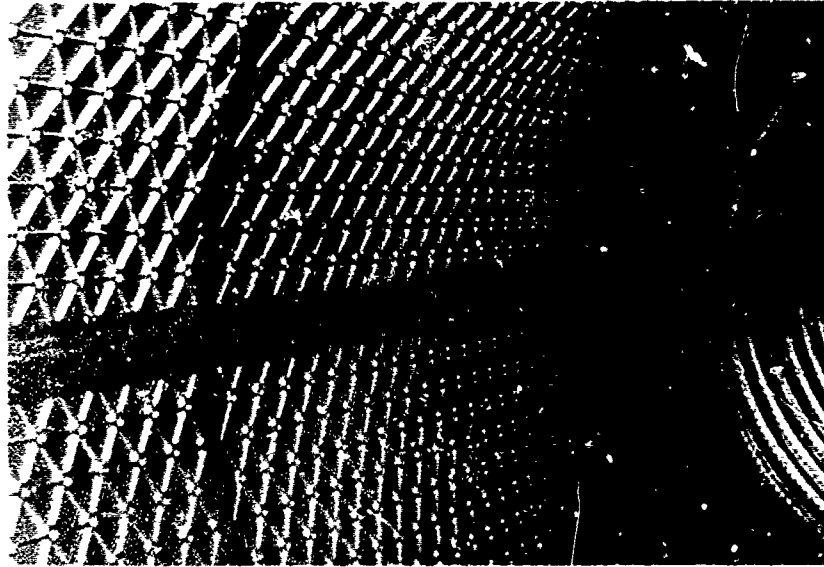


FIGURE 11. 8-FOOT DIAMETER ISOGRID STRUCTURE

High tolerance isogrid structures are machined on numerically controlled equipment, and are readily formed to the desired curvature by brake forming and shot peening. Expensive jigs are not required. These factors combined with reduced manual fitting requirements appreciably reduce manufacturing costs.

MATERIAL SELECTION — A Potential for Reduced Life Cycle Costs

The life cycle costs can be significantly reduced if the structural weight can be reduced without jeopardizing either reliability or maintainability. This can be best achieved by using the material which yields the best combination of the lightest structure and the lowest cost after the design has satisfied all requirements of strength, stiffness, fatigue life, fracture toughness, and corrosion resistance.

This philosophy has always been used in evaluating the most desirable aluminum alloy for each portion of an aircraft. In limited areas of high stress or high temperature, high-strength steel or titanium has proved best in spite of the higher cost per pound.

In recent years the use of composite materials involving graphite or boron fibers of great strength imbedded in an epoxy or aluminum matrix has been much explored. The ratio of strength to density is much greater for these materials than for any of the standard metals.

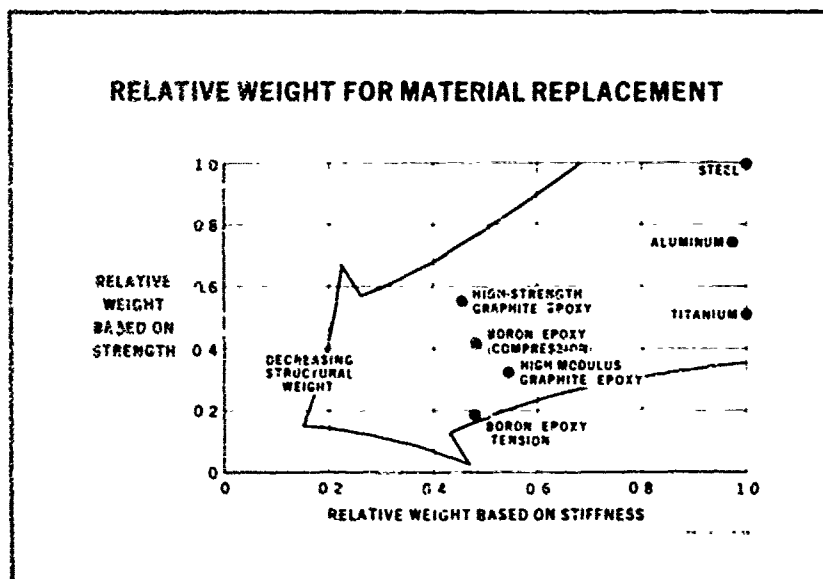


FIGURE 12.

A weight comparison of aircraft structural materials is shown in Figure 12. While important weight saving can be obtained using advanced metal alloys, composites show the greatest potential for significant weight saving for future aircraft designs.

The cost to obtain the reduced weight must be known if minimum life cycle costs are to be obtained. Composites with component weight saving as high as 50 percent can reduce an airplane's empty weight by as much as 15 percent. Although fabrication costs of components made from composite materials may eventually be less than those for metal, this cost saving may be more than offset by material costs. Thus, weight saving, fabrication costs and material costs must be balanced to determine the optimum amount of composite material to be used. Based on Douglas studies the relative importance of weight and composite material cost and composite fabrication costs are shown in Figure 13.

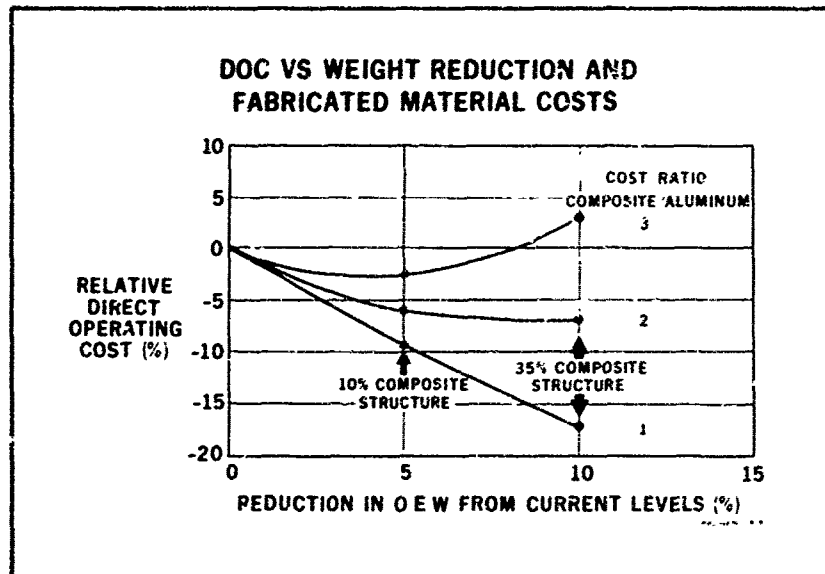


FIGURE 13.

The producers of advanced fibers have made public predictions regarding impressive reductions in the cost forecast for "pre-peg" tape. As the use of this product is increased the cost is expected to drop rapidly. The various in-service demonstration programs will do much to determine the reliability and maintainability of these materials. These programs are, among others, the F-4 rudder, C-5 leading edge flap, F-111 wing trailing edge panels, and the C-141 landing gear doors. None of these test applications has shown any deficiency in application. To date, the production use of composites for the empennage box structure on the F-14 and F-15 has presented no unusual problems. From this experience it can be concluded that many other components could be made from composite materials to save both cost and weight.

The comparison of a wing box utilizing aluminum for the primary structural components with two wing box configurations incorporating composites is of interest, Figure 14. The anticipated near term state-of-technology concepts generally resemble conventional structural arrangements. The details will, however, be quite different in order to fully exploit the properties of composite materials. Further developments in the state-of-composite technology will permit more freedom of design and lead to significant departures from conventional structural configurations. The development of these more radical concepts promises substantial advantages over the more conventional configurations.

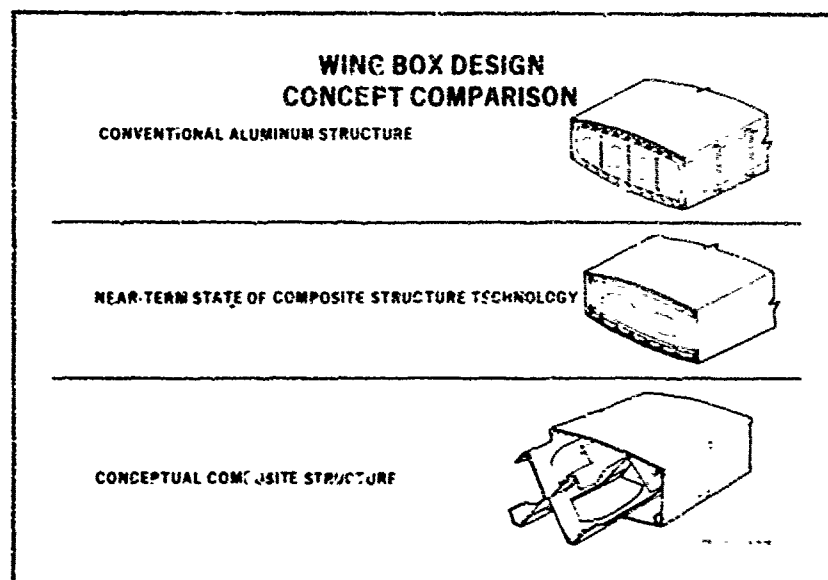


FIGURE 14.

The chief problem in introducing composite materials into primary aircraft structure lies in the need to prove its reliability and maintainability. The basic characteristics of composite structures in day to day airline service with full exposure to the atmosphere, acoustic loads and normal operating loads must be established before extensive use of composites can be expected. A deficiency in the reliability of such structures would quickly eradicate the anticipated economic gains due to weight reduction.

It is an Advanced Design challenge of the first order to determine the proportion of composites to be used to maximize weight saving while minimizing technological risks and life cycle costs for future aircraft. The risks involved when introducing the new structural material are so large that they must be carefully evaluated against the potential gain. Such efforts are fundamental to creative advanced design.

ADVANCED INTERCONTINENTAL AIRCRAFT - Illustrates Benefits of Advanced Composites

A recent Advanced Design study of a large 600-passenger Mach 0.95 intercontinental airplane will serve as an interesting example of the potential reduction in life cycle costs that advanced structural technology can provide. The most efficient interior configuration with the large number of passengers involved, 600, resulted in the distribution of the passengers on two decks, Figure 15. The cargo and passenger baggage are on a third deck below the passengers. This arrangement is the most efficient for very large aircraft because a single deck configuration requires a very long fuselage. This necessitates a long and heavy landing gear to obtain adequate takeoff rotation angles.

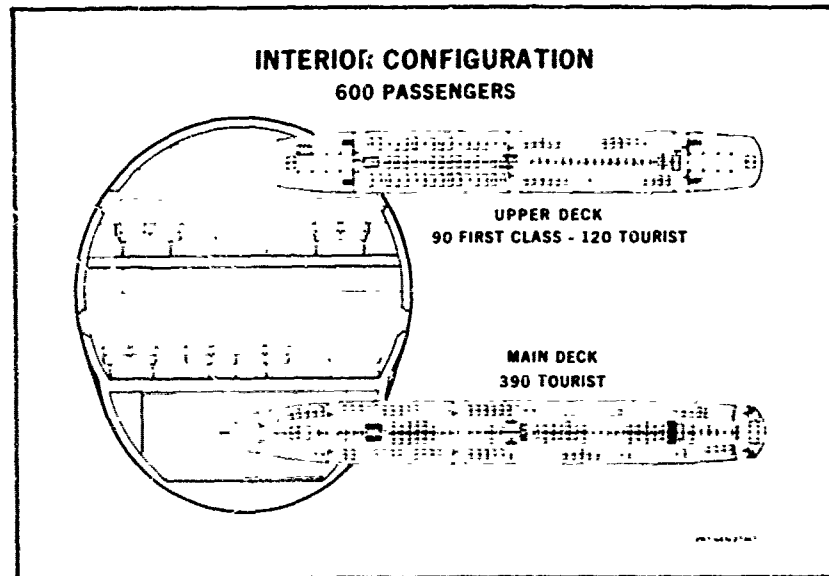


FIGURE 15.

The airplane was configured using three levels of composites. 0 percent, 8 percent and 26 percent of the structural weight plus basic system weight. The reductions of takeoff weight due to the lower weight empty and the reduced fuel required for the mission are dramatic, Figure 16. The area of the community impacted by noise was reduced as the amount of composites used increased, due to an overall reduction in aircraft size.

**ADVANCED INTERCONTINENTAL AIRCRAFT
COMPARATIVE CHARACTERISTICS**

COMPOSITE USAGE % WEIGHT*	0	8	26
ENGINES			
NUMBER AND TYPE	4 NEW	4 NEW	4 NEW
BYPASS RATIO	6	6	6
SLS THRUST ENGINE	105 000	97 000	91 900
NUMBER OF MIXED CLASS PSGR	600	600	600
DESIGN RANGE WITH PSGR AND BAGGAGE	N MI	5250	5250
CRUISE MACH NUMBER	0.95	0.95	0.95
STEP ALTITUDES	FT	31 000 35 000	31 000 35 000
FAR TOFL 54°F FOR DESIGN RANGE	FT.	10 300	10 100
APPROACH SPEED FOR BAGGAGE AND RESERVES	KN	133	133
WING AREA	SQ FT.	8980	8090
OPERATORS EMPTY WT	1000 LB	679	597
MAXIMUM TAKEOFF WT	1000 LB	1 420	1 293
BLOCK TIME AT 5000 N SI RANGE	HR MIN	9-30	9-30
CHANGE IN 100 PNB FOOTPRINT AREA FOR 5000 N MI MISSION WITH 3° GLIDEPATH		0	-9%

*WEIGHT OF STRUCTURE AND BASIC SYSTEMS

FIGURE 16

The reduction in the 100-PNdB footprint area is a very important consideration in any commercial airplane design and has a significant impact on life cycle costs. The reductions in the maximum takeoff weight due to the increased use of composites were found to have a significant impact upon the life cycle costs, Figure 17.

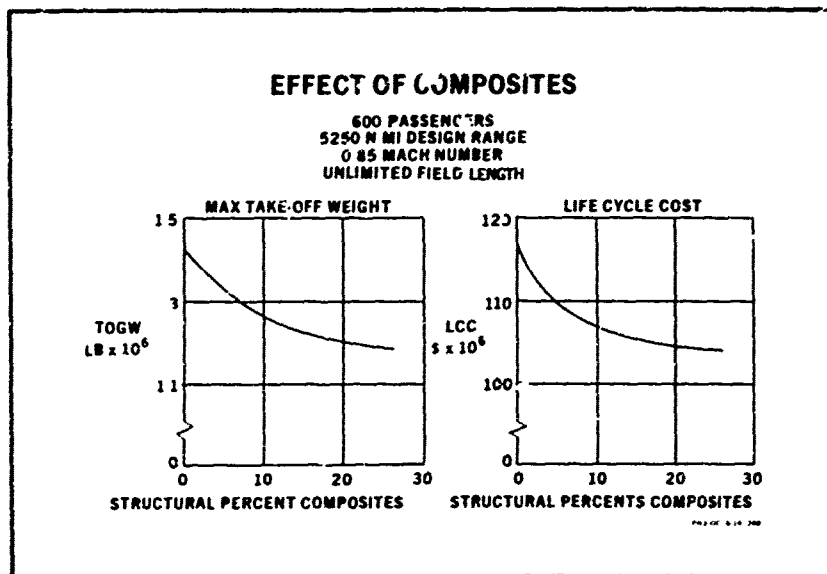


FIGURE 17.

The Douglas advanced international airplane study illustrates the benefits provided by the use of structural materials of higher efficiency. In spite of the increased material costs the overall saving as measured by the life cycle cost benefits is very significant when the amount of composite material is increased.

SYSTEM REDUNDANCY – All-Weather Landing Systems, An Example

Advanced Design studies have proven to be useful in the determination of the costs as well as the resultant potential economic benefits of the Category II and Category III all-weather landing systems. The visibility conditions generally associated with Category II and III are illustrated on Figure 18. The frequency of occurrence of these low visibility conditions is included in a report prepared by R. Dixon Speas Associates, Reference 1.

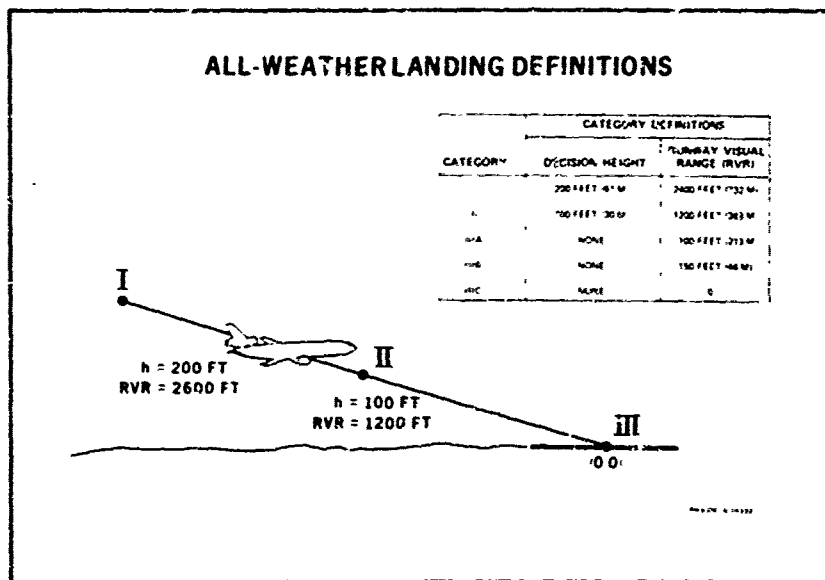


FIGURE 18.

Significantly, one of the world's busiest airports, John F. Kennedy at New York, stands near the top of the U.S. airports with 36.7 hours per year of Category II weather, 15.2 hours per year of Category IIIA and 8.9 hours per year of Category IIIB and IIIC weather.

Category II conditions ranged from 2.6 to 34.3 hours per year at the other airports studied. Category IIIA ranged from 26.6 to 1.7 hours per year while Category IIIB and IIIC weather varied from 27.1 to 0.8 hours per year.

The benefits of the all-weather landing system have been estimated in Reference 1 and are accepted without modification for this paper.

AIRLINE BENEFITS

- Additional Passenger Revenue
- Decreased Fuel Reserves
- Decreased Costs of Weather Interruptions
- Increased Safety

NATIONAL BENEFITS

- Reduction in Delays
- Reduction in Diversions
- Reduction in Cancellations

In incorporating Category III all-weather landing systems in recent aircraft programs, Advanced Design studies indicate that there are significant control system costs over and above those required for the airport and aircraft avionics equipment. Figure 19 outlines the cost per airplane of control system changes needed to provide dual, triple and even quadruple redundancy to assure the safety and reliability of the aircraft fail-operative flight control systems. The development and certification of this equipment in the aircraft adds to the system costs, Table 1. Also included are the aircraft costs for the estimated world fleet of 2665 airplanes equipped for all-weather operation as well as the life cycle costs for this aircraft equipment.

REDUNDANT CONTROL SYSTEMS COST PER AIRCRAFT	
CAT III	
HYDRAULIC SYSTEM	\$ 77,000
ELECTRICAL SYSTEM	2,000
AUTOPILOT	55,000
WIRING	5,000
FABRICATION ASSEMBLY AND CHECKOUT	66,000
TOTAL	\$ 205,000

FIGURE 19.

TABLE 1
COST PER AIRPLANE
1980 Dollars in Thousands

	Category II	Category IIIA	Category IIIB	Category IIIC
Avionics	\$40	\$165	\$205	\$280
Redundant Systems	10	215	215	215
Certification	2.5	20	20	20
Total Investment per Airplane	\$52.5	\$400	\$440	\$515

LIFE CYCLE COSTS PER YEAR
1980 Dollars in Thousands

Aircraft Systems Plus Certification	\$0.8	\$15.7	\$15.7	\$15.7
Operation and Maintenance	8.5	34.5	36.1	39.1
Annual Cost per Year of Additional Systems	\$9.3	\$50.2	\$51.8	\$54.8
Annual Fleet Cost 2665 Aircraft	\$25,600	\$134,000	\$138,000	\$146,000

In the Advanced Design studies it was found that the total costs associated with the all-weather landing systems are significantly higher than those reported. The cost/benefits as a function of the number of airports equipped for the Category II, IIIA, IIIB and IIIC all-weather landing systems are shown by Figure 20. The studies indicate that Category II is cost effective with 16 or more airports in the system. Category III does not become cost effective until all weather systems have been installed on a worldwide basis at more than 50 airports.

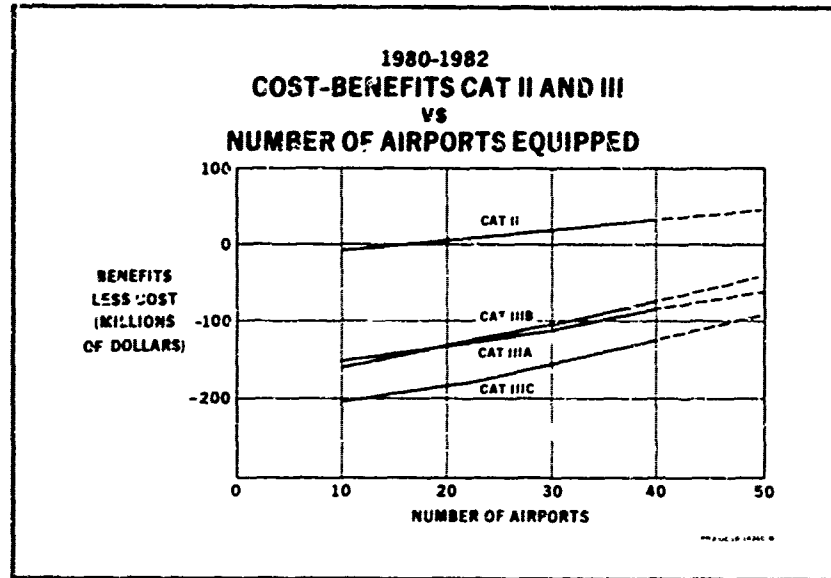


FIGURE 20.

The airport, airplane and system cost benefits for an estimated world fleet of 2665 airplanes equipped for all-weather operations are of interest. Table 2 outlines the airport and airplane costs for three quantities of airports equipped for Category II, IIIA, IIIB and IIIC all-weather landing.

TABLE 2
SUMMARY OF AIRCRAFT AND AIRPORT COSTS
1980 Dollars in Millions/Year
2665 Aircraft

<u>Number of Airports</u>	<u>Category II</u>	<u>Category IIIA</u>	<u>Category IIIB</u>	<u>Category IIIC</u>
10 Airports				
(a) Landside Cost Plus Aircraft Avionics	\$22	\$ 87	\$107	\$146
(b) Aircraft Systems Plus Certification	<u>25</u>	<u>134</u>	<u>138</u>	<u>146</u>
Total	\$47	\$221	\$245	\$292
24 Airports				
(a) Landside Cost Plus Aircraft Avionics	\$25	\$ 93	\$115	\$160
(b) Aircraft Systems Plus Certification	<u>25</u>	<u>134</u>	<u>138</u>	<u>146</u>
Total	\$50	\$227	\$253	\$306
40 Airports				
(a) Landside Cost Plus Aircraft Avionics	\$27	\$ 98	\$120	\$163
(b) Aircraft System Plus Certification	<u>25</u>	<u>134</u>	<u>138</u>	<u>146</u>
Total	\$52	\$232	\$258	\$309

The cost/benefits of Category II, IIIA, IIIB and IIIC landing systems for a fleet of 2665 appropriately equipped airplanes operating from 10, 24 and 40 airports are shown in Table 3.

TABLE 3
SUMMARY OF TOTAL COST/BENEFITS
1980 Dollars in Millions/Year
2665 Aircraft

Number of Airports	Category II	Category IIIA	Category IIIB	Category IIIC
10 Airports				
Benefits	\$ 40	\$ 69	\$ 89	\$ 89
Costs	<u>47</u>	<u>221</u>	<u>245</u>	<u>292</u>
Net	\$-7	\$-152	\$-156	\$-203
24 Airports				
Benefits	\$ 59	\$ 101	\$ 128	\$ 129
Costs	<u>50</u>	<u>227</u>	<u>253</u>	<u>306</u>
Net	\$ 9	\$-126	\$-125	\$-177
40 Airports				
Benefits	\$ 83	\$ 141	\$ 181	\$ 182
Costs	<u>52</u>	<u>232</u>	<u>258</u>	<u>309</u>
Net	\$ 31	\$ -91	\$ -77	\$-127

The role of the advanced designer is to obtain an accurate estimate of the total airplane costs including all of the control system and control surface changes required to implement the fail-operative philosophy required by an all-weather landing system. An example of the increased complexity is shown in Figure 21 which illustrates the necessary changes to the control surfaces and their hydraulic actuators to meet the Category III requirements.

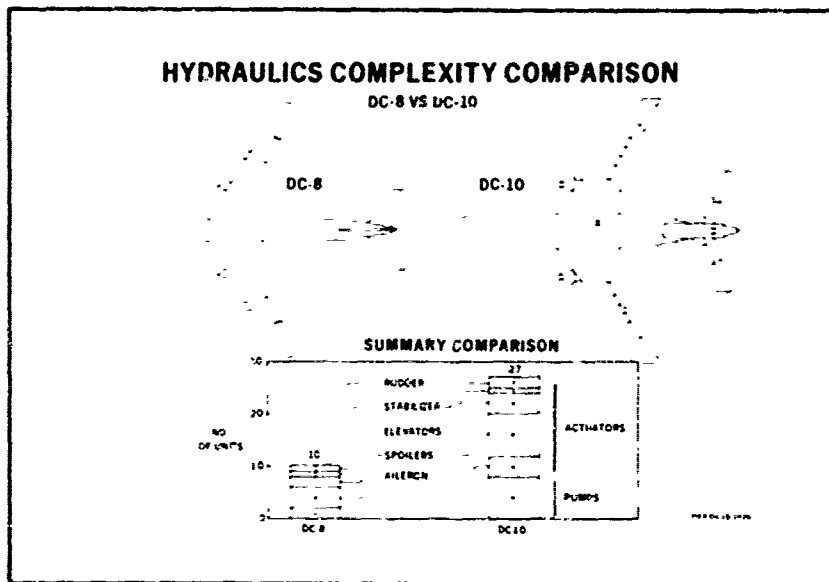


FIGURE 21.

The number of control surfaces is increased primarily by segmenting or dividing the basic surfaces. Each of these surfaces then requires a separate hydraulic actuator which in turn increases the complexity of the hydraulic system as well as the flight control system. It should be noted that predicting the specific cost of complexity itself is very difficult. The changes to the flight control system for Category III are over and above those for the avionics equipment required for Category III and have a significant economic influence on the cost/benefit estimates for the system. These additional costs can be estimated in the preliminary design stage, and may have a strong effect on the ultimate decision to incorporate or not incorporate the capability in the airplane for operating under the various all-weather conditions. However, the improvements in airplane safety and reliability with the redundant flight control systems and the all-weather avionics equipment may be well worth the costs involved even if the economic benefits are less than hoped for.

As previously discussed, the number of airports equipped for all-weather operations has a very significant impact upon the cost/benefits of this capability.

The National Aviation System Plan 1973-1982, Reference 2, indicates that there is now only one airport in the United States equipped with an operational conventional Category IIIA ILS system and that 10 more are planned to be commissioned prior to July 1, 1982 for a total of 11. Starting in 1977, the FAA plans to switch to Category III microwave systems, with a total of 42 systems planned in the United States by July 1982.

The avionics equipment currently installed in aircraft now in production will require modest additions including additional antennas, receivers, and sensors to provide dual capability for using both conventional and microwave ILS. These additional costs were not included in this analysis. The full utilization of the microwave ILS capability which will provide a much wider angular field, variable glide-slopes, and curved approaches among other features will require essentially all new avionics in the aircraft. Again, these costs for this additional capability were not included in this analysis.

CONCLUSIONS

Configurations established in Advanced Design have a strong influence on life cycle costs. In fact, once the airplane design is frozen and emerges from Advanced Design, subsequent changes to reduce life cycle costs are substantially limited. It is therefore vitally important that every consideration be given to life cycle costs as early in the design process as possible. Adequate time must be allotted for an intense evaluation of costs while the program is in Advanced Design.

The example of the all-weather system used in this paper is a good illustration of the importance of including all elements of the system in evaluating life cycle costs. The increased complexity of the flight control system, and consequential changes required in the hydraulic system were as costly as the avionics. The increased safety aspects of the fail-operative redundant flight control system, which are difficult to evaluate in terms of dollars, are generally felt to be worth the extra cost.

Early attention must be given to those factors which have a significant influence on total life cycle costs. These include the number of engines, the design cruise speed, takeoff and landing field length, the approach speed, initial cruise altitude, the selection of equipment and systems, the simplification of design as well as the materials and facilities to be used.

For commercial aircraft the environmental impact of the airplane must be considered on an equal basis with the cost and technical factors. In the real world today, community noise is a most important problem. Low energy consumption per passenger mile may shortly become another important design consideration.

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ANALYSES CRITIQUES ET RECHERCHES EN LABORATOIRE AU STADE DE L'AVANT-PROJET D'UN AVION

par Claude Liévens* et Philippe Poisson-Quinton**

Résumé

A la naissance de tout projet nouveau, les Services Officiels doivent réunir un certain nombre de données techniques permettant le choix d'une configuration bien adaptée à un programme donné ; ces éléments d'appréciation sont demandés à la fois aux bureaux d'études des constructeurs et aux laboratoires de recherches, souvent plusieurs années avant le lancement d'un projet.

Au stade de l'avant-projet, le bureau d'études s'appuie à la fois sur :

- l'expérience acquise au cours des projets antérieurs ;
- des calculs sophistiqués ;
- les résultats d'essais demandés aux laboratoires ;
- un service de documentation bien informé ;
- et sur l'intuition du chef de projet.

De son côté, le Centre de Recherches doit prévoir à temps les orientations majeures de la technique aéronautique pour apporter le maximum d'informations aux services officiels et aux constructeurs ; mais il doit également être capable d'un temps de réponse très court pour satisfaire une demande urgente, soit par ses moyens de calculs et de simulation, soit dans ses laboratoires d'essais.

Cette dernière condition impose des moyens importants en personnel qualifié et en laboratoires sans cesse modernisés ; de tels moyens peuvent paraître coûteux au stade de l'avant-projet, mais l'expérience montre que le bilan coût-efficacité de l'ensemble d'un projet est toujours en faveur d'une étude approfondie au stade de la conception. Elle seule permet de mettre suffisamment tôt en évidence des points critiques dont les conséquences seraient ruineuses si elles apparaissaient au stade du prototype et a fortiori à celui de la production en série.

Ces objectifs à satisfaire dans un Centre de Recherches feront l'objet de la deuxième partie de cet exposé.

Ainsi le rôle des experts techniques des Services Officiels est-il multiple ; il leur appartient :

- de susciter des études préparatoires permettant à l'industrie de présenter un avant-projet valable au moment opportun ;
- de lancer les recherches de base conduisant à une modélisation correcte de phénomènes encore mal connus mais importants ;
- de cerner le risque technique avant le lancement d'un projet et d'éviter les impasses qui risqueraient d'anéantir l'efficacité du système ;
- de suivre attentivement le déroulement des travaux en cours de développement, et éventuellement de les réorienter ;
- de penser aux retombées futures des études, même si celles-ci ne doivent pas être suivies d'une réalisation immédiate.

CRITICAL ANALYSES AND LABORATORY RESEARCH WORK AT THE STAGE OF AIRCRAFT PRELIMINARY DESIGN

Abstract

Whenever a new project is initiated, the government services must collect a number of technical data leading to the choice of a design best suited to a given program. These evaluation factors are frequently requested both from the research laboratories and the constructor design offices, years before the project is initiated.

At the preliminary design stage, the design offices will find themselves equally on :

- experience gained with previous projects,
- sophisticated computations,
- results of the tests ordered from the laboratories,
- a well informed Documentation Center,
- and the intuitive faculties of the project officer.

On the other hand, the Research Center must forecast the main trends of aeronautical techniques in due time, in order to provide the government services and the constructors with a maximum of information. But it should also possess a very short response time capability in order to satisfy urgent requests, either through its computation and simulation facilities or its test laboratories.

The latter requirement calls for large resources, as far as specialized manpower and up to date laboratories are concerned. Such resources may be considered as expensive at the preliminary design stage, but experience has shown that the cost/efficiency of a project benefits in any case from a careful study at the time of its initiation. Only such a study could identify the critical points early enough, whose consequences would be disastrous, should they show up at the prototype, and, a fortiori, at the production stage.

These objectives which a Research Center should meet are described in the second part of this paper.

Therefore, the action of the government technical services is manifold ; they must :

- generate preliminary studies permitting the constructor to produce an acceptable preliminary design in due time,
- start basic research studies leading to a correct modeling of the phenomena that are of importance but still poorly known,
- identify the technical risks before launching a project and avoid deadlocks which may ruin the efficiency of the system,
- watch carefully the progress of the works under development and reorientate them, if needed,
- think of the future fallouts of the studies, even if they were not to end up with immediate production.

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ANALYSE CRITIQUE DES AVANT-PROJETS

par Claude Liévens

1.1 - INTRODUCTION

La méthode utilisée en France pour réaliser un programme d'avion militaire, souvent appelée "approche prototype", est généralement constituée de cinq phases successives relativement distinctes (fig. 1) :

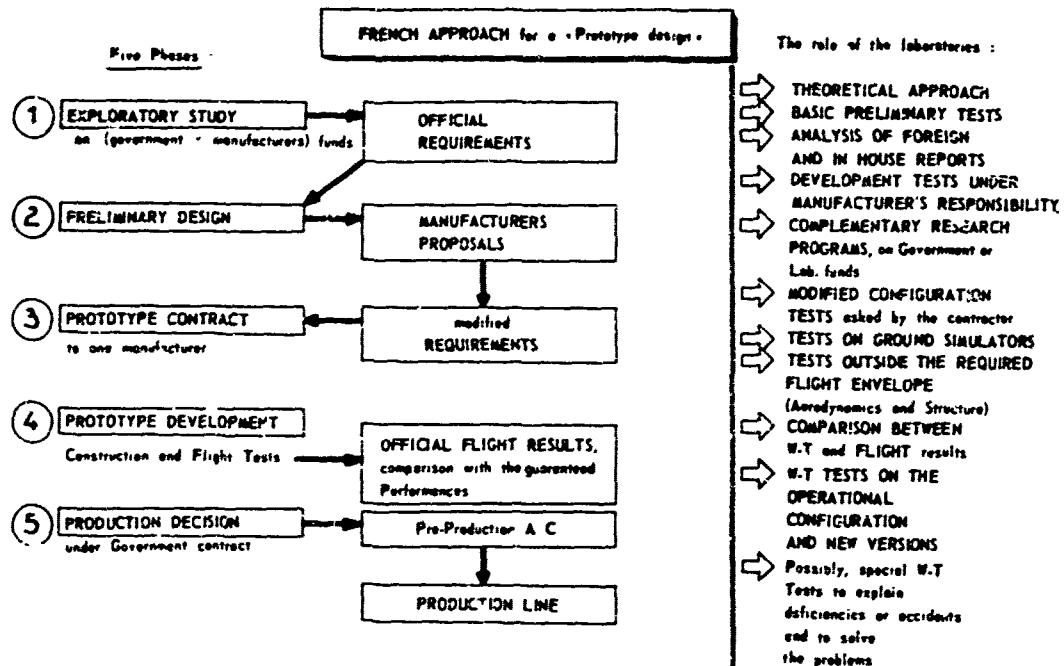


Fig. 1 - Approche « prototype » pour un programme militaire en France.

- les études exploratoires, financées par l'industrie ou par un organisme officiel, permettent d'orienter les investissements, de mettre en évidence des critères de faisabilité et d'inspirer la rédaction d'une éventuelle fiche programme ;
- c'est à la publication de cette "fiche programme" que commence l'avant-projet ; cette phase, qui conduit à des performances et à des coûts estimés, se termine par la remise d'avant-projets détaillés aux services officiels ; ceux-ci doivent vérifier que les affirmations des industriels sont réalistes ou correspondent au moins à un optimisme raisonnable ;
- la préparation du contrat prototype intervient après la sélection de l'avant-projet et se réfère généralement à la publication d'une fiche programme modifiée ou plus détaillée. Le point délicat, dans les négociations entre l'industriel et les services officiels, est d'aboutir à un accord sur les marges permettant de passer des performances estimées aux performances garanties (plus ou moins assorties de pénalités financières). Cette phase aboutit à la notification du contrat prototype ;
- la phase prototype comporte la fabrication du prototype et sa mise au point en vol. Elle aboutit à la réception sur avis des pilotes du Centre d'Essais en Vol. C'est au cours de cette phase que les options aérodynamiques fondamentales sont finalisées en fonction notamment des performances garanties ;
- la phase série s'ouvre à la commande éventuelle de l'avion de série par les services officiels.

Deux caractéristiques de cette approche ne semblent devoir être soulignées : d'une part, l'Etat ne commande la série qu'après avoir jugé la valeur du prototype ; en conséquence, on ne fabrique généralement pas de prototype sur outillage de série. Cette méthode, favorable sur le plan du coût global, a pourtant l'inconvénient d'accroître les délais de remplacement des appareils de la génération précédente. D'autre part, les caractéristiques de l'avion ne sont pas figées au stade de l'avant-projet ; sauf raison impérieuse imposée par les délais de fabrication, les choix ne sont définitifs qu'au cours de la troisième phase.

Pour illustrer cet échelonnement des choix dans le temps, je propose de prendre l'exemple de l'Alphajet (fig. 2). Comme il s'agirait d'un avion d'entraînement, la définition d'un avant-projet donnant une vrille saine constituait l'un des objectifs fondamentaux. Les essais de vrille ont donc été effectués très tôt, antérieurement au choix de l'un des groupes industriels concurrents. La configuration qui était alors qualifiée n'était pas pour autant définitive : il était clairement admis que des évolutions pouvaient être envisagées, à condition qu'elles ne détériorent pas les caractéristiques de vrille. C'est ainsi que des calculs théoriques, confirmés par la soufflerie, ont conduit à changer le dièdre à modifier les profils de voilure, à supprimer les spoilers. Le fuselage et le caisson de voilure ont été figés relativement tard, juste avant le lancement de la fabrication. Les bacs et les volets ont été choisis encore plus tardivement ; encore subsiste-t-il pour la première phase d'essais en vol une solution de remplacement pouvant conduire à un comportement encore meilleur de l'avion au décrochage. Bien entendu, cet échelonnement dans le temps est intimement lié à la durée des cycles de fabrication ; pour un autre programme conduisant à un caisson de voilure en titane, la définition de celui-ci servirait à figer dans la toute première partie de la phase prototype.

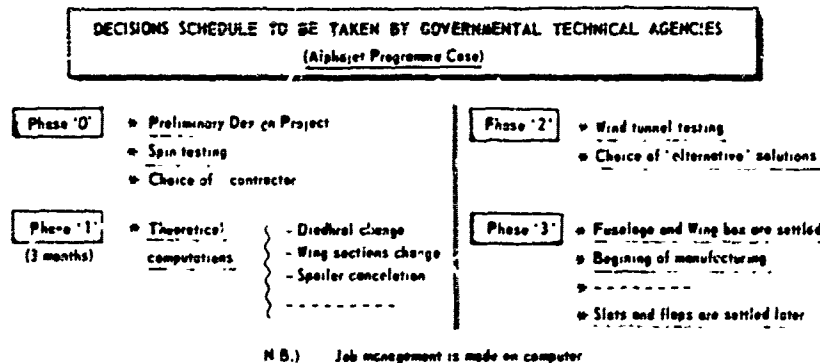


Fig. 2 - Exemples d'échelonnement des choix dans le temps pour l'Alphajet.

Il est clair que l'existence des définitions successives de l'avion pose un problème de gestion des données et exige un travail extrêmement rigoureux sur ordinateur. La discipline requise est d'autant plus stricte que beaucoup de programmes sont menés en coopération, les contractants et leurs usines étant géographiquement très dispersés. L'existence d'un centre de données constamment mises à jour et utilisées par toutes les équipes concernées, pour la conception et la fabrication du matériel, constitue aujourd'hui une nécessité de la première importance.

L'application de la méthode française a été facilitée par l'absence relative, et dommageable par ailleurs, de véritable compétition en matière d'avions militaires. L'industriel n'est pas tenté d'afficher des performances démesurément optimistes pour éliminer les propositions concurrentes. Au stade de l'avant-projet, les services officiels ne demandent pas que les dossiers définissent les moindres détails de l'avion. La documentation à ce stade est peu volumineuse, la définition de l'appareil peut encore évoluer avec beaucoup de souplesse, en fonction des difficultés de réalisation, des délais, des coûts de développement et des risques estimés.

Il peut arriver que l'ordre des diverses opérations ne soit pas aussi clairement respecté ; c'est notamment le cas des avions civils où les besoins des compagnies ne peuvent être traduits en un simple programme comme pour l'aviation militaire, et où les prototypes peuvent être construits sur outillages de série. Néanmoins, il y a au stade exploratoire et lors de l'avant-projet de grandes analogies, comme le montre la figure 3, consacrée au lancement d'un avant-projet civil (STOL).

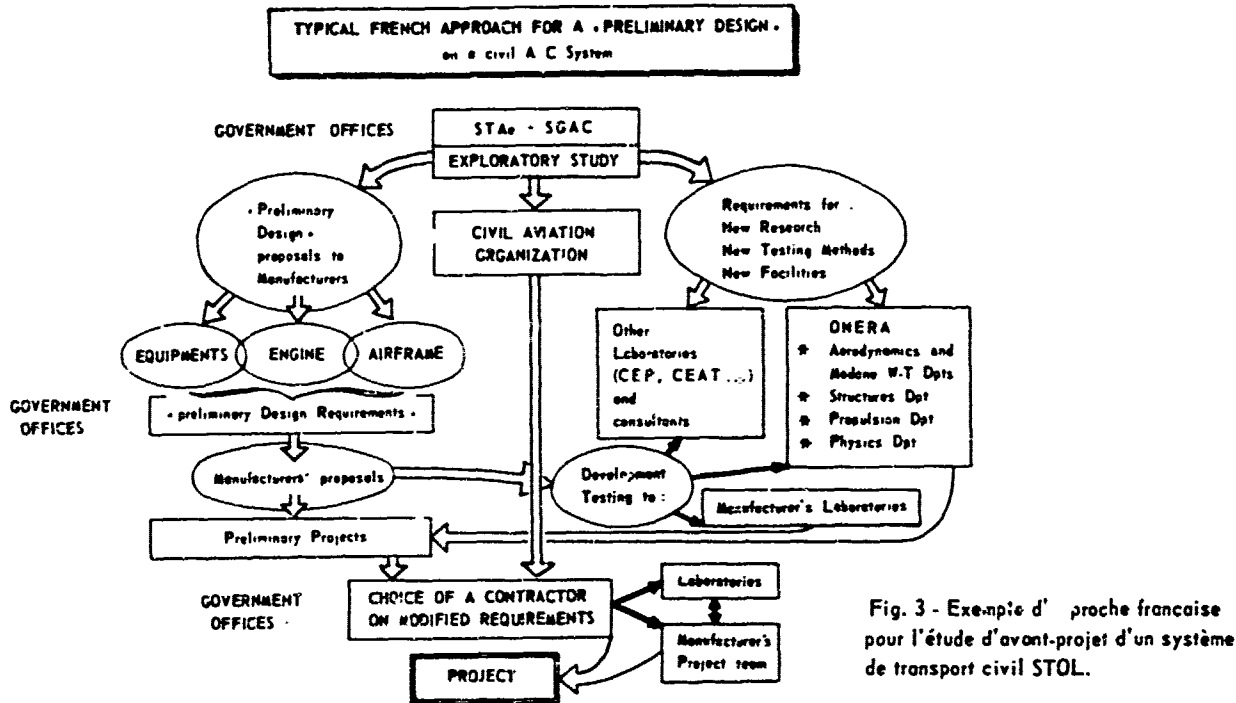


Fig. 3 - Exemple d'approche française pour l'étude d'avant-projet d'un système de transport civil STOL.

Je voudrais souligner que pour les formules totalement nouvelles, qui comportent de gros aléas de développement, il n'est pas souhaitable d'entrer directement dans une séquence dont la production en série constituerait l'aboutissement. Il faut alors passer par l'intermédiaire d'un modèle expérimental, partie intégrante des études exploratoires. Je souligne en particulier les dangers encourus si les Etats Majors laissent des modes, des impulsions passagères ou des enthousiasmes excessifs influencer l'expression de leurs besoins, car on pourrait alors connaître des errements funestes : dans une première phase, la formule nouvelle -comme toute recherche dont la perspective demeure lointaine- ferait l'objet de recherches modestes, d'un montant insuffisant pour résoudre convenablement les problèmes de faisabilité ; puis l'Etat Major trop enthousiaste se persuaderait soudain de l'intérêt incomparable du nouveau concept et passerait immédiatement à la rédaction d'une fiche programme d'avion opérationnel. Dès lors, le malheureux programme cumulerait les difficultés que l'on rencontre dans tout développement et celles propres à la nouvelle formule, et ceci dans un climat peu favorable à la réflexion. C'est à des circonstances de ce genre qu'est imputable l'échec relatif du programme français de décollage et

atterrissage vertical. La définition d'un modèle expérimental est d'ailleurs une tâche difficile, car si un appareil à prétentions opérationnelles est vraisemblablement voué à l'échec, de même, une expérimentation sur un thème manifestement irréaliste ne prouverait rien : avantages et inconvénients ne sauraient être comparés de manière significative.

1.2 - L'EXAMEN CRITIQUE AU STADE DE L'AVANT-PROJET

Nous sommes donc maintenant au stade où, basés sur des essais effectués dans les laboratoires et sur les études propres des industriels, les avant-projets sont soumis aux services officiels (fig. 4) ; même en cas d'absence relative de concurrence, l'analyse critique technique de ces avant-projets constitue une opération importante. Il s'agit en effet de savoir si le lancement d'une phase prototype suppose ou non un pari technique démesuré. Comprendons bien qu'il serait absurde de se porter garant de toutes les performances estimées : un projet sans pari donnerait un avion sans avenir. Mais nous devons dire si ces paris demeurent raisonnables ; et, dans la mesure où nous sommes moins influencés que les autres protagonistes par l'état des plans de charge et l'urgence des besoins, nous devons pouvoir faire ce diagnostic mieux que quiconque. En résumé, nous sommes tenus de donner sans ambiguïté des informations techniques à des décideurs, qui ne sont généralement pas des techniciens, et qui doivent intégrer à leur choix une grande quantité d'éléments non techniques.

Cette tâche pose le double problème des critères de jugement et des techniques d'analyse.

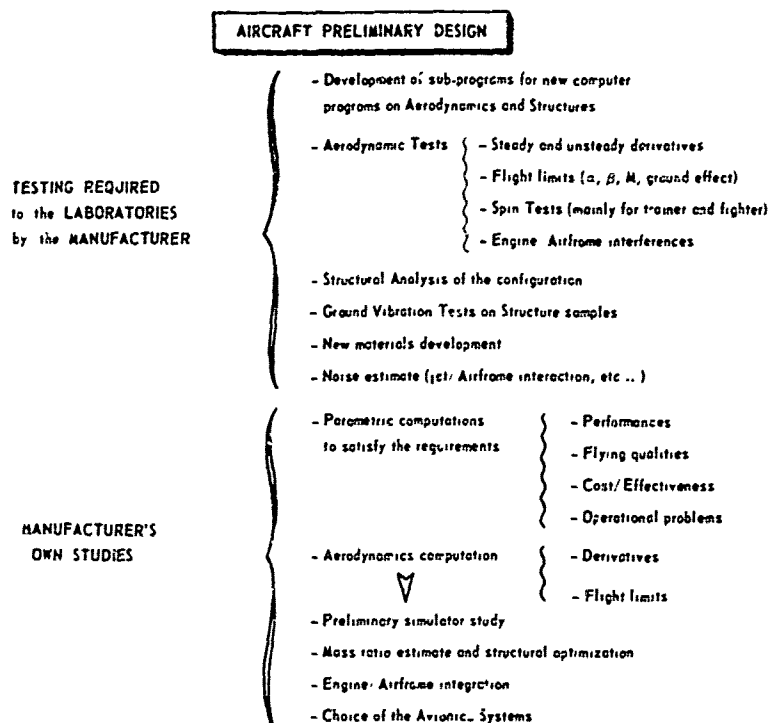


Fig. 4 - Etudes à entreprendre au stade de l'avant-projet.

1.2.1 - Les critères de jugement (voir Ann. 5)

On peut évidemment demander une bonne conception aérodynamique ; mais celle-ci n'est pas un but en soi ; elle est au service des performances et des qualités de vol, qui dépendent aussi des systèmes de propulsion disponibles, de la poussée et de la consommation spécifique qu'on peut en attendre, de leur potentiel de développement ultérieur. Il faut faire intervenir le devis de masse, donc les solutions adoptées au point de vue structural ; de plus en plus, il faudra tenir compte, dès l'avant-projet, des systèmes d'aide au pilotage, de stabilisateur, etc., initialement utilisés pour corriger les défauts d'appareils existants, mais destinés, dans une optique d'intégration et de CCV, à bouleverser nos instruments d'étude traditionnels des qualités de vol. Aérodynamique, propulsion, structure, masses et systèmes conditionnent donc les performances et les qualités de vol des avions et doivent, à ce titre, être examinés au stade de l'avant-projet.

Pour ce qui est des appareils militaires, les objectifs de performances correspondent aux missions types figurant sur la fiche programme ; à cet égard, l'analyse critique de l'avant-projet se base sur des études de sensibilité des performances aux variations "prévisibles ou vraisemblables" des paramètres aérodynamiques et propulsifs. L'objectif de qualités de vol, à long terme, est de satisfaire les pilotes du Centre d'Essais en Vol et de l'Armée de l'Air ; au stade de l'avant-projet, nous utilisons des critères quantitatifs voisins de ceux figurant dans la norme américaine MIL-F-8785 B ; mais ces critères ne constituent que des guides, sans jamais être érigés en exigences. En ce qui concerne les appareils civils, les objectifs de performances correspondent à des missions types, expression plus ou moins directe et plus ou moins aléatoire des besoins des clients potentiels ; les objectifs de qualités de vol correspondent aux exigences des conditions techniques de navigabilité et à la satisfaction des pilotes des services officiels et des compagnies exploitantes.

Mais on ne peut se prononcer sur le niveau d'optimisation d'un avant-projet sans se référer à un objectif de coût opérationnel ; en plus des données de performances, ceci exige une première analyse de

fiabilité et d'aptitude à la maintenance, débouchant sur le coût des réparations et des opérations de maintenance préventive.

D'autres éléments interviennent comme contraintes ; ce sont d'abord des conditions opérationnelles, ce sont ensuite les problèmes d'environnement apparus récemment : limitation du bruit au décollage et en approche, et à plus long terme, limitations sur les diverses formes de pollution atmosphérique. Toutes ces contraintes, dont l'évolution prochaine est difficile à prévoir, pèseront désormais sur tous les projets d'avions civils, et peut-être militaires.

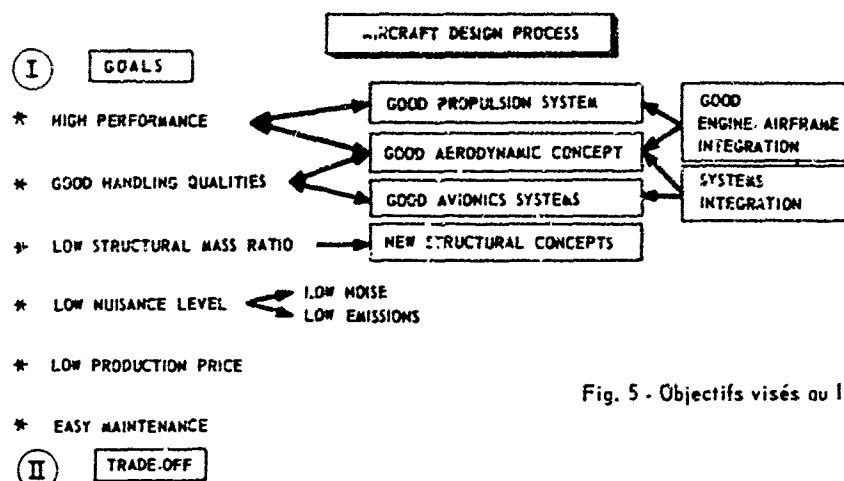


Fig. 5 - Objectifs visés au lancement d'un projet.

L'avant-projet apparaît en tout état de cause comme un compromis entre une multitude de critères, d'où le problème de coordination entre toutes les équipes impliquées. Une première méthode consisterait à donner, chez l'industriel et à chaque équipe concernée, un objectif partiel compatible avec les objectifs globaux et l'ensemble des contraintes ; dans cette perspective, les spécialistes interviendraient très tôt. L'industrie française a préféré une deuxième voie : réduire au début le rôle des spécialistes, laisser le choix des grandes options techniques au chef de projet et à son équipe. Cela exige, chez l'industriel, des ingénieurs de haut niveau, ayant une vaste culture dans tous les domaines de l'aéronautique ; et, dans les services officiels, des généralistes suffisamment éclairés.

Après avoir parlé des critères et des objectifs, il nous faut maintenant mettre l'accent sur le problème de la flexibilité de ceux-ci. Prenons à cet égard quelques exemples :

- pour un appareil commercial à grande capacité, l'incapacité à effectuer une mission de référence ne conduit pas à une condamnation inéluctable. En pratique, l'une des missions les plus critiques envisagées par un éventuel client pourrait ne pas être accomplie dans les meilleures conditions de rentabilité ; on décollerait par exemple de Mexico avec un ou deux passagers de moins dans des conditions climatiques défavorables ! Ceci pourrait compromettre quelques commandes sans porter un coup fatal au programme ;
- imaginez maintenant que Concorde ne puisse faire Paris-New York avec la charge marchande requise : le programme serait vraisemblablement condamné. Heureusement, cette éventualité est extrêmement improbable, ce qui est d'ailleurs miraculeux, dans la mesure où le projet a abordé sa phase de développement avec un objectif de moyen courrier !
- le problème du bruit peut dès maintenant introduire des limitations assez sévères pour compromettre un programme civil, et il est d'autant plus difficile à prendre en considération que la lutte contre le bruit coûte cher et qu'on ne connaît guère le niveau qui sera toléré dans dix ans. Or, on ne peut par exemple lancer un programme STOL sans faire un pari sur cette évolution, et ce pari conditionne largement le choix de la formule et certains caractéristiques du projet. Le bruit au décollage de Concorde constitue aussi un autre exemple frappant : c'est aujourd'hui un problème, alors qu'au moment du lancement de la phase prototype ni l'opinion publique ni les spécialistes n'y étaient vraiment sensibilisés. Sans doute pensait-on alors que les terrains d'aviation pouvaient être construits hors des zones urbanisées sans attirer aussitôt les appétits de la promotion immobilière et susciter la prolifération des immeubles et des lotissements.

L'estimation de la flexibilité des objectifs constitue une phase importante dans le travail des services officiels et de l'industrie. En cas de non flexibilité, il importe de prendre des marges, d'autant plus élevées que les aléas techniques sont importants, et qui éloignent le compromis choisi de ce que serait un optimum mathématiquement exprimé.

1.2.2 - Les techniques d'analyse

J'en arrive maintenant aux techniques utilisées dans l'analyse critique des avant-projets. Ces techniques varient suivant que ceux-ci sont définis :

- à partir d'études paramétriques ;
- à partir de l'expérience antérieure ;
- à partir de calculs sophistiqués ;
- à partir de résultats d'essais ;
- ou à partir de l'intuition, (fig. 6).

THE KEY FACTORS IN AERODYNAMICS FOR A GOOD PRELIMINARY DESIGN

ACTIONS	from Manufacturers :	from Official Agencies :
• Parametric studies.		* Criticism on the processes.
• Previous experiences		* Good knowledge of previous projects, <u>including other competitors' studies.</u>
• Sophisticated computations.		* Criticism on the hypothesis, and on the 'modélisations'.
* Testing		* Extrapolation Validity, taking account of the flow state.
* Intuition		* Intuition
* or last	THE BEST PRELIMINARY DESIGN STUDY is made with a good	



Fig. 5 - Les facteurs aérodynamiques conduisant à un avant-projet satisfaisant.

1.2.2.1 - Etudes paramétriques

Dans l'abstrait, des méthodes mathématiques utilisant l'ordinateur devraient permettre à elles seules d'optimiser complètement un projet à partir des objectifs et contraintes déjà évoqués. De telles études ainsi comprises ont généralement assez peu d'audience en France et ne paraissent dangereuses (fig. 7) :

- elles sont généralement très sensibles, et il suffit de légères modifications dans les hypothèses, dans les missions de base ou dans la pondération des critères pour aboutir à des solutions totalement différentes. Ce fait est apparu très clairement dans certaines communications faites au congrès AGARD sur les V/STOL (Bruxelles, 1968) ;
- le choix des marges tenant compte des aléas techniques et de la flexibilité des objectifs ne peut pas recevoir de solution purement mathématique ;
- dans l'évolution d'un programme se produisent des événements qu'il appartient au chef de projet de pressentir, sans que cela puisse donner lieu à une formulation mathématique générale ;
- enfin, et surtout, les hypothèses sur lesquelles reposent les études paramétriques ne sont valables que dans des domaines très restreints. L'optimum mathématique a toutes les chances de se situer dans le domaine marginal de l'une au moins des hypothèses, et de ne pas correspondre à un avant-projet réaliste.

GLOBAL OPTIMIZATION THROUGH COMPUTER ALONE	CONCEPTION WITH PARTIAL USE OF COMPUTER
UNSUCCESSFULL	SUCCESSFULL
<ul style="list-style-type: none"> • Results too sensitive to input data • No possible modelling of technical contingencies. • Experiments difficult to incorporate. • Assumptions valid only within a too confined area 	<ul style="list-style-type: none"> • Rigorous management and updating of projects characteristics. • Sophisticated computation required to master certain problems.

Fig. 7 - Deux approches pour une étude paramétrique.

Bien entendu, les critiques faites à l'encontre d'une optimisation globale, purement mathématique, faite par ordinateur ne condamnent pas la conception aidée par ordinateur. Celle-ci constitue un outillage absolument indispensable (voir § 1.2.3).

Plutôt que de faire des études mathématiques, nous préférons balayer les domaines envisageables en dessinant plusieurs avant-projets, en les évaluant séparément et en comparant les résultats obtenus. On pourrait, à titre d'exemple, prendre l'optimisation d'une surface de voilure d'avion de combat à formule aérodynamique donnée (voilure et empennage horizontal fixés en forme et en position relative, à une homothétie près). Ce problème, simple au point de vue du concept, ne peut être étudié mathématiquement : très vite, compte tenu de l'architecture générale de l'avion, de la quantité de carburant, de la fixation des charges externes, etc. on s'aperçoit que l'effet d'échelle interdit l'emploi de la même formule aérodynamique... Nous sommes alors conduits à faire plusieurs avant-projets dans les gammes de surfaces envisageables.

L'analyse critique des études paramétriques revient alors, pour l'ingénieur des services officiels, à l'étude de plusieurs avant-projets.

1.2.2.2. L'expérimentation antérieure

Dans toute la mesure du possible, nous cherchons à n'avancer que par petits pas. L'imagination, certes, est nécessaire pour concevoir un avant-projet, mais il faut la concentrer sur l'essentiel : nous ne croyons pas au succès d'un programme basé sur une multiplicité de petits gadgets astucieux, de mise au point difficile et faisant appel à des techniques mal connues.

Je crois que le succès de l'aéronautique militaire française est liée au fait que chaque avion bénéficie directement de l'expérience acquise par ses prédécesseurs. Regardez la figure 8 : les quatre avions, bien que très différents au point de vue aérodynamique - il y a une voilure delta, une aile en flèche, une géométrie variable et un VTOL - ont un air de famille. Les prises d'air sont sensiblement les mêmes. Avant la géométrie variable, il y avait eu le Mirage F2 et la position de l'empennage horizontal n'a guère changé. Tous les avions qui ont suivi ceux représentés ici - Mirage F1, Mirage G8, avant-projets d'avion de combat futur - sont dans la ligne directe de l'un des quatre appareils. D'ailleurs, le fait que la même équipe chargée des études avancées aux Avions Marcel-Dassault ait suivi, au cours des dix dernières années, une bonne douzaine de prototypes depuis l'étude exploratoire jusqu'à la mise au point en vol constitue un capital d'expérience extrêmement précieux.

Du côté des ingénieurs des services officiels, il faut donc, pour faire l'analyse critique des avant-projets, une bonne connaissance des programmes antérieurs et des problèmes rencontrés jusqu'à leur mise au point en vol. Lorsque plusieurs industriels sont mis en compétition au stade de l'avant-projet, les services officiels bénéficient aussi de la connaissance des avant-projets issus d'industriels différents, ce qui leur permet de mettre mieux en évidence les points critiques de chacun d'entre eux. Autant l'absence relative de compétition nous permettait une plus grande souplesse dans l'échelonnement des choix techniques, autant elle est préjudiciable au stade de l'analyse critique. Au cours des cinq dernières années, nous avons néanmoins pu bénéficier d'une situation concurrentielle :

- dans la définition de l'avion d'entraînement qui est devenu l'Alphajet franco-allemand ;
- dans les avant-projets STOL commandés par le SGAC.

D'autres projets importants (Concorde, Airbus) avaient fait eux aussi l'objet d'une concurrence entre groupes industriels.

* M=2.2 multissions
 * All concepts tailored for
 LO-LO 300 NM radius mission

	①	②	③	④
WEIGHT (mission)				
Empty weight (empty)	7.9 T	8.8 T	9 T	10.5 T
Fuel	3.5	3.7	3.6	5.2
Military load (bomb, mun, etc)	1.8	1.9	1.8	1.8
Take-off weight	12.9 T	14.0 T	14.7 T	17.5 T
CHARACTERISTICS				
Wing area (reference)	37 m ²	25 m ²	27 m ²	40 m ²
Wing loading	125 kg/m ²	130 kg/m ²	130 kg/m ²	165 kg/m ²
Wing loading w/s	350	360	370	440
Thrust to weight ratio M=1	0.80	0.76	0.73	0.59
MISSION: CRO PENETRATION				
0.500 (altitude)	300	270	360	300
Cost sensitivity at 300 m/s	2.4	2.4	2.9	2.7
Manoeuvrability at 300 m/s (1/3g)	4g	3g	3g	3g
MISSION: INTERCEPTION				
Climb time to M=2.2 at 3000 m	43 s	48 s	46 s	107 s
Turned time to M=2.2 at 7000 m	72 s	57 s	51 s	23 s
Capture to M=2.2 at 3000 m	13.5 s	12.2 s	12.7 s	6.7 s
PERFORMANCE				
Take-off run (150-150%)	770 m	700 m	500 m	0
Turned to clear 50 ft (150-150%)	1250 m	1100 m	800 m	0
Maximum speed (150-150%)	1800 m/s	1700 m/s	1700 m/s	1200 m/s
Max. altitude (150-150%)	13000 m	12000 m	12000 m	10000 m

Fig. 8 - Quatre conceptions d'un avion de combat adapté à une mission de pénétration à basse altitude sur un objectif à X = 300 NM, avec le même réacteur de croisière (TF-306, T = 10,4 tonnes avec P.C.).

4 Dassault «Mirage» combat A/C

2.2.3 - Les méthodes de calcul sophistiquées

Dans les paragraphes suivants, afin de raisonner sur des cas concrets, je citerai quelques exemples empruntés au domaine de l'aérodynamique. Il est clair que les autres disciplines offrent des situations analogues... Les méthodes traditionnelles de conception des avions, qui constituaient il y a dix ou quinze ans un édifice assez cohérent, ont été ébranlées depuis par plusieurs événements :

- la nécessité, pour les avions volant à haut subsonique, d'acquiescer leur nombre de Mach de croisière a conduit à l'abandon des profils traditionnels. Que l'on crée des répartitions de vitesses "en plateau", conduisant à des nombres de Mach légèrement supérieurs à un nombre critique dans une partie notable de la corde, ou que l'on développe une région supersonique suivie d'une recompression sans choc ou avec un choc d'intensité assez faible pour qu'il n'introduise aucun décollement, dans les deux cas, le travail ne peut se faire qu'en disposant de méthodes sophistiquées. Il commence à en être de même pour d'autres problèmes aérodynamiques (optimisation de dispositifs hypersustentateurs, par exemple) ;
- l'incursion dans le subsonique élevé exclut que les problèmes d'interaction soient traités in fine après optimisation partielle des différents éléments de l'avion. Et on ne peut plus se contenter d'approximations grossières du type "loi des aires". Les moyens de calcul doivent se référer à une base de données prenant en compte la totalité de l'appareil, avec toutes les difficultés que représente le stockage des informations nécessaires à une représentation correcte de l'avion dans ses diverses configurations, hypersustentées ou non... Il y aurait aussi la nécessité de "boucler" sur les déformations aéroélastiques mais il faut bien reconnaître que l'on demeure peu servait en ce domaine ;
- le volume des essais en soufflerie effectués sur les avions des générations récentes s'est constamment accru. Leur difficulté technique, la précision que l'on exige des résultats, la nature plus rigoureuse des problèmes inhérents aux formules nouvelles en augmentant le coût ; cette inflation est encore accrue par le jeu de méthodes de fabrication discutables, discutées et très éloignées de l'optimum au plan national. En outre, le nombre et la qualité des souffleries disponibles n'augmentent pas proportionnellement aux besoins. Dès lors, il devient nécessaire de limiter le nombre des solutions à essayer en soufflerie et de réduire les périodes de mise au point en cours de campagne, donc d'aboutir rapidement à des solutions presque optimales. C'est le rôle du calcul de nous permettre de mener à bien cette démarche.

On ne peut certes détailler ici les éléments qui entrent dans les gros programmes de calcul ; concernant l'exemple de l'aérodynamique, je dirai - à mon point de vue - que les méthodes utilisées se répartissent en trois groupes (fig. 9) :

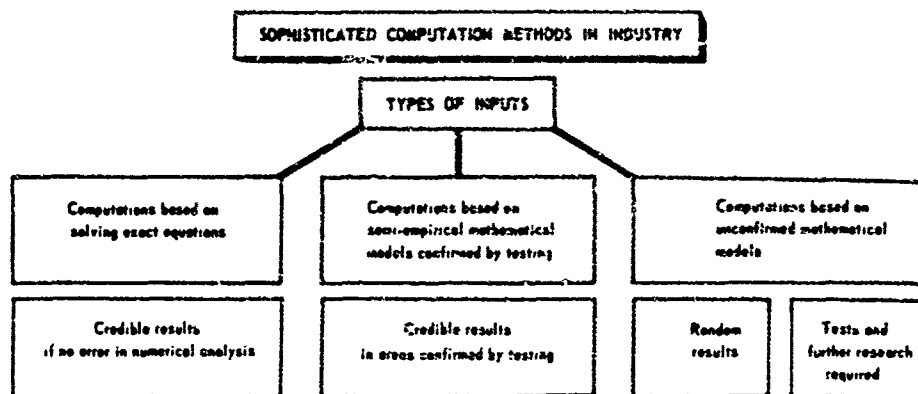


Fig. 9 - Méthodes de calculs sophistiqués utilisées dans l'industrie.

- quelques-unes sont très pures sur le plan théorique et consistent à résoudre les équations de la mécanique des fluides : on arrive à un problème d'analyse numérique ;
- d'autres, dans l'impossibilité théorique ou pratique d'une démarche rigoureuse, font appel à des données semi-empiriques, mais confirmées par de nombreuses expériences disponibles ;
- d'autres, enfin, en l'absence de données théoriques ou expérimentales suffisantes, font appel à une modélisation grossière, valable en première approximation et à défaut d'autre chose.

Il est clair que de tels programmes de calcul, compte tenu de leur caractère semi-empirique, plus efficace que rigoureux, et de la complexité d'introduction des caractéristiques géométriques des différentes configurations, constituent typiquement un outil d'industriel. Je vais préciser ce que doivent être, dans ces conditions, le rôle des services officiels et le rôle des laboratoires de recherche.

- Il n'est pas pensable que les spécialistes chargés de l'analyse critique des avant-projets dans les services officiels développent des méthodes de calcul sophistiquées concurrençant celles de l'industrie. Bien sûr, il leur faut disposer de "recettes" et de points de recoupements pour déceler de grosses erreurs éventuelles sur les données aérodynamiques ; et c'est la connaissance des projets antérieurs qui fournira le plus souvent la meilleure base. Mais surtout, il faut que les ingénieurs des services officiels sachent toujours très exactement où en sont les programmes de calcul de l'industrie, quels sont les phénomènes convenablement modélisés (compte tenu de recoupements expérimentaux) et donnant lieu à des résultats crédibles, quels sont au contraire les phénomènes mal connus dont la modélisation est aléatoire. Il faut qu'ils puissent juger l'impact de telles incertitudes sur les performances du projet. Je ferai remarquer que le dialogue avec l'industrie sera d'autant plus franc, et l'efficacité des services officiels d'autant plus élevée que ces mêmes ingénieurs auront un pouvoir effectif dans la définition des programmes de recherche : connaissant les insuffisances de l'industrie, ils peuvent l'appuyer dans l'expression de ses besoins.
- Au cours des dix dernières années, il me semble que le rôle des laboratoires de recherche a connu une mutation. La deuxième partie de cette communication montrera que des configurations complètes d'avions de transport supersoniques avaient fait l'objet d'essais en soufflerie, en amont du lancement du programme Concorde et à l'initiative de l'ONERA. Dans la définition des avions futurs, il me semble peu probable que cet office étudiera en amont de l'industrie des configurations complètes. Et je crois que c'est une bonne chose, car son rôle est maintenant ailleurs. Nombreux sont les phénomènes aérodynamiques encore mal modélisés, faute de connaissances de base suffisantes : l'interaction onde de choc - couche limite, les écoulements décollés, le mélange turbulent, etc. donnent lieu à des modélisations très grossières qui entraînent des aléas trop importants dans la prévision du comportement des avions en cours de développement ou d'avant-projet. Il appartient aux laboratoires de recherche de faire des expériences de base "simples et propres", d'en recueillir et d'en stocker toutes les données de quelque importance, de prendre l'initiative dans la recherche d'une modélisation convenable. Par contre, s'attacher à développer des méthodes de calcul qui sont déjà opérationnelles dans l'industrie et fournissent des résultats crédibles, s'obstiner d'année en année à rattraper tel ou tel retard, constitue pour un organisme de recherche l'un des moyens les plus efficaces de gaspiller son budget. La distinction classique entre les objectifs de l'industrie - à court terme - et ceux de la recherche - à long terme - est obsolète. Ce que l'on demande aujourd'hui aux organismes de recherche, c'est d'abord de fournir des renseignements et des schématisations sur des phénomènes de base mal connus, qui font peser des incertitudes excessives sur les programmes en cours et risquent de le faire aussi sur les programmes de demain.

Cette digression sur le rôle du laboratoire étant terminée, j'en reviens à mon sujet.

1.2.2.4 - Les résultats d'essais

L'introduction des méthodes de calcul d'écoulements ne supprime pas l'importance de la soufflerie. On a beaucoup parlé de la nécessité de développer des moyens d'essais à grands nombres de Reynolds, de manière à permettre une transposition plus directe aux conditions de vol. Il est vrai par exemple que si l'on passe d'un coefficient de portance maximale mesuré en soufflerie (à fait le nombre de Reynolds) à la valeur estimée pour le vol en appliquant un écart forfaitaire, on risque de s'attirer de sérieux déboires (cf Specialist's Meeting, AGARD/FDF Lisbonne 1972). Il se peut que dans un cas on ait un décollement par bulbe au bord d'attaque du bec, et dans l'autre cas un décollement de bord de fuite sur le dernier volet. Les deux écoulements sont tellement différents qu'on ne peut passer de l'un à l'autre par l'intermédiaire d'une simple correction globale.

Je pense que la meilleure approche est de coupler les essais et le calcul comme l'indique la figure 10. On évalue sur ordinateur, compte tenu des types d'écoulement, les coefficients aérodynamiques significatifs au nombre de Reynolds de vol d'une part, et ces coefficients aux nombres de Reynolds de

soufflerie d'autre part. Ces derniers sont comparés aux résultats de soufflerie et on s'interdit de retenir une configuration tant que toutes les divergences n'ont pas été expliquées et les modèles de calcul modifiés en conséquence. Ceci impose que la configuration essayée en soufflerie et préalablement optimisée par le calcul soit bonne à la fois aux nombres de Reynolds de vol et de soufflerie ; cette double optimisation est-elle possible ? Je ne saurais aujourd'hui répondre à cette question dans toute sa généralité ; toutefois, nous avons vécu deux cas de définition d'hypersustentateurs à basse vitesse sur deux types d'avions très différents (l'un de transport, l'autre de combat) et dans les deux cas la double optimisation a été possible moyennant d'assez longs tâtonnements.

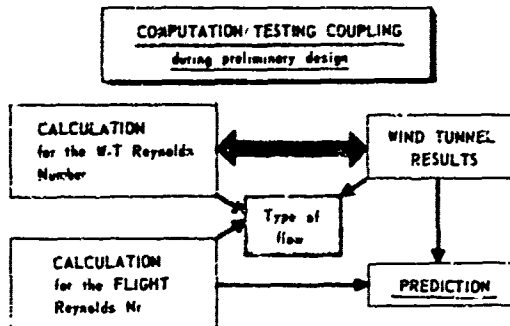


Fig. 10 - Couplage calcul/essai au cours d'un avant-projet.

Toute tentative d'extrapolation vol-soufflerie doit donc tenir compte des types d'écoulement aux différents nombres de Reynolds et se baser éventuellement sur des méthodes de calcul sophistiquées. Certaines difficultés que nous avons découvertes en vol sur des appareils français récents et pour lesquelles le remède ou les conséquences ont été coûteux apparaissaient déjà dans les résultats d'essais en soufflerie. Dans plusieurs cas, les spécialistes concernés, submergés sous le volume des résultats, n'avaient pas dépouillé le domaine intéressant ; pour les autres, les aérodynamiciens s'étaient hâtivement convaincus que "les choses s'arrangeraient avec le nombre de Reynolds". La conséquence est théoriquement simple en ce qui concerne le rôle des spécialistes des services officiels : s'assurer que les phénomènes significatifs ont été testés en soufflerie, vérifier que toutes les anomalies décelées en cours de campagne ont été sagement expliquées, n'accepter de transposition au Reynolds de vol qu'après une analyse des types d'écoulement et, en cas de litige, sur l'intermédiaire de méthodes de calcul crédibles.

Je mentionnerai encore deux conséquences importantes de ce qui vient d'être dit :

- il n'existe pas de nombre de Reynolds "magique" au-delà duquel la transposition aux conditions de vol serait immédiate et où le spécialiste pourrait se dispenser a priori d'être intelligent ;
- il faut être capable de modéliser les phénomènes aérodynamiques dans une large gamme de nombres de Reynolds, d'où l'importance de qualifier des méthodes de calcul par des expériences de base et des études fines d'écoulement effectuées dans les souffleries à grands nombres de Reynolds d'une part, en vol d'autre part.

1.2.2.5 - Enfin, au-delà de ce qui vient d'être dit, il y a une part considérable pour l'intuition. Le bon chef de projet et le bon critique des services officiels doivent l'une et l'autre en avoir une dose suffisante... mais cela n'est guère descriptible dans le cadre de cette communication !

1.2.2.5 - Illustration par un exemple

Je me contenterai d'un exemple décrivant notre approche d'un problème d'aérodynamique externe dans le cas de notre avion de combat futur (avion "B"). Cet appareil, sans préjuger de la solution qui sera retenue, dérivera probablement du Mirage GB (avion "A"). Pour tous les problèmes d'aérodynamique externe, l'approche suivante sera adoptée (fig. 11) :

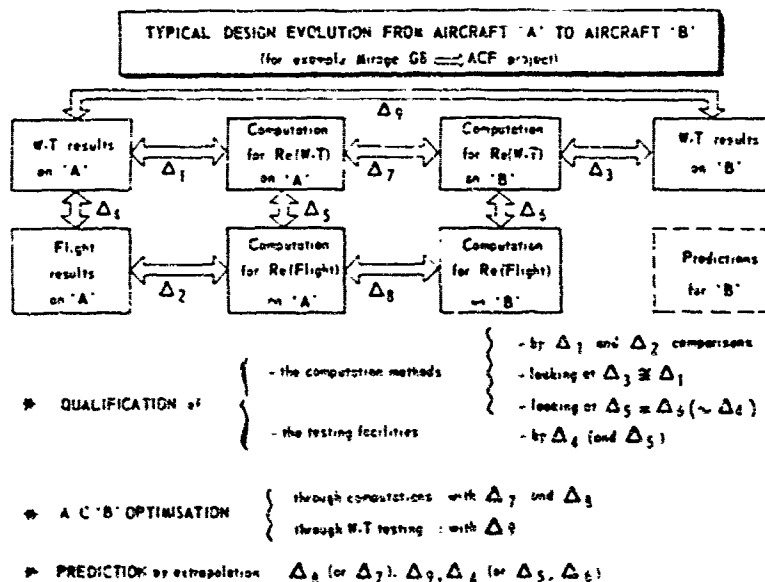


Fig. 11 - Utilisation de l'expérience acquise sur un projet antérieur.

- qualification des méthodes de calcul :
 - . comparaison calculs/essais en soufflerie sur A et B ;
 - . comparaison calculs/essais en vol sur A ;
- qualification des moyens d'essais :
 - . comparaison soufflerie/vol sur A (appuyée sur les résultats de calculs) ;
- optimisation de l'avion B :
 - double optimisation par le calcul (Reynolds de vol - Reynolds de soufflerie) ;
- prévision des coefficients aérodynamiques de B :
 - . application de différentes variations partant des résultats - des essais en vol de A - des calculs (Reynolds de vol) de B - des résultats en soufflerie de B.

Le problème de l'écoulement en transsonique (surtout avec des bords de bord d'attaque) a retenu toute notre attention, non que les résultats d'avions antérieurs aient été mauvais en ce domaine, mais parce que les méthodes d'investigation nous paraissent insuffisantes. L'action est menée sur deux fronts :

- Recherche :
 - essai d'une grande demi-voilure équipée de bords à fente au bord d'attaque (soufflerie S1 de Modane) ; peignage fin de l'écoulement dans les fentes et derrière les bords ; recherche d'une modélisation ;
- Développement :
 - la maquette de base de l'avion B (échelle 1/19) est destinée à la soufflerie S2 de Modane (1,70 m x 1,70 m ; pressurisée à 2,5 bars). La dimension correspondante permet d'étudier des bords sans fente, sans toutefois laisser la possibilité d'une analyse fine de l'écoulement. Une maquette de l'avion A ayant mêmes dimensions sera construite pour S2 et la comparaison entre S2 et le vol permettra de qualifier ce moyen d'essai ;
 - la réalisation de bords à fente n'étant pas possible sur une maquette de cette dimension, nous projetons de construire une autre maquette à l'échelle 1/8 destinée à la soufflerie S1 de Modane (8 m de diamètre, non pressurisée). La comparaison de la maquette au 1/19 dans S2 et de la maquette au 1/8 dans S1 au même nombre de Reynolds permettra de qualifier ce second montage ;
 - enfin, la maquette au 1/8 ne permet pas de peigner l'écoulement dans les fentes sur toute l'envergure de la voilure. Nous projetons donc de construire une demi-voilure au 1/4 pour S1 dont la validité du montage sera testée par comparaison avec les résultats de la maquette au 1/8.

En conclusion, cette qualification "en cascade" de moyens d'essais ne paraît illustrer une attitude délibérément orientée vers la minimisation des risques et l'utilisation maximale des résultats acquis lors du programme antérieur.

1.2.3 - Aperçu sur l'organisation

L'analyse critique d'un avant-projet est faite, au Service Technique Aéronautique, sous la direction d'un ingénieur responsable de l'ensemble de l'évaluation (y compris des coûts). Cet ingénieur de "marque" s'appuie sur deux ou trois équipes dont la plus importante, formée de cinq ingénieurs, traite les questions d'aérodynamique, de masses, de performances, de qualités de vol et de fiabilité. Ces cinq ingénieurs sont également chargés, dans les domaines les concernant :

- de suivre le développement des programmes civils et militaires en cours ;
- de participer au choix des opérations de recherches ;
- de suivre l'évolution des études et des méthodes mises en place dans l'industrie.

Cette situation ne semble présenter de grands avantages car elle permet aux ingénieurs chargés de l'analyse des avant-projets :

- de bien connaître les programmes antérieurs ;
- de pouvoir infléchir les recherches en fonction des besoins de l'industrie ;
- de savoir quel niveau de confiance on peut accorder aux méthodes de prévision de l'industrie.

Enfin, la taille très réduite de cette équipe la dissuade de mener par elle-même des travaux qui incombent à l'industrie, et l'incite à se concentrer sur l'essentiel de sa mission.

Tous ces éléments me semblent très favorables à la qualité de l'analyse critique au stade de l'avant-projet.

Deuxième Partie

LE RÔLE DU LABORATOIRE AU STADE DE L'AVANT-PROJET

par Philippe Poisson-Quinton

2.1 - INTRODUCTION

Dans la première partie de cet exposé, on a évoqué à plusieurs reprises certains rôles du laboratoire :

- recherches fondamentales très spécifiques, indispensables à une modélisation plus correcte de phénomènes mal connus, particulièrement dans les domaines de l'aérodynamique et de l'aéroélasticité (ceci est vrai aussi bien pour les cellules que pour les turbomachines) ;
- essais spécifiques effectués, dans les meilleures conditions de rapidité et de précision, à la demande des constructeurs.

En fait, si ce double rôle est absolument nécessaire, je pense qu'il n'est pas suffisant, en particulier lorsqu'on étudie, non plus un avant-projet relativement classique qui s'appuie sur une certaine continuité de l'"état de l'Art", mais au contraire un concept tout à fait nouveau qui nécessite un "saut technologique"; la partie la plus "vivante" du Centre de Recherche doit préparer l'avenir grâce à des recherches exploratoires entrant dans le cadre de programmes prospectifs à long terme initiés par les services officiels ; ce petit groupe chargé de susciter des innovations puis de les qualifier par des "coups de sonde" peut ne représenter qu'une faible proportion de l'effectif global, par exemple moins de 5 %, mais il doit bénéficier d'une grande liberté de manoeuvre et du soutien logistique du Centre de Recherches (accès aux moyens généraux de calculs et d'essais, avec son personnel spécialisé) ; je donnerai plus loin quelques exemples de telles recherches exploratoires vécues à l'ONERA ces dernières années. Au cours des prochaines années, il est aisé de prévoir plusieurs "sauts technologiques" sur les nouvelles générations d'avions, qui porteront par exemple sur l'application des nouveaux systèmes de contrôle actif (C.C.V.) et des nouveaux profils supercritiques ; le rôle du laboratoire semble particulièrement important en amont et au cours du développement de ces concepts.

C'est pourquoi le rôle du Centre de Recherches semble important au stade de l'avant-projet, en particulier lorsqu'il s'agit de concepts nouveaux et ceci au cours des trois phases suivantes (fig. 12) :

- avant le lancement d'une demande d'avant-projet par les organismes gouvernementaux, le Centre de Recherches est particulièrement apte à participer à la synthèse sur l'"état de l'Art" concernant le concept proposé, puis à fournir certains éléments d'un choix à partir d'études paramétriques théoriques ou semi-empiriques, et enfin à lancer des expérimentations préliminaires ("coups de sonde") pour confirmer, ou infirmer, l'intérêt des formules envisagées ;
- lorsque les contractants pour l'avant-projet ont été choisis, le rôle du laboratoire est d'abord de satisfaire le plus rapidement possible aux demandes des constructeurs, qu'il s'agisse de calculs théoriques ou d'essais spécifiques ;
- enfin, en conclusion de l'étude d'avant-projet, les services officiels ont souvent à donner une nouvelle orientation aux Programmes de Recherches, soit pour que les laboratoires approfondissent les points faibles révélés au cours de l'étude, soit pour qu'ils développent des moyens d'essais nouveaux qui devront être opérationnels au moment où un nouveau projet majeur sera lancé ; ce dernier point est particulièrement important pour le Centre de Recherches, car il assure sa future efficacité et engage son avenir.

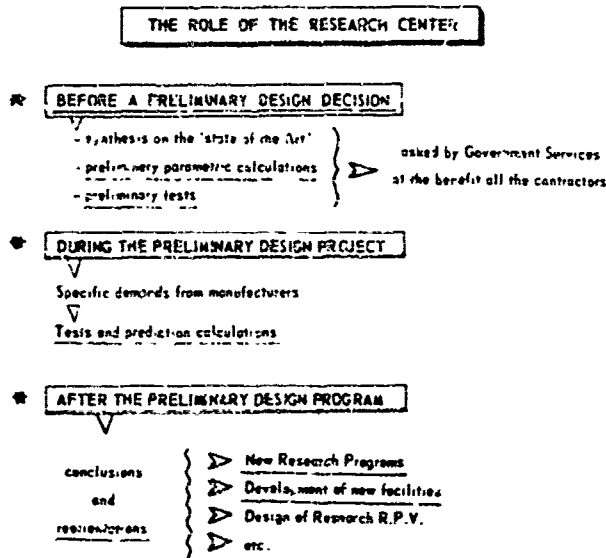


Fig. 12 - Le rôle du centre de recherches autour d'un avant-projet.

En effet, le Centre de Recherches doit, par définition, être techniquement "en amont" du constructeur chargé d'un avant-projet ; c'est ici la première difficulté, car il faut prévoir à temps, c'est-à-dire au moins cinq ans à l'avance, quels seront les besoins du constructeur en moyens d'essais ; (il faut en effet au moins cinq ans pour développer par exemple une nouvelle soufflerie), et c'est le rôle des plans à long terme, que le Centre de Recherches doit réorienter continuellement pour tenir compte de l'évolution rapide de la technique aéronautique et des besoins à la fois militaires et civils.

En fait, les chercheurs doivent s'adapter très rapidement à des changements d'orientation allant par exemple du problème de l'essor vertical d'un VTOL à celui de l'échauffement d'une structure en vol hypersonique, en passant par l'adaptation optimale de la forme d'une aile au vol transsonique ; dans ces trois cas, il lui faudra préparer des moyens d'essais souvent très spécialisés et coûteux, dont le plan de charge est très difficile à prévoir. Il y a donc toujours à prendre un pari sur l'avenir lorsqu'on lance une nouvelle installation ; quelque fois, elle est utilisée pour des essais que l'on n'avait pas prévus à l'origine (par exemple, une grande partie de l'activité de St Modane est actuellement consacrée aux basses vitesses alors que cette soufflerie était surtout destinée aux essais en subsonique élevé). Souvent, c'est à l'occasion d'une étude d'avant-projet qu'apparaissent les futurs besoins, et nous en avons vécu un exemple lors d'une étude récente de prospective sur les STOL de transport civil demandée aux constructeurs par les Services Officiels français (fig. 13) : cette étude d'avant-projet a fait clairement apparaître les lacunes actuelles de nos moyens d'investigation concernant le bruit des avions STOL et l'analyse de leurs qualités de vol avec une simulation correcte de l'environnement (effet de sol, rafales, vitesse de descente à l'approche, etc.). Pour répondre à ces besoins à moyen terme, l'ONERA a lancé deux nouveaux moyens d'essais importants grâce au soutien financier de l'Aviation Civile (S.G.A.C.) :

- un montage très sophistiqué (fig. 14, 15) pour l'étude des qualités de vol, en profitant de la grande taille de la soufflerie St de Modane ;

RESEARCH CENTER MUST PREPARE NEW TESTING FACILITIES ON TIME

EXAMPLE : STOL DEVELOPMENT IN THE 1980:

CONCLUSIONS OF A FRENCH PRELIMINARY DESIGN STUDY :

THE MAIN PROBLEMS TO SOLVE ARE :

- Noise
- Engine/Airframe Interference
- Flying qualities near the ground
- Flight in bld weather

DECISIONS OF THE FRENCH AGENCIES (SGAC, STAO, CEV, ONERA) :

- 1) Development of a special V/STOL Rig inside SI Modane W-T (1971-75)
 - - Static and Dynamic derivatives on 3 axis.
 - - Ground effect simulation
 - - Gust simulation
 - - Flare simulation in ground-effect
- 2) Construction of an anechoic W-T at Petersham ONERA (1972-75)
 - - Noise environment around Engine/Airframe models, without and with flight speed simulation
 - - Research on acoustic treatment (Engine inlet and exhaust) with speed simulation
- 3) Utilization of the Brueet 941 STOL at the CEV
 - - New cockpit displays
 - - Fly-over Noise : $f(V_0, T, h, \theta)$
 - - Flight Simulator comparisons
 - - Certification procedures

Fig. 13 - Planification de nouveaux moyens d'essais à la suite d'une étude d'avant-projet.

Scale w.t. : $70 < V_0 < 150 \text{ m/s}$
 $Re_T \sim 2.5 \cdot 10^6 \text{ at } V_0 = 70 \text{ m/s}$

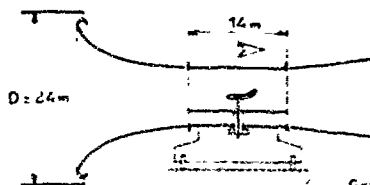


Fig. 14 - Montage d'étude des qualités de vol dans la soufflerie SI de Modane.

Special V/STOL rig in SI Modane W.T. for

- steady and unsteady derivatives
- performance evaluation
- handling qualities, in and out of ground effect
- gust sensitivity.

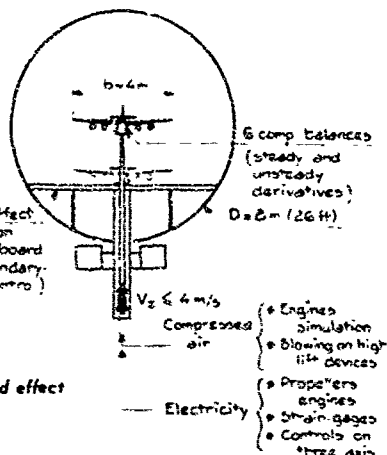
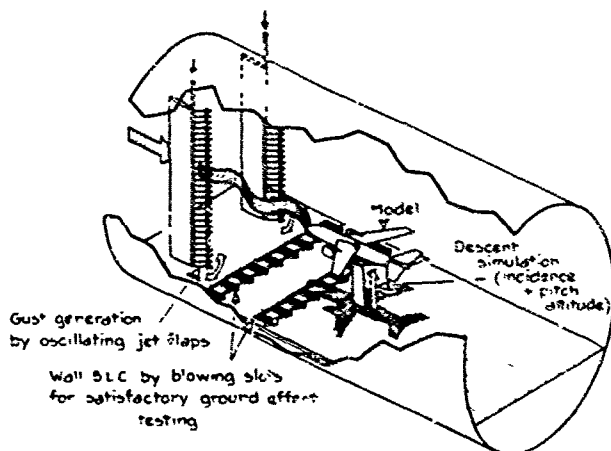


Fig. 15 - Simulation de rafales transversales au cours de l'approche, dans l'effet de sol, à SI Modane.

ONERA - SI Modane W.T. projected rig to simulate lateral gusts during approach in ground effect.



- un tunnel anéchoïque (fig. 16, 17) pour l'étude du bruit de survol, et non plus seulement du bruit émis au point fixe, grâce à l'essai de maquettes motorisées ou de propulseurs autour desquels on étudiera la propagation du bruit.

Simultanément, les problèmes spécifiques liés aux caractéristiques STOL sont regardés en vol et sur simulateur (au CEV et à Ames) en profitant d'un Bréguet 941 opérationnel de l'Armée de l'Air.

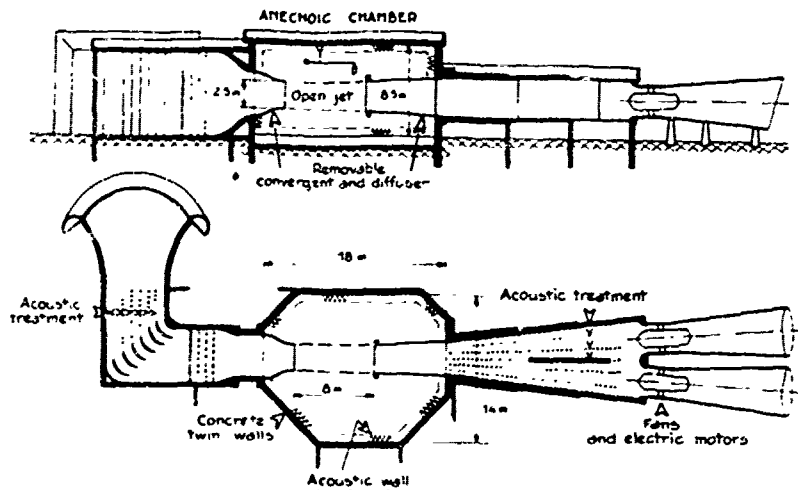
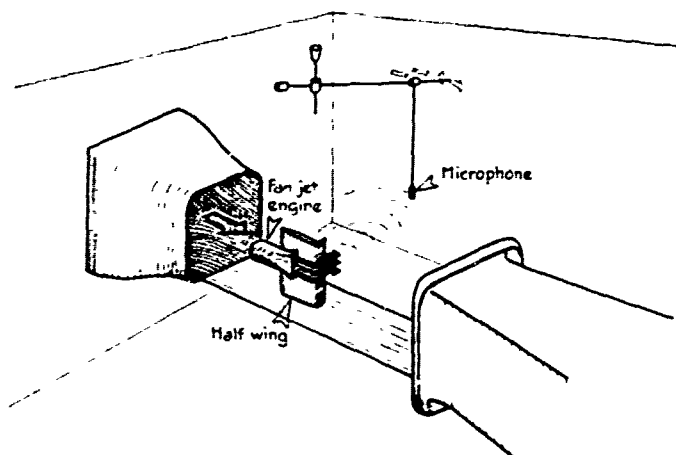


Fig. 16 - Projet de soufflerie anéchoïque de l'ONERA à Palaiseau.

ONERA Palaiseau - Project of an anechoic chamber with integrated open tunnel.

$$(S_{\max} = 10 \text{ m}^2, V_{0\max} = 55 \text{ m/s})$$

Fig. 17 - Etude du bruit de survol sur maquette motorisée dans le jet libre du tunnel anéchoïque.



Essays de reconstituer maintenant les liaisons laboratoire/constructeur souhaitables au stade de l'avant-projet ; il faut d'abord souligner (fig. 18) que le lancement d'un avant-projet constitue, pour le chercheur, une motivation essentielle lui permettant :

- de rencontrer les constructeurs, et de parler le même langage ;
- de savoir quels sont ses problèmes à résoudre à court terme ;
- de développer de nouvelles approches théoriques et de nouvelles techniques d'essais ;
- enfin, d'apprendre à travailler vite.

C'est également l'occasion, pour le groupe d'avant-projet du constructeur :

- de rencontrer des chercheurs dans leurs laboratoires ;
- de les convaincre de travailler tout de suite sur leurs projets ;
- de découvrir des idées nouvelles et d'essayer de les appliquer ;
- enfin, d'encourager le développement de nouveaux moyens de Recherches, dont ils seront les premiers bénéficiaires.

C'est enfin le rôle des Services Officiels de coordonner et de répartir les tâches entre le constructeur et le laboratoire. Au premier stade du projet, le Centre de Recherche apporte souvent une contribution importante pour la synthèse des informations techniques disponibles : un bon service de documentation et la connaissance des rapports essentiels au niveau du spécialiste sont fondamentaux dans un Centre de Recherche efficace.

IMPACT OF A PRELIMINARY DESIGN POLICY ON THE RESEARCH CENTER EFFICIENCY .

1) For Governmental RESEARCHERS.

It is the best MOTIVATION :

- ✦ TO MEET MANUFACTURERS,
- ✦ TO KNOW THE SHORT TERM PROBLEMS,
- ✦ TO DEVELOPP NEW THEORETICAL APPROACH AND NEW TESTING TECHNIQUES,
- ✦ TO WORK QUICKLY

2) For the MANUFACTURERS,

It is the best OPPORTUNITY

- ✦ TO MEET LAB'S PEOPLE,
- ✦ TO CONVINCe them to work on their short-term project,
- ✦ TO look at NEW IDEAS and apply them,
- ✦ TO PUSH NEW RESEARCH CAPABILITIES

Fig. 18 - Importance des études d'avant-projet sur l'efficacité d'un Centre de Recherches.

La deuxième phase est la préparation en commun des essais indispensables au stade de l'avant-projet (fig. 19) ; les principaux paramètres entrant dans le choix des essais sont peut-être d'abord la rapidité d'exécution et leur prix, puis les conditions d'essais offertes (Reynolds, Mach, type de maquette et dimensions).

A ce stade, le constructeur n'a souvent qu'un faible budget (fig. 20), et il a tendance à économiser sur les essais, surtout si le temps de réponse pour avoir accès aux résultats est trop long ; il faut souligner ici l'intérêt d'un traitement rapide de l'information permettant au constructeur de suivre les essais pratiquement en temps réel et de modifier le programme à sa convenance.

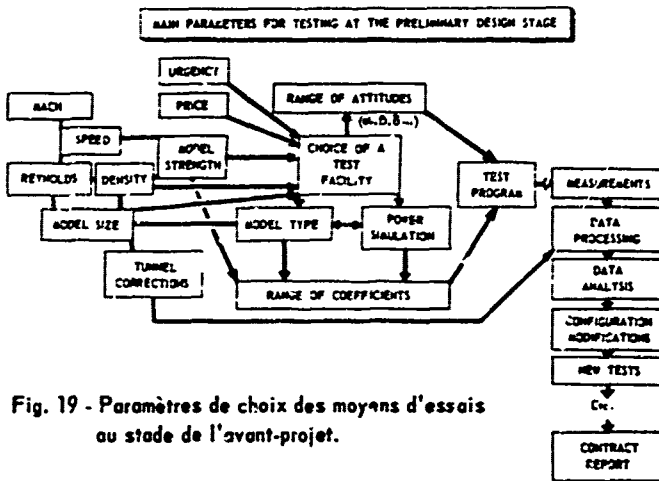


Fig. 19 - Paramètres de choix des moyens d'essais au stade de l'avant-projet.

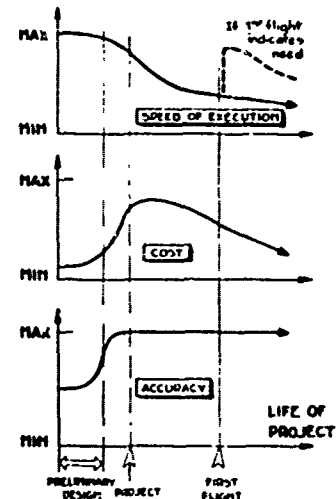


Fig. 20 - Evolution des facteurs (rapidité, précision) pour les essais sur un projet. Relative desirability of test attributes (after C. Russel, BAC/UK).

A mesure que la forme du projet se précise, le constructeur tend à demander une plus grande précision sur les résultats des essais, qui nécessite de continuer l'étude sur des maquettes plus grandes et plus sophistiquées.

Dès les premiers essais, il est capital de faire un très large balayage des attitudes de maquette (incidence, dérapage, etc.), qui doivent dépasser très largement les valeurs prévues en vol, surtout si le projet est basé sur un concept nouveau : nous avons vécu un exemple typique d'un avion VTOL (fig. 21) qui présentait une très forte interaction aérodynamique entre la voilure et les jets sustentateurs ; à cette époque, ces problèmes n'apparaissent pas clairement au cours d'essais préliminaires assez grossiers ; en fait, après un accident survenu en vol sur le prototype expérimental, des études très détaillées permirent d'expliquer l'accident dont l'origine était un couple de roulis considérable apparaissant lorsque l'appareil était en dérapage (différence de perte de portance par "effet douche" entre l'aile gauche et l'aile droite).

Dans ce même ordre d'idée, je rappellerai que pour certaines configurations d'avions classiques, il est apparu aux grandes incidences une instabilité longitudinale (fig. 22), la tendance au "pitch-up", qui pouvait entraîner l'avion à des incidences extrêmes, de l'ordre de 40°, où l'avion redevenait stable, mais ne pouvait sortir de cette difficile position par manque de puissance de la gouverne de profondeur ; ce n'est qu'après un accident fatal que les laboratoires reconstituèrent le phénomène en soufflerie.

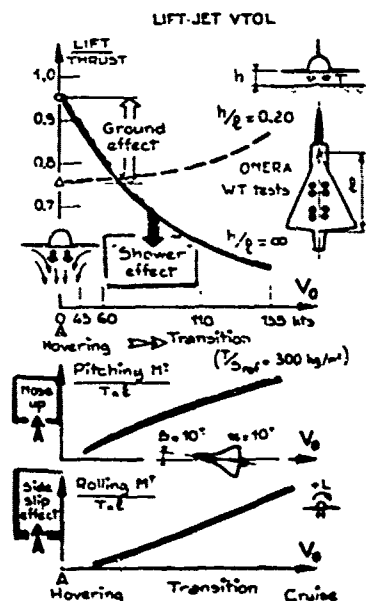
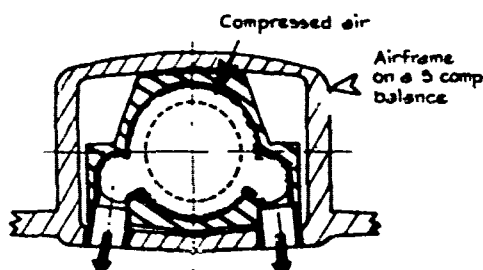


1/22 scale model ONERA-Cannes w.t.

Fig. 21 -

Détérioration des qualités de vol transversales d'un VTOL liée à l'interaction voilure/jets de sustentation.

ONERA - Lift jets/airframe interference on a VTOL in hovering and during low-speed acceleration.



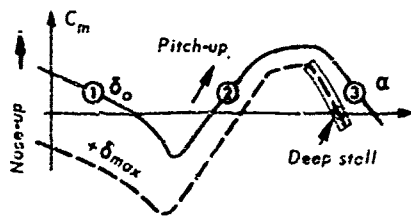


Fig. 22 - Détérioration des qualités de vol longitudinales d'un avion classique aux grandes incidences.

Wind tunnel analysis of the deep-stall after an accident.

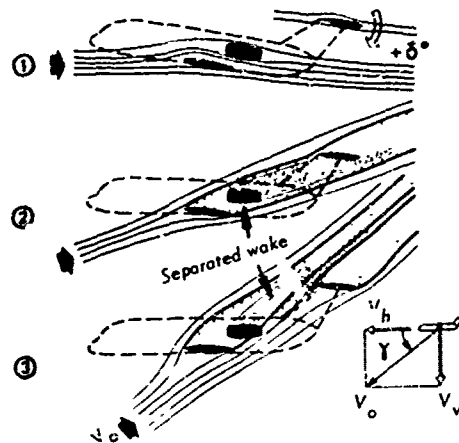


Fig. 23 - Exemples d'avant-projets nécessitant des sauts technologiques.

Ces deux exemples, parmi beaucoup d'autres malheureusement, montrent qu'il ne suffit d'accumuler les heures d'essais en soufflerie au moment de l'étude d'un projet, mais qu'il est essentiel d'analyser dans le détail les résultats obtenus et si besoin, de les compléter immédiatement : c'est le travail du représentant du constructeur aidé par l'ingénieur chargé des essais ; celui-ci doit être informé des objectifs recherchés et non pas considéré comme un simple "marchand de vent" : son expérience peut être précieuse pour le constructeur à ce stade exploratoire.

2.2 - LE ROLE DE LA RECHERCHE DANS LES SAUTS TECHNOLOGIQUES

Je voudrais maintenant montrer que des "sauts technologiques" importants peuvent aboutir à des développements satisfaisants si une étroite collaboration entre le chercheur et le constructeur est orchestrée par les Services Officiels ; je prendrai comme exemples (fig. 23) certains des projets français dont l'étude d'avant-projet avait soulevé des problèmes tout à fait nouveaux à l'époque de leur lancement :

SOME EXAMPLES of French PRELIMINARY DESIGN on a NEW AIRCRAFT CONCEPT

- VTOL SUPERSONIC FIGHTER (A.M.D. 'Balsac' and 'Mirage 3V')
- STOL TRANSPORT (Breguet 940 \Rightarrow 941)
- VARIABLE SWEEP FIGHTER (A.M.D. 'Mirage G \Rightarrow G-8')
- SUPERSONIC TRANSPORT (UK/FR 'Concorde')
- RAM-JET and TURBO-RAM-JET (ONERA RPV, ONERA SCRAM-JET, Nord 'Griffon' A/C)

2.2.1 - Avion de combat supersonique à essor vertical

Il s'agissait d'intégrer à une configuration delta, qui avait déjà fait ses preuves pour des chasseurs et bombardiers supersoniques, un système de sustentation par réacteurs verticaux ; comme nous l'avons vu plus haut, de graves problèmes d'interactions jets verticaux/voilure n'apparurent qu'au stade du vol, faute de méthodes d'essais sophistiquées au stade de l'avant-projet ; par contre, la soufflerie permit l'étude en vraie grandeur du fonctionnement, puis l'optimisation de l'ensemble des réacteurs de sustentation dans la soufflerie S1 de Modane grâce à une collaboration étroite entre les Sociétés Dassault/Rolls-Royce et l'ONERA au moment du lancement du projet Mirage 3V ; grâce à cette étude en soufflerie, la mise au point en vol du système sustentateur fut très rapide.

2.2.2 - Transport à décollage court

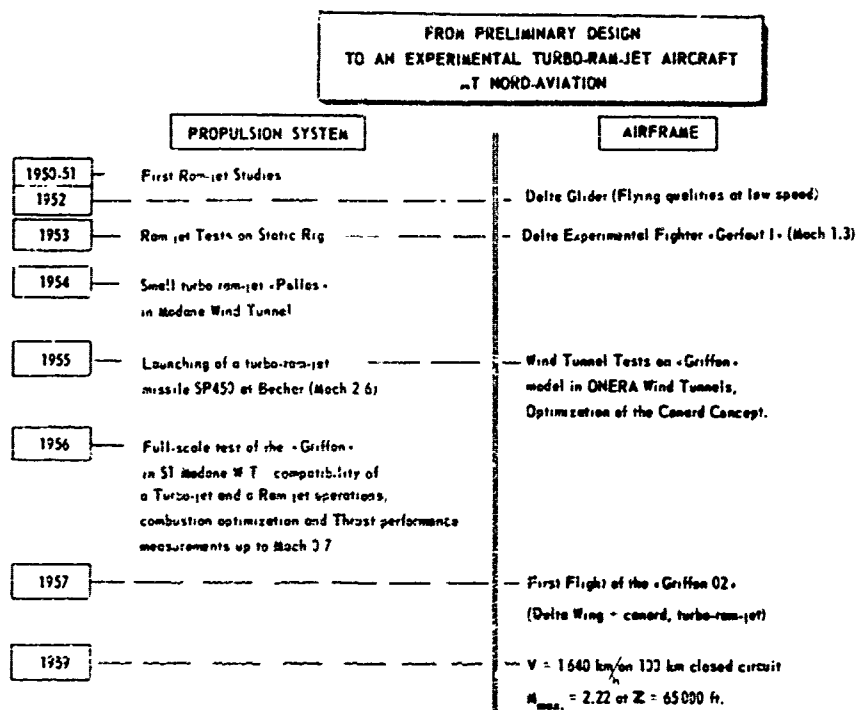
Ce concept avait été proposé dès la fin de la guerre par la Société Bréguet, qui entreprit à cette époque des recherches fondamentales sur la déflexion du flux d'une hélice au moyen de volets de courbure multiples, à l'Institut Aérotechnique de Saint-Cyr ; les étapes successives du développement de ce concept STOL : avion expérimental Bréguet 940, puis avion cargo militaire 941, furent précédées de minutieuses recherches dans les souffleries du constructeur et de l'ONERA, allant jusqu'à l'essai en vol semi-libre d'une maquette motorisée dynamiquement semblable dans la soufflerie S1 de Chalais ; cette technique d'essai nouvelle permit aux pilotes de se familiariser avec le comportement et le contrôle de la machine aux basses vitesses avec flux intense et fortement défléchi des hélices.

2.2.3 - Propulsion par stato-réacteur

L'idée de la propulsion par stato-réacteur est contemporaine du développement de l'aviation (Lorin, 1913) et son pionnier en fut René Leduc, qui fit voler les premières machines expérimentales au lendemain de la dernière guerre ; les Services Officiels français stimulèrent également à cette époque les recherches sur les applications militaires du stato-réacteur à l'Arsenal de l'Aéronautique, qui devait devenir par la suite Nord-Aviation ; pendant plus de dix ans (fig. 24), une équipe de spécialistes aidée de plusieurs Centres de Recherches et d'Essais se consacra à une succession d'avant-projets suivis de réalisations expérimentales essayées d'abord en soufflerie puis en vol ; ce processus alterné "recherche/développement" devait conduire à la remarquable réalisation de l'avion expérimental supersonique "Griffon" propulsé par un turbo-stato-réacteur (fig. 25) et à des missiles à stato-réacteurs opérationnels.

À la suite de ces succès, les Services Officiels demandèrent à l'ONERA des études d'avant-garde sur l'application du stato-réacteur au vol hypersonique (fig. 26), qui devaient aboutir d'une part au

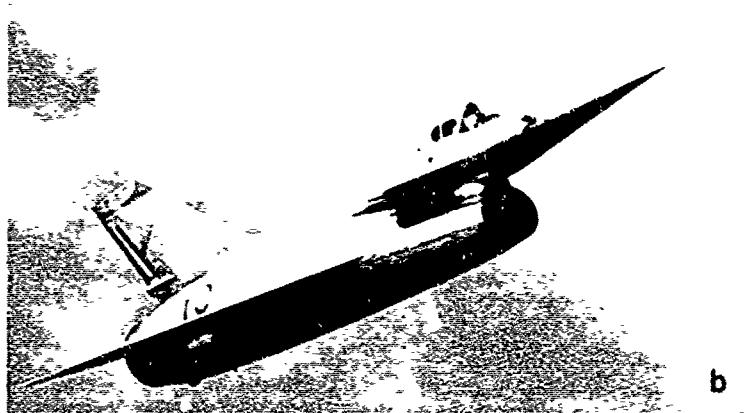
Fig. 24 - Développement de la propulsion par stato-réacteurs à Nord-Aviation.



a) Full scale tests on the turbo-ram-jet in SI Ma W.T. (Combustion problems and thrust up to M = 0.7).

b) Nord-Aviation «Griffon-02» experimental supersonic aircraft.

Fig. 25 - Etude en soufflerie et en vol de l'avion à turbo-stato-réacteur «Griffon 02».



lancement réussi de plusieurs missiles expérimentaux qui atteignirent un nombre de mach de 5 en vol à haute altitude, et d'autre part, à l'essai de qualification d'un stato-réacteur à combustion supersonique d'hydrogène adapté au vol à Mach 6 ; de tels essais ont été rendus possibles grâce au développement, au Centre de Modane, d'une soufflerie hypersonique capable de simuler les conditions de température et de pression d'un vol réel à Mach 6 ; cette installation va permettre d'étudier maintenant en laboratoire le missile expérimental "Scorpion" en vraie grandeur en vue d'optimiser son aérodynamisme interne et son stato-réacteur à combustion subsonique de kérosène à Mach 6.

Une telle continuité pendant plus de vingt ans dans l'effort de recherche suivi de développements expérimentaux est assez exemplaire : les succès obtenus sont justement liés à cette continuité au

sein de petites équipes de chercheurs et de réalisateurs étroitement associés dans les responsabilités jusqu'à la qualification de chaque projet.

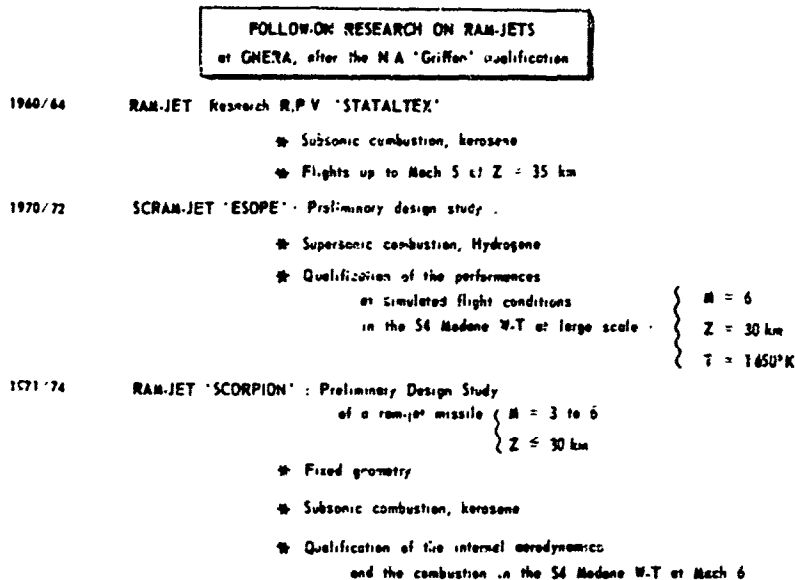


Fig. 26 - Programme de l'ONERA sur les applications des statoréacteurs au vol hypersonique.

2.2.4 - Avion de transport supersonique

Le cas "Concorde" a marqué profondément, depuis plus de dix ans, l'évolution des centres de Recherches et d'essais, particulièrement en Angleterre et en France, mais aussi en Hollande (NLR), qui ont participé activement au programme :

- d'abord avant la naissance du programme,
- puis pendant l'avant-projet,
- et enfin au cours du développement de l'avion.

L'élément nouveau a été ici la mise en commun des connaissances scientifiques et techniques acquises au moment du lancement du projet, respectivement en Angleterre et en France ; au niveau de la recherche, le RAE et l'ONERA collaboraient déjà depuis plusieurs années sur les problèmes aérodynamiques posés par le vol supersonique et ils furent alors étroitement associés aux avant-projets demandés par les services officiels respectifs ; je n'évoquerai ici que quelques recherches de l'ONERA qui ont été plus ou moins directement utilisées, ou engagées, pour le projet Concorde dans les domaines de l'aérodynamique et de la propulsion. La figure 27* résume très schématiquement les différentes études entreprises dès 1950 sur l'aérodynamique des ailes élancées, particulièrement aptes au vol supersonique.

A cette époque, la grande surprise fut la découverte du supplément de portance que pouvait apporter un régime tourbillonnaire stable sur l'extrados des ailes à forte flèche (tourbillon en cornet, figure 27A), qui compensait quelque peu leur faible portance unitaire aux basses vitesses ; de minutieuses recherches expérimentales permirent de dégager des formes évolutives de bord d'attaque avec apex à forte flèche et bout d'aile arrondi (aussitôt baptisées "gothiques flamboyantes", fig. 27C), permettant respectivement d'accroître la portance tourbillonnaire et d'éviter le pitch-up aux grandes incidences ; une surprise désagréable apparut par contre dès les premiers vols d'un petit avion expérimental lancé par l'ONERA, le "Deltavreix" (fig. 27B), qui avait une aile en "queue d'hirondelle" avec une flèche accentuée au bord d'attaque : la combinaison du faible allongement et de la petite échelle de cet avion entraînait en effet un manque d'amortissement en roulis qui le rendait pratiquement insupportable ; ce leçon fut cependant fructueuse puisqu'elle nous obligea à développer un système original de stabilisation automatique en roulis : ce fut certainement la première application du concept "jet-flap" sous forme de rampes de soufflage installées au bord de fuite de l'aile et alimentées par prélèvement d'air sur le compresseur du turboréacteur ; le temps de réponse de ce "contrôle pneumatique" asservi à un gyromètre de roulis était de quelques centièmes de seconde, ce qui permit d'assurer un amortissement satisfaisant de l'avion au cours des vols qui suivirent.

Une autre "retombée" de cette étude exploratoire d'avant-garde fut de permettre aux chercheurs de quitter un moment leur laboratoire pour se frotter aux difficiles réalités de l'expérimentation en vol : de cette époque date la formation à l'ONERA d'une division de recherche chargée à la fois de la conception d'engins expérimentaux, de la responsabilité de leur expérimentation en vol et enfin de l'exploitation des résultats obtenus.

Dans le domaine des grandes vitesses, les études d'avant-projet de l'époque nous orientèrent vers le calcul de l'adaptation du bord d'attaque (cambrure conique, figure 27E) ou de l'ensemble d'une aile delta (village et cambrure, fig. 27F) permettant d'améliorer considérablement la finesse, respectivement en subsonique et en supersonique.

Le grand tournant de la recherche sur les avions supersoniques fut pris en 1958 lorsque le Service Technique Aéronautique demanda à l'ONERA de "regarder" quelles pourraient être les caractéristiques d'un avion de transport capable d'une croisière supersonique sous le double aspect de l'aérodynamique et de la propulsion.

* - Voir à ce sujet l'exposé de M. Salmon "Concorde et la recherche aéronautique" - Aéronautique et Astronautique n° 11 (4-1969).

Des études modestes, à la fois théoriques et expérimentales, furent rapidement lancées pour cerner les problèmes essentiels posés pour un tel objectif civil qui exigeait à la fois un rendement aérodynamique et un niveau de sécurité très supérieurs à ceux obtenus sur les avions supersoniques militaires contemporains.

Une étude paramétrique, basée sur les connaissances théoriques de l'époque, fut entreprise pour rechercher la configuration optimale d'une configuration "delta" en croisière supersonique (fig. 27) ; cet exercice fut fort instructif car il permit pour la première fois de préciser la sensibilité d'un projet :

- à l'élanement et au diamètre du fuselage ;
- à l'adaptation de la voilure et à son épaisseur relative.

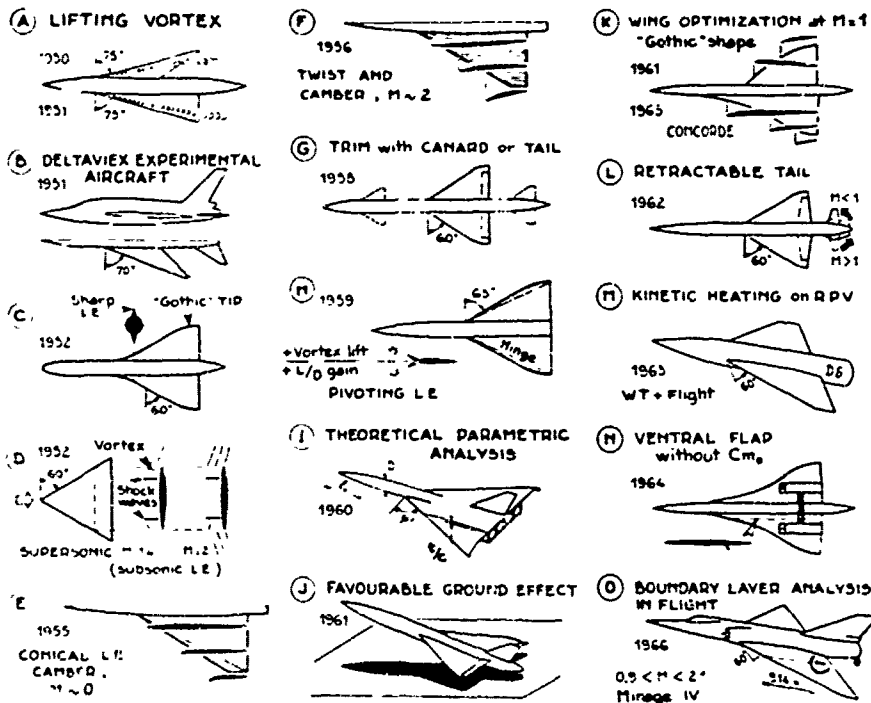


Fig. 27 - Programme de l'ONERA sur les ailes élanées avant et pendant l'avant-projet de « Concorde ». ONERA (1950-56). Slender wing aerodynamic research before and during the Concorde design.

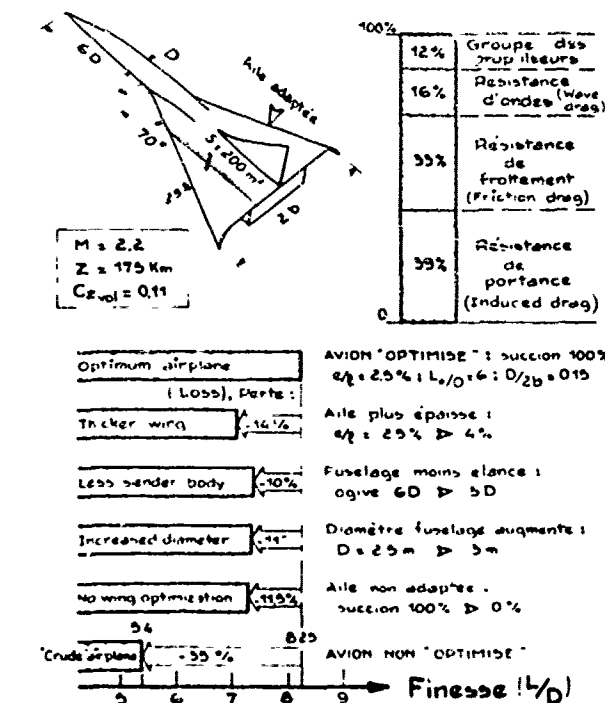


Fig. 28 - Calcul d'optimisation d'une configuration d'avion de transport supersonique.

Supersonic drag calculation on S.S.T. (ONERA, 1959).

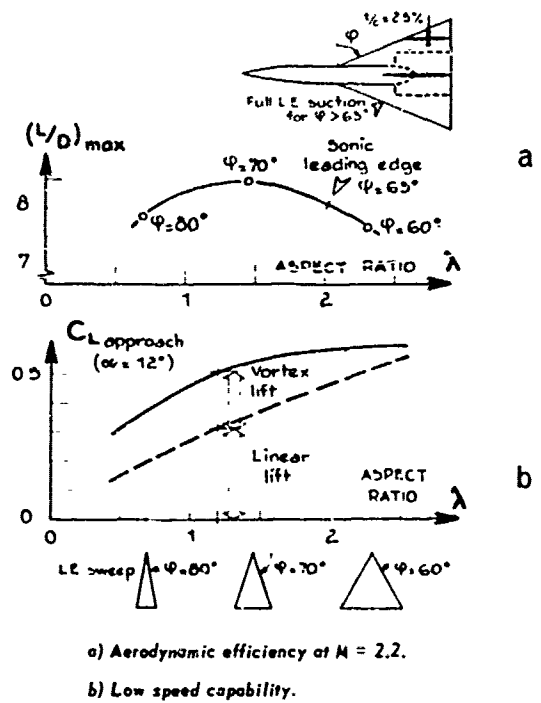


Fig. 29 - Etude paramétrique d'une famille T.S.S. ; finesse maximale en croisière supersonique et performances en approche.

S.S.T. parametric study on delta wings (ONERA, 1959).

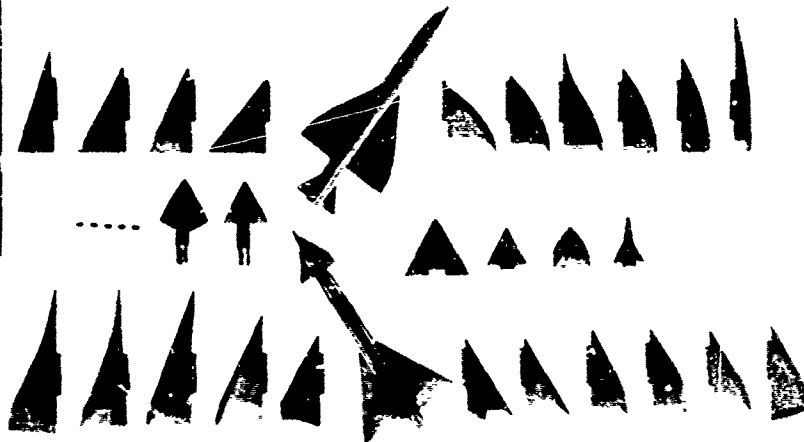
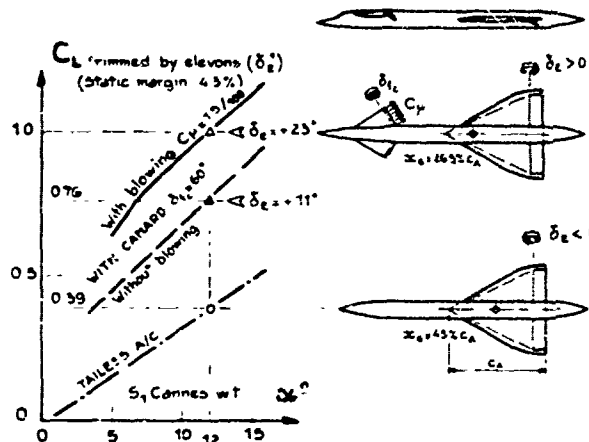
La figure 28 précise ces tendances à la croisière $M = 2,2$ dans le cas d'une aile delta à 70° de flèche : il nous fallait gagner un bon tiers de finesse aérodynamique par rapport aux avions militaires existants si nous voulions avoir une chance d'être compétitifs sur le marché civil. Encore fallait-il que la configuration optimale pour le vol supersonique soit acceptable au décollage et à l'atterrissage ; ici encore, une étude expérimentale préliminaire montra (fig. 29b) que le supplément de portance tourbillonnaire compensait suffisamment la perte liée à l'accroissement de la flèche pour que la portance globale soit encore acceptable pour des flèches accentuées, de l'ordre de 70° , souhaitables pour une bonne finesse supersonique à $M = 2,2$; simultanément, l'expérience montrait qu'un bord d'attaque pivotant (fig. 27H) permettait soit d'augmenter la portance à l'approche (accentuation de la portance tourbillonnaire par braquage vers le haut), soit d'améliorer la finesse en montée et en croisière subsonique (braquage vers le bas) ; des portances équilibrées beaucoup élevées furent obtenues en soufflerie grâce à un empennage à l'arrière ou à un plan canard (fig. 27G) ; un plan canard à volet soufflé très efficace fut ensuite développé qui permit de doubler la portance d'approche par rapport à la solution "avion sans queue" sans interaction aérodynamique grave aux grandes incidences (fig. 30).

Fig. 30 - Recherches sur l'hypersustentation d'une configuration de transport supersonique avec un plan canard actif aux basses vitesses.

ONERA - Research on SST lift increase at low speed with a canard configuration.

Fig. 31 - Etude expérimentale préliminaire des formes d'un projet de transport supersonique.

ONERA, 1959 - First experimental approach on SST shapes in small transonic and supersonic tunnels.



Simultanément, des études expérimentales préliminaires en transsonique et en supersonique sur des maquettes simplifiées permettaient d'orienter les choix et de fournir rapidement les dérivées aérodynamiques suffisantes au stade de l'avant-projet (fig. 31) ; au cours de cette expérimentation modeste, nous avons montré l'intérêt d'inverser la forme de l'habitacle pour qu'il participe à l'équilibrage longitudinal de l'avion en croisière supersonique (fig. 32) tout en permettant une bonne vision vers le bas à l'approche aux grandes incidences...

En novembre 1959, le Service Technique Aéronautique lançait des études d'avant-projets pour un transport supersonique moyen courrier ($R = 3000$ km, 80 passagers, Mach compris entre 2 et 3) auquel répondaient trois constructeurs (Nord, Sud, Dassault) ; il est intéressant de revoir deux des avant-projets présentés, tous deux adaptés à une croisière à un nombre de Mach de 2-2,2 pour pouvoir utiliser une structure en alliage d'aluminium :

- un quadriréacteur à aile "gothique" sans empennage, par Sud-Aviation (fig. 33) ;
- un quadriréacteur à aile delta et empennage canard "soufflé", par G.A.M. Dassault, dont la voilure et les prises d'air s'inspiraient fortement du bombardier Mirage IV (fig. 34).

Du côté britannique, un comité d'études réunissant les organismes d'Etat et l'Industrie Aéronautique avait été créé dès 1956 pour entreprendre des recherches aérodynamiques et structurales sur la forme à donner à un avion de transport supersonique ; en 1960, la British Aircraft Corporation reçut un contrat de "SST long courrier" capable d'une croisière à $M = 2,2$; la figure 35 montre le résultat de cette étude : configuration à aile gothique et nacelles propulsives intégrant chacune trois réacteurs, fuselage

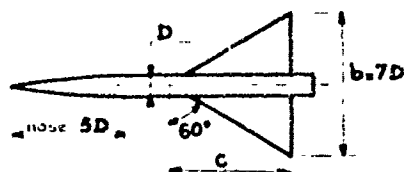


Fig. 32 - Étude d'un habitacle inverse participant à l'équilibre longitudinal d'un T.S.S. en croisière supersonique.

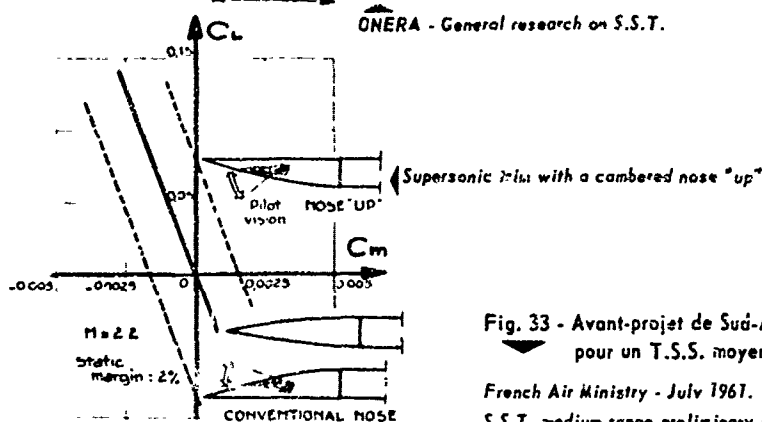
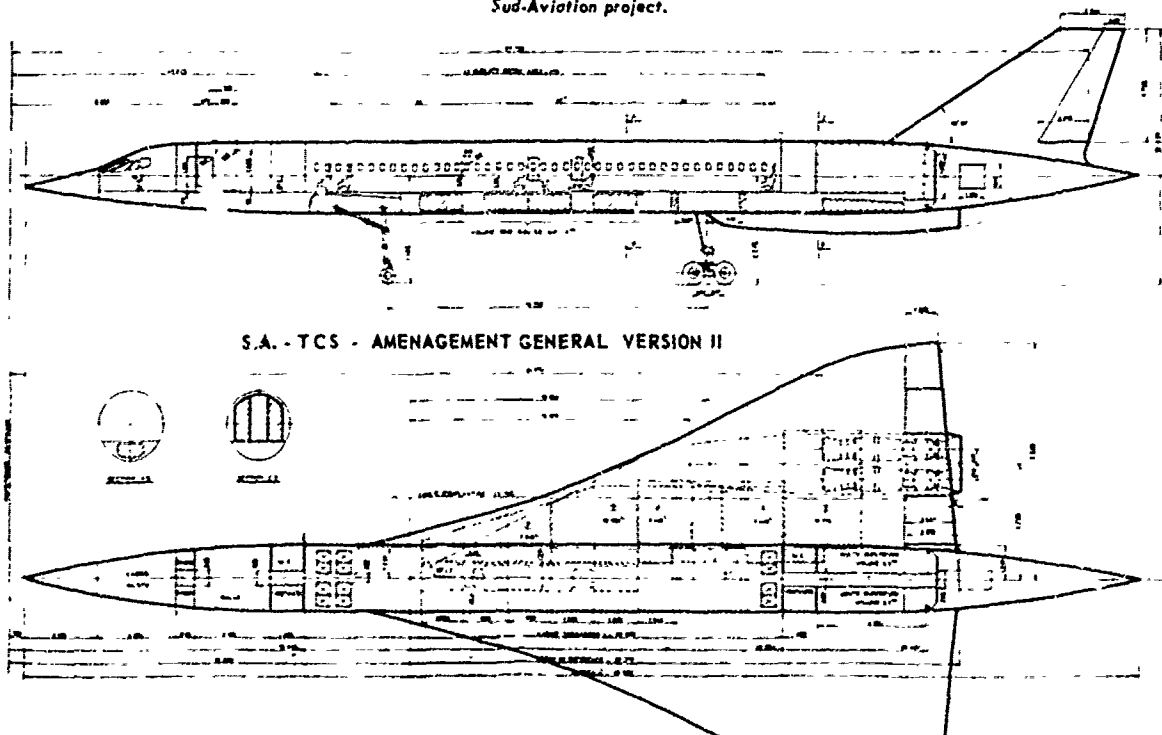


Fig. 33 - Avant-projet de Sud-Aviation pour un T.S.S. moyen courrier.

French Air Ministry - July 1961.
S.S.T. medium range preliminary proposals
Sud-Aviation project.



évolutif et nez basculant aux basses vitesses. Comme pour le projet de Sud-Aviation, cette forme découlait directement des recherches sur les ailes élanées menées dans les deux pays ; dès 1961, des accords bilatéraux B.A.C.-SUD et BRISTOL-SNECA étaient conclus pour étudier en commun un long courrier quadrimoteur directement dérivé des avant-projets respectifs ; enfin, la décision de construire des prototypes de ce qui était devenu Concorde fut prise par les gouvernements français et britannique en Novembre 1962.

Le lancement des avant-projets par le Service Technique français avait été accompagné d'une "mobilisation" des moyens de recherche et d'essais de l'ONERA au bénéfice des constructeurs :

- pour ce qui concerne l'aérodynamisme externe, des recherches théoriques entreprises pour l'optimisation de l'aile élanée conduisirent à proposer (fig. 27K) une adaptation de ses formes à Mach 1 dans le cadre de la théorie des corps élanés parce que, pour la mission demandée, il fallait que l'avion ait une bonne finesse, non seulement en croisière à $M = 2,2$ mais aussi en croisière subsonique éventuelle vers $M = 0,93$ (vol au-dessus des régions sensibles aux brags, cas de panne d'un moteur) et enfin en régimes de décollage et d'attente vers $M = 0,75$; les lois de vrillage et cambrure calculées par l'ONERA servirent de base à la génération de la voilure de Concorde.

À la même époque, l'ONERA mit en évidence, à la soufflerie de Cannes, un effet de sol favorable notable sur les ailes élanées qui s'ajoutait au gain lié au développement tourbillonnaire aux grandes incidences de décollage et d'atterrissage ; ce double avantage fut prouvé en soufflerie puis en vol par la NASA-Ames grâce aux essais de l'avion expérimental F-5D équipé d'une voilure ayant un contour voisin de celui de Concorde (fig. 36).

Enfin, il était vital d'être capable d'établir un bilan de traînée précis du projet dans tout le domaine de vol en extrapolant par le calcul les résultats obtenus en soufflerie à des nombres de

SUPER CARAVELLE
PROJET REIL DASSAULT
ENSEMBLE

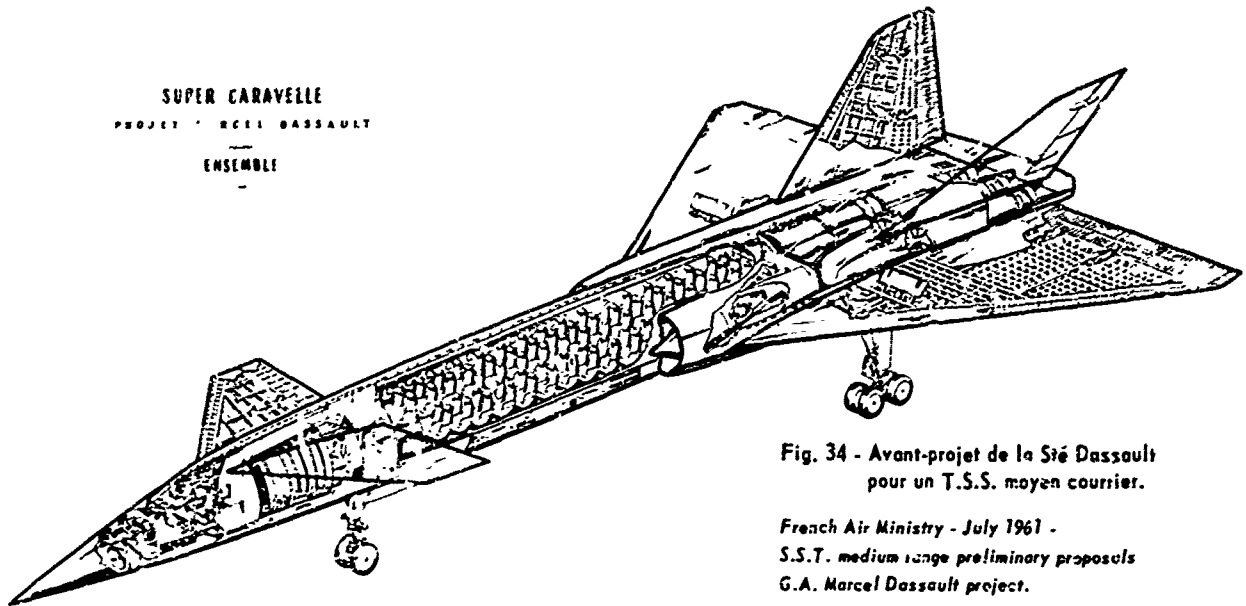


Fig. 34 - Avant-projet de la Sié Dassault pour un T.S.S. moyen courrier.

French Air Ministry - July 1961 -
S.S.T. medium range preliminary proposals
G.A. Marcel Dassault project.

Reynolds d'un ordre de grandeur plus faible qu'en vol ; le programme Concorde permit pour la première fois de se pencher sérieusement sur la mesure précise de la couche limite en soufflerie, en tenant compte de la position de la transition de la couche limite ou en imposant artificiellement l'écoulement turbulent sur les surfaces avec des rugosités calibrées ; grâce à des essais minutieux effectués sur une même maquette du projet (fig. 37) dans les grandes souffleries transsoniques disponibles en France, en Angleterre et en Hollande, il fut possible d'obtenir pour la première fois des corrélations correctes entre les différents laboratoires qui furent utilisés par la suite pour le développement de Concorde, et qui le sont encore actuellement pour d'incessantes améliorations sur l'avion de série.

Deux grands programmes de recherches fondamentales en soufflerie puis en vol furent également demandés à l'ONERA par le Service Technique pour vérifier la validité des calculs théoriques du frottement et de l'échauffement cinétique en fonction des nombres de Mach et de Reynolds ; une bonne prévision de ces caractéristiques était en effet vitale pour le calcul des performances et l'évolution de l'échauffement de la structure de Concorde en croisière supersonique.

L'influence des nombres de Reynolds et de Mach, ainsi que l'effet de la rugosité du revêtement de l'aile fut étudié par l'analyse de la couche limite sur l'extrados de l'aile delta du bombardier Mirage IV avec le concours de la Société Dassault (fig. 37D) entre $M = 0,9$ et $2,1$; des essais similaires sur une demi-aile delta en soufflerie aux faibles nombres de Reynolds permirent de vérifier convenablement les formules employées pour le calcul du frottement turbulent de Concorde dans un large domaine de nombres de Reynolds (fig. 38).

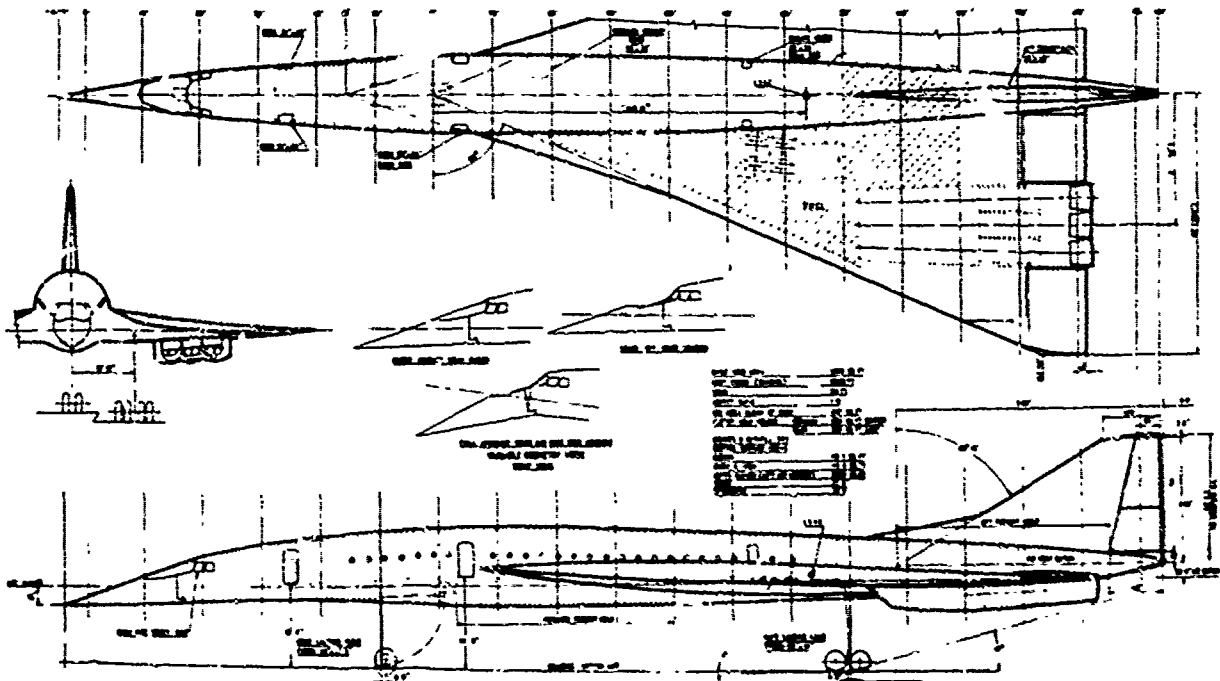


Fig. 35 - Avant projet de la B.A.C. pour un T.S.S. long courrier.
Bristol Aircraft Ltd S.S.T. project - 1961.

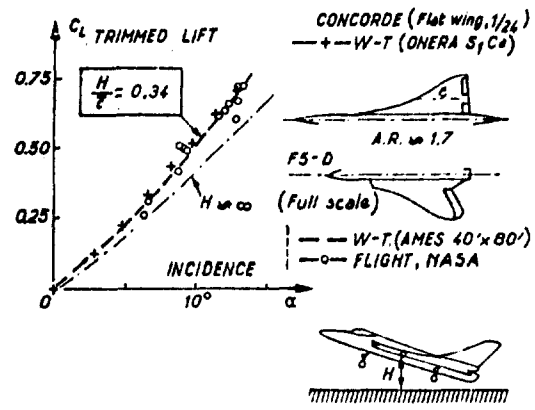
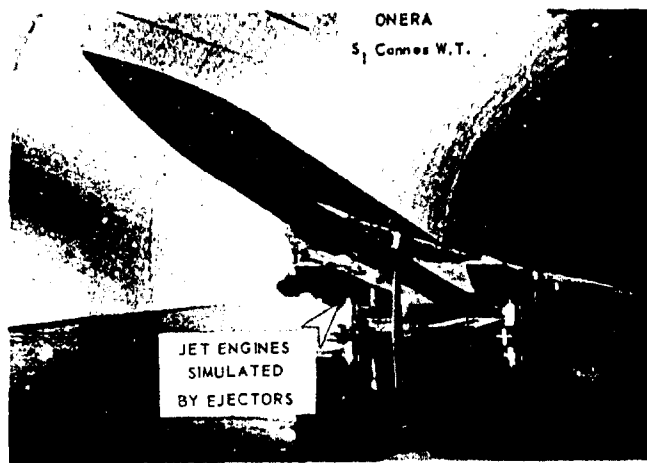


Fig. 36 - Effets favorables de la portance tourbillonnaire et de l'effet de sol pour une aile élancée du type «Concorde».

Vortex lift and favourable ground effect on slender wings.

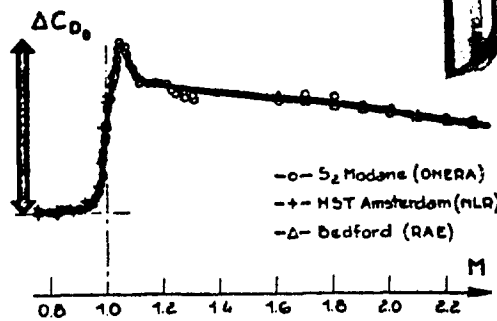
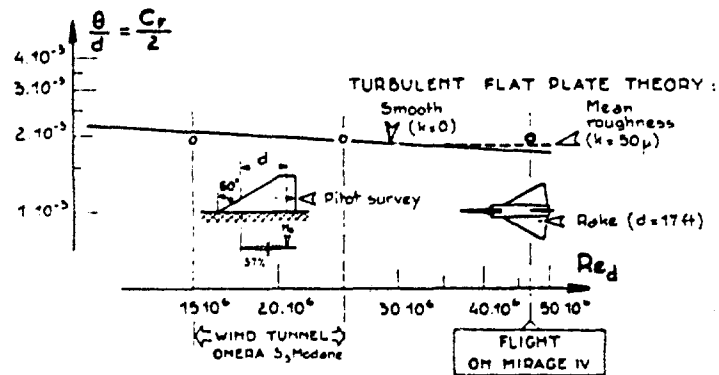


Fig. 38 - Traînée de frottement en supersonique sur une aile delta ; comparaison du calcul avec l'essai en soufflerie et en vol.

Mean friction drag at supersonic speed ($M_\infty = 2.15$). Correlation between theoretical estimation and experiments in W.T. and in flight.

Fig. 37 - Corrélation des bilans de traînée sur l'avant-projet de «Concorde» dans les grandes souffleries européennes.

Comparative tests on the transonic and supersonic drag on the Concorde preliminary design model (scale 1/30th).



L'étude expérimentale de l'échauffement cinétique posait des problèmes tout à fait nouveaux et difficiles de méthodes d'essais et d'instrumentation ; pour mener à bien cette recherche, une série de voilures delta (fig. 27H), ayant un revêtement en acier inoxydable parfaitement calibré et munies d'un grand nombre de thermocouples, fut essayée successivement à la paroi d'une soufflerie à rafales chaudes (S3 Modane) et sur un missile expérimental (D-6, fig. 39) ; plusieurs missiles furent tirés et les mesures de température télétransmises au sol, au cours du vol stabilisé au voisinage de $M = 2,2$ à 9 km d'altitude, permirent de calculer des flux thermiques en bon accord avec ceux déduits des essais en soufflerie ; par contre, ces valeurs expérimentales étaient sensiblement inférieures à celles prévues par le calcul basé sur le facteur classique de l'analogie de Reynolds, dont on put ainsi rectifier la valeur.

L'ensemble de ces recherches de pointe, financées sur le programme Concorde, permirent de former une pépinière de chercheurs et de techniciens et d'améliorer considérablement les méthodes d'essais au sol et en vol.

Pour terminer, je voudrais évoquer les importantes recherches d'aérodynamique interne qui furent entreprises dès 1958 à l'ONERA en vue du développement du système propulsif du transport supersonique.

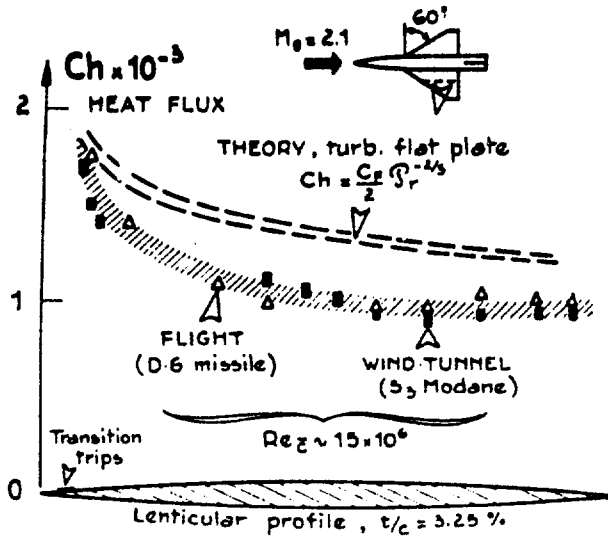


Fig. 39 - Recherches théoriques et expérimentales sur l'échauffement cinétique au régime de croisière supersonique de « Concorde ».

ONERA - Research on supersonic kinetic heating during Concorde study.

D.6 experimental missile (3 stages)

Une longue expérience avait été accumulée sur les prises d'air d'avions supersoniques militaires (familles Mirage III et IV) et de missiles expérimentaux à stato-réacteurs ; cependant, comme pour la cellule, les prises d'air d'un transport supersonique devaient être sensiblement plus performantes que celles des avions militaires contemporains pour obtenir la faible consommation spécifique indispensable à un grand rayon d'action ; le choix s'est porté dès le départ sur une prise d'air bidimensionnelle à géométrie variable ; calculs et expériences préliminaires au cours des avant-projets montrèrent rapidement que, pour la croisière à $M = 2,2$, et en profitant de la précompression favorable de l'aile lorsque la prise d'air est plaquée sous la voilure, la solution "compression externe" était plus simple et pratiquement aussi efficace qu'une solution "compression mixte" ; un banc d'essai original fut installé dans la soufflerie S5 de l'ONERA à CHALAIS, qui permit la mise au point des rampes de compression mobiles, ainsi que le piège à couche limite dont le rôle était fondamental pour obtenir un écoulement correct au droit du réacteur ; cette configuration fut adoptée d'emblée pour le projet Concorde, mais le dessin définitif a évolué au cours du développement du projet à la suite de très nombreux essais qui se poursuivent encore dans plusieurs souffleries anglaises et françaises, et pour lesquels l'ONERA est encore étroitement lié, aux constructeurs SNIAS/BAC. (fig. 40a : montage identique à celui de S5 ch, actuellement utilisé à Vernon)

En ce qui concerne les tuyères propulsives, le problème était encore plus difficile car nos connaissances et nos moyens d'essai étaient tout à fait insuffisants au moment de l'avant-projet ; il fallut d'abord développer des montages d'essais sophistiqués pour la mesure de la poussée de tuyères multi-flux dans le cœur d'un écoulement transsonique ou supersonique (fig. 40b) et également des bancs pour la mesure très précise de la poussée au point fixe ; tous ces montages servent encore pour améliorer

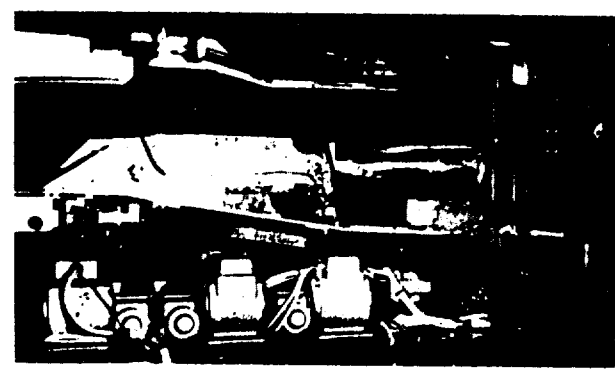
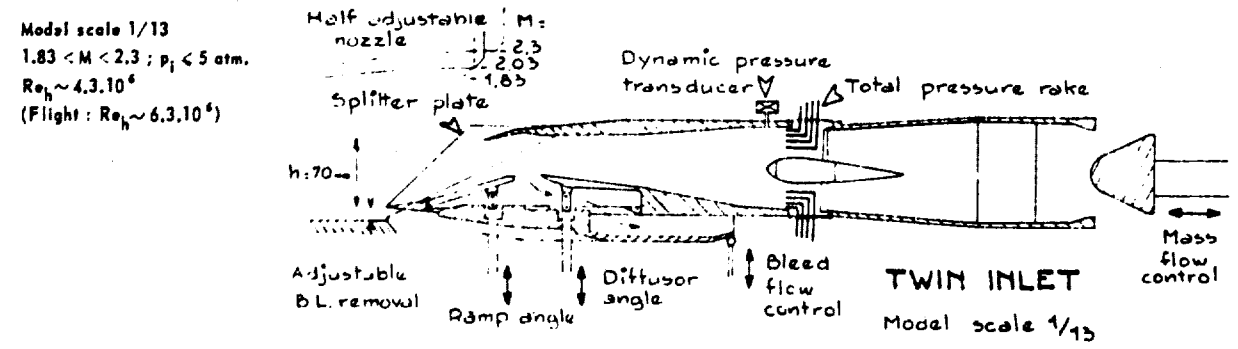


Fig. 40a - Montage expérimental pour l'optimisation des prises d'air doubles de « Concorde » dans la soufflerie à densité variable du L.R.B.A. à Vernon.

ONERA/LRBA (Vernon) - Wind tunnel tests on Concorde twin inlet at high Reynolds numbers.



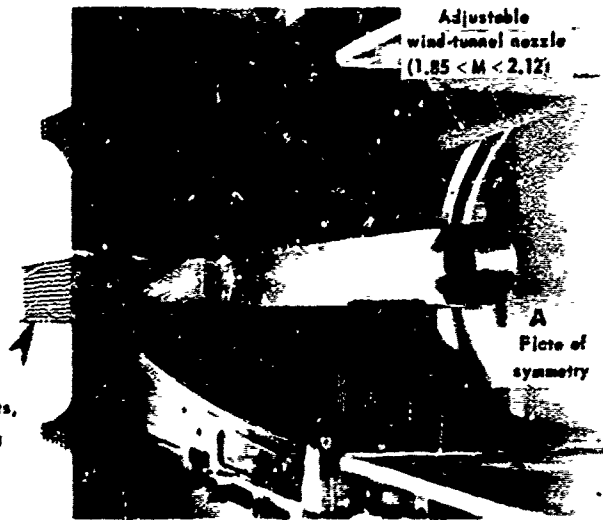
Model scale 1/13
 $1.83 < M < 2.3$; $p_1 < 5 \text{ atm}$
 $Re_x \sim 4.3 \cdot 10^6$
 (Flight : $Re_x \sim 6.3 \cdot 10^6$)

TWIN INLET
 Model scale 1/13

Fig. 40b - Montage expérimental pour l'étude des bilans propulsifs sur les tuyères de « Concorde » dans la soufflerie S5 de Chalais.

ONERA S5 wind tunnel Chalais - Thrust and drag on Concorde nacelle after-body (SNECMA).
Scale : 1/20.

Balance with 3 co-axial jets, in the settling chamber.



pas à pas les performances des tuyères de Concorde dans les différents domaines du vol, en collaboration étroite avec les constructeurs SNECMA/Rolls-Royce ; simultanément, l'ONERA participe aux études acoustiques, théoriques et expérimentales, destinées à réduire le niveau de bruit de ces tuyères.

Au cours du développement du projet Concorde, l'ONERA a également fourni une assistance technique continue aux constructeurs dans le domaine de l'aéroélasticité, particulièrement important pour cette configuration à voilure très souple : participation aux calculs de la structure, essais de vibrations au sol sur le prototype, étude du flottement sur maquettes aéroélastiquement semblables en soufflerie, analyse en vol de la réponse de l'avion à la turbulence, etc...

2.2.5 - Avion à flèche variable

Pour terminer, je voudrais évoquer brièvement le rôle que l'ONERA joue avant et au cours de l'avant-projet d'un avion multi-missions à géométrie variable :

En 1963, le Service Technique de l'Aéronautique demanda à l'ONERA une étude générale, à la fois théorique et expérimentale, sur le principe de la flèche variable et simultanément à la Société Bréguet, d'entreprendre une étude d'avant-projet sur un avion de combat multi-missions (Fig. 41) ; peu après une étude analogue était demandée à la Société Dassault qui devait être suivie de la commande à cette Société d'un prototype expérimental monoréacteur, le Mirage G, en octobre 1965 ; 25 mois plus tard, l'appareil entreprenait son premier vol et une grande partie de l'enveloppe de vol était explorée dans les deux mois qui suivirent (début 1968) ; le succès incontestable de ce programme et la rapidité de son exécution justifie le bien fondé d'une politique de prototype basée sur l'expérience acquise sur des appareils antérieurs ; la Société Dassault avait en effet repris, pour la mise au point d'un concept nouveau (aile à flèche variable), la plupart des éléments déjà éprouvés en vol sur l'avion expérimental à flèche fixe Mirage F₂ (même réacteur FW-1F-306, même fuselage avec prises d'air latérales et tuyère déjà qualifiées, empennage horizontal pivotant pratiquement identique, voir figure 8).

Du côté de l'ONERA, il s'agissait ici de fournir très rapidement des données aérodynamiques sur des configurations tout à fait nouvelles pour nous (les études étrangères étant alors classifiées), et ceci dans un domaine étendu de nombres de Mach.

Cependant, sur le plan théorique, nous avions acquis à cette époque une bonne maîtrise dans l'utilisation des analogies électriques Perès-Malavard pour le calcul des surfaces portantes en écoulement subsonique (fig. 42) ; par ailleurs, ce même laboratoire venait de mettre au point une méthode

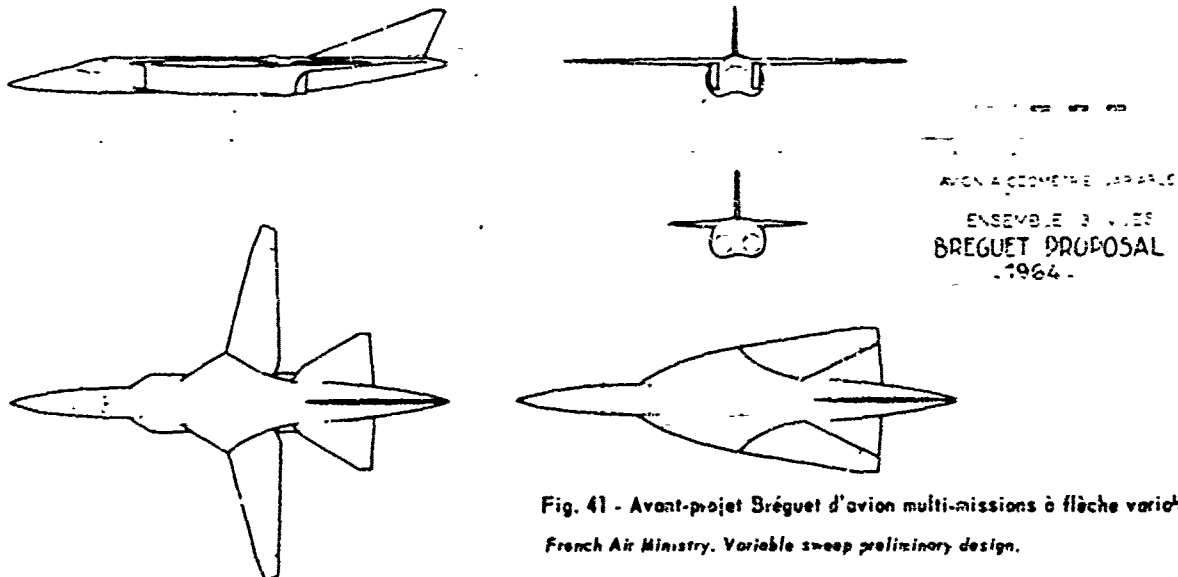


Fig. 41 - Avant-projet Bréguet d'avion multi-missions à flèche variable.
French Air Ministry. Variable sweep preliminary design.

analogique de calcul des ailes en supersonique basée sur l'utilisation d'un réseau inductance/capacité ; enfin, la Division d'Aérodynamique théorique avait développé à cette époque des méthodes éprouvées de calcul numérique, aussi bien pour l'optimisation d'une voilure à Mach 1, dans le cadre de la théorie des corps élancés, que pour le calcul des ailes de forme en plan quelconque en supersonique.

Sur le plan expérimental, il était indispensable de pouvoir vérifier rapidement en soufflerie la validité de ces différentes approches théoriques sur des maquettes schématisques ; on utilisa par exemple, pour obtenir les dérivées aérodynamiques, des familles d'ailes plates, dont la construction était rapide et bon marché, aussi bien en incompressible qu'en transsonique et supersonique (fig. 43).

Tous ces moyens de calcul et d'essais modestes furent utilisés simultanément pendant un an environ sur l'étude paramétrique d'une famille d'ailes à flèche variable ; on s'était fixé la même forme en plan en configuration déployée, et différentes positions du pivot assurant la rotation de l'aile mobile jusqu'à une flèche de 70° (fig. 44) ; l'étude analogique montra que le recul du centre de poussée entre les flèches extrêmes (15° et 70°) variait linéairement avec la position du pivot en envergure ; dans ces conditions, il existait une position excentrée de ce pivot conduisant à une stabilité longitudinale identique aux régimes extrêmes de fonctionnement (ailes déployées et repliées) ; toutes ces caractéristiques théoriques en incompressible furent rapidement confirmées par l'expérience dans la soufflerie de Cannes (fig. 43 a) ; ces essais permirent également de mettre en évidence des troubles de stabilité longitudinale aux grandes incidences avec les ailes déployées lorsque la surface de l'apex fixe devenait importante (cas d'un pivot excentré souhaitable par ailleurs pour une marge statique peu sensible à la mise en flèche).

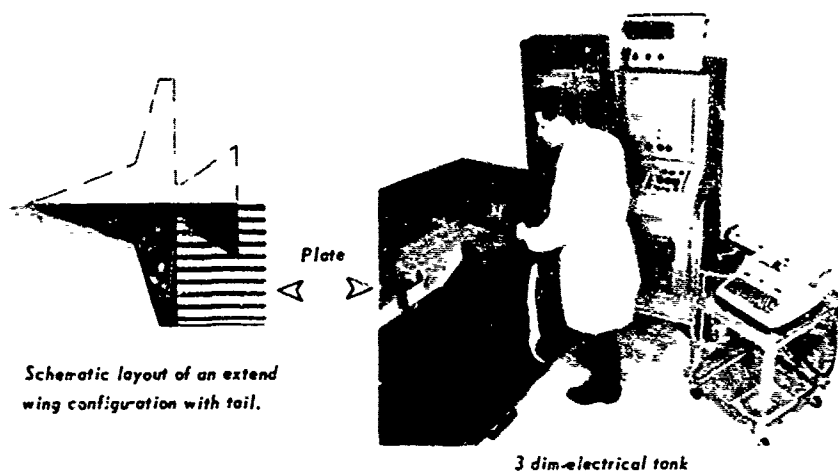


Fig. 42 - Utilisation des analogies électriques pour le calcul d'une famille d'ailes à flèche variable.

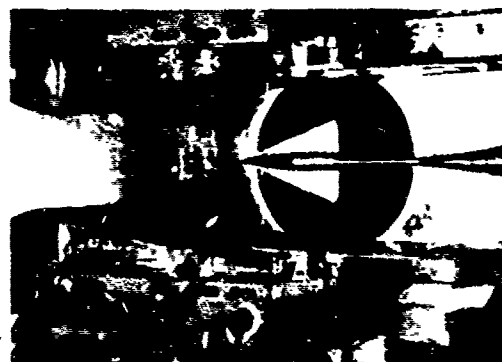
ONERA - Basic research on variable sweep concept. Theoretical approach with the electrical analogy method.



a) S_1 Cannes N.T. (low speed)

Fig. 43 - Essais préliminaires sur une famille de configurations à flèche variable pour vérifier l'étude paramétrique théorique.

ONERA - Preliminary tests to check theoretical approach on variable sweep aircraft concept.



b) S_5 Malais N.T. (trans-supersonic)

L'expérimentation en supersonique sur maquettes schématisées dans la soufflerie S5 de MALAIS confirma également les évolutions de centre de poussée avec la position du pivot en envergure, calculées en supersonique par les deux approches théoriques ; le recul du centre de poussée entre $M = 0,1$ et $M = 2,1$ (fig. 44) était sensiblement le même pour toutes les configurations étudiées avec ailes repliées à 70° , et donc indépendant de la position du pivot.

Finalement, l'expérience antérieure acquise à l'ONERA sur l'augmentation de la finesse maximale apportée par une cambrure concave au bord d'attaque d'une aile delta (Mirage III), nous conduisit à calculer une telle cambrure sur l'aile en position repliée (aile avec 70° de flèche, complétée par l'empennage horizontal, figure 45), ce calcul d'adaptation au bord d'attaque étant effectué à $M = 1$, dans le cadre de la théorie des corps élancés ; l'idée originale était de profiter à la fois d'une bonne adaptation de l'aile au combat transsonique (ailes repliées) et d'une forte cambrure en configuration "ailes déployées", qui devait permettre d'améliorer les caractéristiques de décrochage aux hautes vitesses. L'expérience en soufflerie, puis l'application en vol de cette approche théorique confirmèrent pleinement cet espoir.

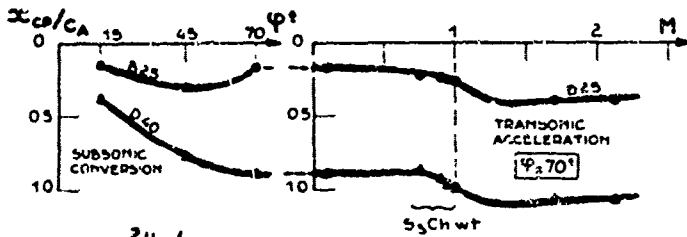


Fig. 44 - Influence de la position en envergure du pivot sur une famille d'ailes à flèche variable ; comparaison théories/expériences en sub et supersonique.

Basic theoretical and experimental research on variable sweep concept at ONERA before the preliminary design program.

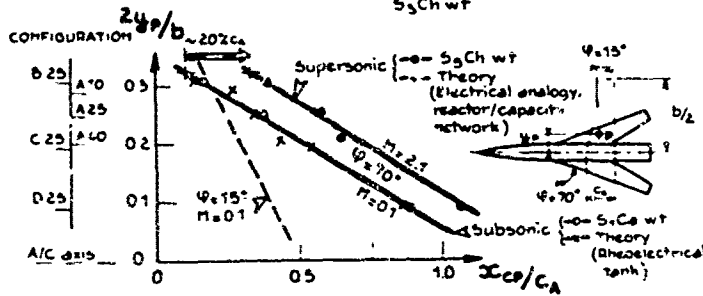


Fig. 45 - Etude théorique ($M = 1$) d'une cambrure conique au bord d'attaque sur l'aile repliée.

Theoretical approach for a L.E. conical camber on a variable sweep configuration.

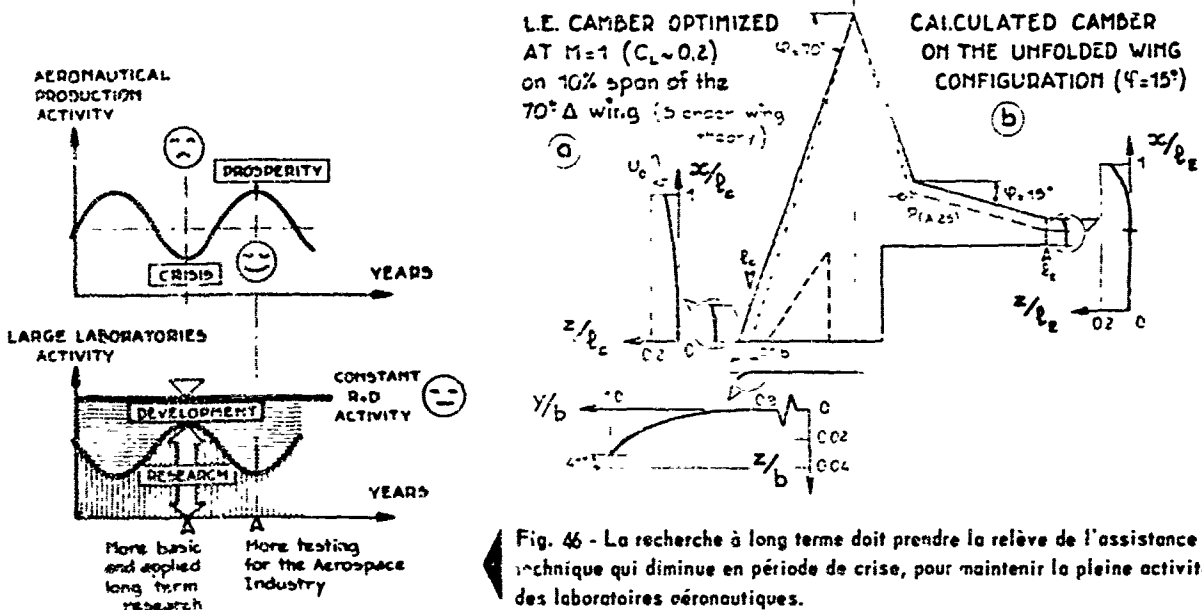


Fig. 46 - La recherche à long terme doit prendre la relève de l'assistance technique qui diminue en période de crise, pour maintenir la pleine activité des laboratoires aéronautiques.

Ce dernier exemple montre que le Centre de Recherches peut être efficace au stade de l'avant-projet avec un budget modeste grâce à un temps de réponse extrêmement court obtenu par la "mobilisation" de petits groupes de chercheurs, théoriciens et expérimentateurs, sur un objectif bien défini et en liaison directe avec les futurs utilisateurs.

2.3 - CONCLUSIONS

A partir d'un certain nombre d'expériences vécues en France au cours des deux dernières décennies, j'ai essayé de montrer qu'il avait été souvent rentable d'associer le laboratoire de recherche au lancement d'un avant-projet d'avion nouveau, le constructeur continuant ensuite à faire appel à l'assistance technique de ce laboratoire au cours du développement du projet et après les premiers vols.

Il faut reconnaître, et déplorer, qu'un tel processus est de plus en plus rare actuellement, dans la plupart des pays, pour plusieurs raisons qui sont d'ailleurs liées entre elles :

- raréfaction des projets nouveaux,
- pas de politique aéronautique à long terme, qui permettrait le développement d'avions expérimentaux,
- ampleur financière et complexité des quelques projets restants, qui sont alors confiés directement à des groupements de constructeurs, souvent multinationaux.

Cette évolution rend évidemment plus rare et difficile le dialogue direct entre services officiels, constructeurs et laboratoires au stade de l'avant-projet ; de plus, en raison du risque financier à prendre lorsque le projet n'est pas "conservatif", il est de plus en plus difficile d'introduire des idées nouvelles à ce stade ; c'est pourquoi le rôle des services Officiels doit être également de lancer des recherches générales propres à susciter des idées nouvelles longtemps en amont du lancement des projets chez les constructeurs, afin que ces idées soient directement applicables le moment venu.

Cette action "pilote" est particulièrement indispensable au temps des crises touchant cycliquement l'aéronautique, pour que le Centre de Recherche ne subisse pas ces caprices de l'activité industrielle (fig. 46) ; un programme de recherches à long terme permet de compenser la baisse momentanée de l'assistance technique et assurer le plein emploi d'un personnel hautement qualifié ; c'est également la meilleure solution pour que le laboratoire ait un court temps de réponse aux demandes des constructeurs dans les périodes fastes où plusieurs projets sont lancés simultanément.

COMPUTERIZED PRELIMINARY DESIGN AT THE EARLY STAGES OF VEHICLE DEFINITION

by

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SUMMARY

The conceptual and preliminary design processes are used to provide information regarding the feasibility and selection of various approaches to aircraft mission requirements. Decisions influenced by this information often have enormous cost implications at the later stages of the development process and during vehicle operation, yet the resources expended during the early phases are usually relatively small and distributed over several alternate approaches. The information provided during these early conceptual and preliminary design phases needs to be credible and complete, even though it must be generated with limited resources. This paper describes criteria for acceptance of early design information, modern methods of providing it and suggestions for defining adequate levels of resources to accomplish the objectives of the activity. Specific examples of the most difficult type of early design studies, which are those requiring significant undeveloped technology, are used to discuss these points. The examples include design studies and cost estimates of liquid hydrogen fueled aircraft, oblique winged aircraft, and remotely piloted vehicles.

INTRODUCTION

The preliminary design process, if used effectively, can save aircraft system costs during the development, acquisition and operation of future vehicles. The key to this saving is adequate exploration of the many design approaches suggested during the early definition of the vehicle concepts and the selection for further study of those concepts that indicate significant saving. In many cases, promising approaches are discarded prematurely because there are not sufficient resources to investigate many of them. Instead, there is a tendency to focus on the most conventional approach as a baseline and then to increase the design effort on this concept (or minor derivatives) to provide confidence in the approach. The result is a credible but unimaginative design that does not indicate the potential of other approaches.

How can this common occurrence in the early design process be avoided when the resources available for the conceptual and preliminary design phases are limited? One distinct possibility is to increase the efficiency of the design process at these early stages by the use of computerized and automatic methods. Design concepts can then be evaluated rapidly using accepted engineering computation methods. A considerable quantity of valid information from this process can then be used to help select the most promising concepts for additional study and design.

Most agencies and companies have developed computerized tools for preliminary and conceptual design (Refs. 1 through 9); these programs have been used to a large extent in recent years in providing information for future vehicle concepts (Refs. 10 through 14). These programs usually take the form shown on the first figure, which indicates the individual disciplines required in the aircraft early design process. In each discipline the design function is mathematically modeled to an appropriate level and the results integrated to provide a vehicle definition for a specific mission. This general arrangement is common to many of the programs, but the method of interconnection, the level of detail and the emphasis on particular aircraft types are used to distinguish between these programs. The Ames Research Center, NASA, aircraft synthesis program, called ACSYNT, uses the common arrangement shown in Fig. 1 and will be used here to help describe the general features in computerized preliminary design programs and the trends toward improving their usefulness, efficiency, and credibility. Suggestions are offered regarding the approximate level of resources required to develop and utilize these types of programs.

DESIGN LEVELS

Prior to describing the characteristics of the computerized preliminary design process, it is important to understand the objective and scope of the activities at the early stages of vehicle definition. The primary purpose at these levels is to provide technical and economic feasibility information for guiding larger efforts during the detailed design phases. It is important to realize this purpose and not confuse it with the detailed or final design process, where the objective is to provide enough information to build the vehicle or to develop a careful plan to build it. In order to help define the appropriate level of detail to meet the objectives at the early design stages, Fig. 2 shows the definition, used at Ames, NASA, of four design levels in aircraft development.

The conceptual level is characterized as the idea stage during which form, cost activity, component placement, and approximate size are defined. At this stage, limited engineering calculations are performed to provide a three-view drawing and cross sections showing approximate placement and size of all the major vehicle components. An equally important output is a list of attributes based on physical principles that suggest the reasons this system has potential. This list and the physical principles help the advocate of an idea to define the source and limits of the potential improvements that are expected from his idea. This stage usually requires little resources and yet is a primary source of ideas. Significantly increased resources are needed to evaluate these ideas and compare them with other approaches. Emphasis in the conceptual stage is on clear definition of the idea and the theoretical or empirical basis for suggesting why the concept has merit. Little emphasis or reliance is placed on the simplified and isolated computations that may suggest what the quantitative performance of the concept is.

At the next design level, preliminary design, these concepts are tentatively evaluated with simplified and rapid engineering computations in a balanced effort. At this point, the emphasis is on the quantitative

measures of performance in order to make comparisons. For these computations, standard methods of analysis are used and historical correlations are relied upon for validity. However, with concepts that are novel or unusual, the existing engineering approaches may not be well suited, especially when they rely on empirical approaches that do not match the study concept closely. Under these circumstances, significant theoretical study is required to validate the performance of the design. This key area will be discussed later in describing the resources needed in the computerized design process.

Figure 2 indicates that the output from the preliminary design process provides, again, a three-view drawing showing computed shape, size, and connectivity as well as computed multi-element weight and cost statements. The results also include description of the computed aerodynamic, structural, and propulsive performance parameters. Note the emphasis is on computation of the results, since this quantified information is used to compare with other results in the selection process. At this level of design, analytical calculations (both theoretical and empirical) are usually relied upon entirely; however, these need to be based on experimental correlations and information from past or current experimental research programs. In the preliminary design stage as defined here, no experimental work is directed to design verification. This work is left for the detailed design stage.

Figure 2 indicates that the detailed design stage emphasizes accurate and detailed investigation of a fixed configuration but does include tradeoff studies to improve and refine the vehicle performance. Major design concept changes usually are not permitted unless conducted in parallel detailed studies. This level of design usually is augmented by experimental verification in the aerodynamic and possibly critical structural areas. The major output of this activity is the detailed assembly drawings with specific definition of the major components and a careful plan indicating the schedule and resources required for the development, acquisition, and operation of the vehicle.

In the final design level, the objective is to provide the component and assembly drawings for construction of the vehicle. An integrated computerized process to perform this activity has been studied (Refs. 15 and 16), but has not been implemented. Of course, mechanized drafting and preparation of tapes for numerically controlled tools are done at present on a nonintegrated or isolated basis. These functions are key elements in automated design that may be expected to evolve into integrated final designs in the future.

PRELIMINARY DESIGN STUDY REQUIREMENTS

The focus of this paper is on the conceptual and/or preliminary design level and not the detailed and final design level as defined above. What are the requirements for effective computerized conceptual and preliminary designs at these early stages of vehicle definition? Figure 3 indicates the main features or characteristics found in these programs. First, the programs usually are extremely modular so that the functions to be performed are clearly separated and can be assembled to perform various sequences of computations, depending on the problem. Modularity is very important since each module of a synthesis program should be verified on an individual basis prior to integration with others. Modularity also greatly simplifies data transfer and error identification during program development and operation. Each module provides the intermediate data from a particular discipline that are required by other modules as well as the final data from that discipline that are needed for adequate vehicle definition.

The concept of extreme modularity actually leads to more simplification in obtaining a truly integrated design procedure. A control program is used to exercise the individual modules and the data from these modules are transferred to other modules for additional computations. In many computerized design processes, transfer of input and output between programs (integration) is not well developed; in some cases, it is performed manually. Integration of the individual modules that make up a program can be accomplished in either of two ways. First, it is possible to connect the basic analytical modules that estimate the information in a particular discipline; for example, aerodynamic modules may compute lift and drag directly from vehicle geometry. Another method is to use these modules on an isolated basis to develop scaling information for estimating changes in a precomputed baseline concept. In most cases, the scaling programs can be considerably simplified and require minimum resources to develop or operate. However, the resources necessary to perform the baseline design and then develop the scaling relationships are the major part of the whole effort. The total resources, both engineering man-hours and computer time, necessary to perform a design using the scaling approach actually are comparable to the direct method. In addition, the scaling relationships usually are simplified by omitting parameters of secondary importance, whereas direct use of basic modules to compute performance in each discipline assures the incorporation of all parameter effects as accurately as they are being modeled.

Design programs that use scaling information are well suited to computations in the later stages of design, where computations in each of the disciplines becomes much more detailed and time consuming, and yet updates on the configuration design are required on an almost continuous basis. However, at the early stages of preliminary design, direct transfer of information between programs that provide basic data can be a more flexible and usable approach than the use of scaling programs.

Another characteristic of aircraft synthesis programs is the capability to provide calculations at different levels of detail. Simplified computations are required to estimate the initial characteristics of the vehicle so that more detailed computations in each discipline area can proceed. In an efficient integrated program the selection of the appropriate levels is under control of the designer. At Ames Research Center, these design levels are characterized by the descriptions given in Fig. 4. Note that at each level there is sufficient information for a completed design in terms of the quantitative measures indicated earlier. The primary reason for increased level of detail is to incorporate more credible methods. The computation time and cycle time required to complete a vehicle design increase significantly with the level of detail in the computations. Since a major requirement in preliminary design is the assessment of many ideas that have been selected for study, it is particularly important that these alternate approaches be compared rapidly. A guideline for the development of the Ames, NASA, ACSYN program is that a vehicle analysis in Level I requires less than one minute of computer time (IBM 360/67) and in Level II, requires less than five minutes. A vehicle studied to design convergence usually requires two to five vehicle analyses. Design convergence means that the vehicle performs the specified mission within a

stated tolerance and that all computations are performed on essentially the correct configuration geometry and weight. Unless design convergence is accomplished, results for different approaches cannot be compared closely.

An important required output in the computerized preliminary design process is sensitivity information that permits an assessment of critical areas. Extreme sensitivity can lead to discarding a design or indicate the need for technology improvement before the design can be considered feasible. Figure 5 shows an example of the sensitivity information provided by the ACSYNT program during the assessment of an oblique winged transonic transport at Ames Research Center. The oblique winged airplane (Ref. 17) is a unique approach to high aerodynamic efficiency at cruise and good low-speed characteristics. Sensitivity is defined as the percentage change in a performance measure due to a percentage change in the parameter (i.e., the percent change in gross weight due to a percent change in a mission or a vehicle parameter). There are three categories of parameters for which sensitivity information is usually desired. The first category includes the mission parameters such as range, payload, turn rates, endurance, etc. The second includes the vehicle design parameters such as wing loading, wing aspect ratio, body fineness ratio, etc. The third category includes the efficiency parameters such as engine compressor efficiency, span load efficiency, minimum drag coefficient, etc.

Sensitivity to mission parameters indicates the ease or difficulty with which the general concept can perform the specified mission, that is, feasibility. Sensitivity to parameters in the second category provides information for optimizing the vehicle, since this is essentially gradient information indicating the direction of improved performance measure.

Sensitivity information from the third category is used to identify overall improvement in the vehicle due to changes in the technology of a specific area. This type of information helps focus research on the most significant areas.

Sensitivities which can be provided automatically by the ACSYNT program are difficult to obtain in manual design processes and therefore usually are not provided. In computerized early design they are readily computed and should be required as part of the study results.

Another important requirement in computerized preliminary design is the need to provide optimization and tradeoff type calculations that result in the best performance for a vehicle concept. Figure 6 shows the results of an automatic seven parameter search for a minimum gross weight airplane design to perform a remotely piloted research vehicle mission. This remotely piloted vehicle was constrained to a specified level of performance in terms of sustained turn rates at two different Mach numbers and altitude conditions. The optimization process selected the best combination of wing area, sweep, thickness ratio, taper ratio, vehicle thrust-to-weight ratio, body diameter, and body fineness ratio to perform the mission with its constraints. Note that the design variables did not change significantly, yet the vehicle weight changed by 27%. Unless these types of optimizations are performed, the study concepts have poorer performance than necessary, which makes comparisons less valid. The code to perform this constrained minimization (Ref. 18) has been applied to many different types of engineering computations, and is useful in suboptimization in the individual discipline modules.

Another example of the use of the optimization process is shown in Fig. 7 in which a liquid hydrogen fueled hypersonic aircraft was optimized for maximum passenger load by changing the geometric characteristics of the vehicle (Ref. 11). The vehicle propulsion system also was optimized automatically; in this case, the use of an automatic procedure was extremely important, since the problem complexity would have required considerable engineering time to resolve by manual methods. Here, the propulsion system consisted of three engines: a ramjet; a turbojet; and a rocket. The problem was to maximize the passenger load subject to a sonic boom constraint. Optimization was accomplished by manipulating the size of each of the three propulsion systems, which operated simultaneously in the transonic region, and determining the appropriate time for the rocket to ignite and shutdown. This optimization permitted the computation of maximum passenger loads for a given gross weight vehicle as a function of the sonic boom constraint as shown in the figure. For unusual aircraft concepts such as in this example, it is very difficult to select combinations of design parameters on the basis of intuition; automated optimization is therefore essential.

CRITERIA FOR PRELIMINARY DESIGN INFORMATION

The computerized preliminary design process should include the information described earlier, that is, final information from each of the discipline areas at various levels of detail, sensitivity of the vehicle to mission and vehicle parameters, and optimization of the vehicle geometry and design parameters. In addition, these results should be presented so that the user has an assessment of the accuracy and the credibility of the information. An effective means of providing this credibility is to show the results in combination with correlation attempts in the individual module or discipline areas. For example, in the aerodynamics area, correlations between the analytical results and experimental data should be provided as part of the preliminary design process output. Figure 8 shows the correlation of the estimated high angle of attack aerodynamic characteristics as predicted by analytical methods, and also shows a comparison with experimental results from a study aircraft. These comparisons can be used to provide statistical information regarding standard deviations and probable errors that indicate the accuracy expected from each of the discipline areas. This accuracy information, when combined with the sensitivity information described earlier, gives an overall assessment of the accuracy of the results of the studies. For example, if the probable error in estimating drag at high lift is 10% and the vehicle weight sensitivity to this parameter is 0.7, then the probable error in gross weight due to drag at high lift is 7%. This type of information should be provided routinely to the users of preliminary design results so that they are not misled by small differences between alternate design concepts.

An additional means of providing credibility for the computerized design process is to compare the total integrated computation results for a specified mission with an actual aircraft designed for that mission. Figure 9 shows the results of a comparison between the ACSYNT program and a Boeing 727-200 aircraft. In this case, the inputs to the ACSYNT program were the following: the seating arrangement; the wing, body and tail geometry; the engine characteristics in terms of design bypass ratio, pressure ratios, turbine

inlet temperature, etc.; and the one-dimensional cycle efficiency parameters for the engine components. The results indicate adequate correlation in terms of the geometry and total weight of the system; however, individual elements in the weight statement do not agree well and suggest that further investigation is required. It is necessary to perform these correlations continually at the total aircraft level to check the analytical methods, since they are under continuous development or enhancement. The results of these correlation studies or their references should be included as part of any preliminary design study.

RESOURCES REQUIRED IN COMPUTERIZED PRELIMINARY AIRCRAFT DESIGN

The resources needed to perform the computerized process at the early stages of vehicle definition are primarily engineering man-hours and computer time. By far the most costly and important resource expended during this process is engineering time, even when the process is highly automated. Engineering time is usually divided between program development or modification and the preparation and analysis of input and output data.

The development and modification of modules to compute design information in the various disciplines has been underway for several years and in many cases the modules or methods are available. However, most of these modules require continual enhancement and further development to apply them to the variety of concepts studied in preliminary design. Unusual or unique designs require the development or modification of existing methods to provide good results. An example is the oblique winged aircraft that was mentioned earlier. This vehicle (Fig. 10) operates with increasing wing sweep as speed increases and the changing geometry and antisymmetry of the configuration required significant modification to existing aerodynamic modules.

Figure 11 shows a liquid hydrogen fueled aircraft (Ref. 10) with a tankage arrangement that had no historical counterpart or empirical data base and hence required the development of specialized theoretical computer programs to investigate the structure. The tanks in the vehicle support all the vehicle loads and are pressurized. These examples of unique or unusual configurations indicate that a majority of the resources necessary to perform computerized preliminary design are spent in the investigation of unusual or unique features. Conventional design features can be studied with existing modules in many cases.

At the present time considerable resources are spent during computerized preliminary design in the preparation of input and output information. This area can be made much more efficient. Digital computer technology to automate the input data function and to display engineering results has reached a mature point of development and should be included in efficient preliminary computerized design processes. Computer terminal hardware (Fig. 12), which may be located directly in the preliminary design work area, consists of cathode ray tube devices for display of the information and input devices of various forms, including keyboards and digitizing electrical tablets. In the last few years, the price of these devices has been reduced significantly and they can be connected to large-scale computers through telephone or data communication networks. The key element in the use of these devices is the time-shared operation of the main computer which permits access to substantial computer resources without committing the computer and its expense during periods of no computation. The software to support these types of computer systems is substantial, but has been undergoing development and enhancement for several years. At the present time these systems are operational and performing satisfactorily, although the rate of technological progress (and obsolescence) is relatively high.

The figure shows the computer terminals at Ames, NASA, that connect to a time-shared central computer with sufficient power to perform large-scale engineering computations (Ref. 19). The vehicle geometry displayed on the tube was defined from the tablet and sketch shown in Fig. 12. The output of configuration drawings (Fig. 13) and engineering results (Fig. 14) from this equipment is suitable for inclusion directly in publications and the generation of the information is becoming automatic.

The resources and time needed to initiate or expand a preliminary design activity to the point described in this paper are significant. The modules used in the ACSYNT program represent more than 100 man-years of development and enhancement over a period of more than ten years. The program is capable of analyzing conventional transport and fighter type aircraft, but is still in a state of rapid evolution as new capabilities and theoretical methods are continually incorporated.

The specialized personnel needed to integrate and develop the programs to perform this type of design activity are the major factor in the success of the activity. Each of the technical discipline areas is complex and the development and use of efficient programs in these areas require the dedication and skill of technical experts. Professional engineers and scientists with proper training are needed to insure that the methods and correlations required in the preliminary design activity are sound. Attempts to center these activities around personnel skilled only in computer programming have not been effective. It also should be pointed out that it is difficult to pursue this process on an ad hoc or part-time basis, since as stated before, continual development and correlations between analytical methods and experimental information are required. The high growth rate in computer technology has permitted a corresponding improvement rate in the analytical methods used to perform preliminary design. Hence, the rate of development of new analytical methods and the obsolescence rate of old ones are sufficiently great to require the continued attention of professional specialists.

Once an operational capability as described above has been achieved, the computerized preliminary design of new but relatively conventional concepts is anticipated to require only modest resources and time. The goal for the ACSYNT program is to provide all the results described earlier (including optimization and sensitivities) for a single vehicle in less than one week at Level I and in less than one month at Level II. Approximately six professionals and two hours per day of computer time are anticipated during the activity. Of course the objective in early design studies is to compare many approaches. If several aircraft concepts with similar missions and features are studied, then the computerized design process can investigate several vehicles much more readily than manual methods can. The advantage depends on the similarity of the vehicles and their components.

CONCLUDING REMARKS

The computerized preliminary design process can provide significant information that may influence the future costs of aircraft development and operation. This information can guide the design process toward concepts that have significant promise. The early study results can be based on established engineering methods that are adapted for computerized techniques and integrated for rapid use. The preliminary design process needs to be rapid and efficient in order to provide enough credible information to make sufficient comparisons. The results should include correlations with existing experimental data and should provide optimization and sensitivity information.

The primary resource required to perform this activity is engineering time which is best directed toward the development of methods and the analysis of unique concepts. The preparation of input data and the manipulation of output information for rapid review and analysis should be highly automated. The computer technology is available to modernize and automate much of this early design activity, but the total process is still entirely dependent on professional specialists using the most modern methods and equipment.

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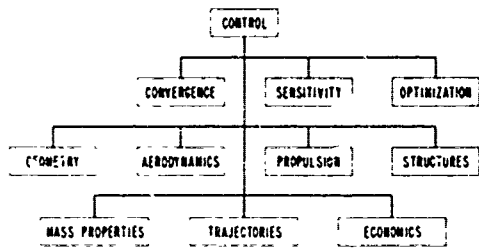


Fig. 1. Aircraft synthesis program disciplines.

- | | |
|-------------------------------|------------------------|
| GEOMETRY | STRUCTURES |
| I STANDARD COMPONENT SHAPES | I BEAM THEORY |
| II ARBITRARY COMPONENT SHAPES | II FINITE ELEMENTS |
| AERODYNAMICS | MASS PROPERTIES |
| I COMPONENT DRAG FORMULAS | I 25 ELEMENTS |
| II FINITE ELEMENT PANELING | VOLUME AND C G |
| AREA RULING | II >100 ELEMENTS |
| | INERTIAS |
| PROPULSION | TRAJECTORIES |
| I 1-DIM CYCLE ANALYSIS | I INTEGRATED |
| INSTALLATION EFFECT | FLIGHT SEGMENTS |
| FORMULAS | II OPTIMIZED FLIGHT |
| II TABLE INTERPOLATION | SEGMENTS |
| (MANUFACTURERS DATA) | |

Fig. 4. ACSYNT preliminary design levels.

	CONCEPTUAL	PRELIMINARY	DETAIL	FINAL
OBJECTIVE	DETA DEFINITION • 3-VIEW SKETCH • ATTRIBUTES • PRINCIPLES DEFINED	COMPARISON • COMPUTED CHARACTERISTICS (3-VIEW DRAWING) • MANY CONCEPTS • FEASIBILITY INFORMATION	RESOURCES PLAN • COMPONENT DEFINITION • ASSEMBLY DRAWINGS • PLAN • SCHEDULE • CONTRACT	CONSTRUCTION PLANS • SHOP DRAWINGS • MATERIALS LIST • PROOF TEST
APPROACH	DRAWINGS • COMPONENT SHAPE • PLACEMENT • CONNECTIVITY CALCULATIONS • LIMITED • ISOLATED	RAPID CALCULATIONS • BALANCED EFFORT • SENSITIVITIES • OPTIMIZATION	ACCURATE METHODS • EXPERIMENTAL EFFORTS • MOCK UPS • DETAILED CALCULATIONS	FORMALIZED METHODS (CERTIFICATION) • TEST PROGRAM • ESTABLISHED CALCULATED METHODS • ALL FLIGHT CONDITIONS CHECKED

Fig. 2. Definition of aircraft design levels.

SENSITIVITIES
OBLIQUE WING TRANSPORT

WEIGHT SENSITIVITY = $\frac{\Delta \text{GROSS WEIGHT/GROSS WEIGHT}}{\Delta \text{PARAMETER/PARAMETER}}$

MISSION PARAMETER	NOMINAL VALUE	WEIGHT SENSITIVITY
PASSENGERS	200	.600
RANGE	3000 n mi	.691
CRUISE MACH NO	1.15	.906
DESIGN PARAMETERS		
WING LOADING	120 psf	-.263
NOSE FINENESS RATIO	10	.094
WING ASPECT RATIO	12	-.286
WING THICKNESS RATIO	10	-.0643
BYPASS RATIO	2	.138
EFFICIENCY PARAMETER		
MINIMUM DRAG COEFFICIENT	-	.772
COMPRESSOR EFFICIENCY	-	.713

Fig. 5. Typical sensitivity information.

- MODULAR
- INTEGRATED
- MULTILEVEL
- SENSITIVITIES
- OPTIMIZATION

Fig. 3. Computerized preliminary design program features.

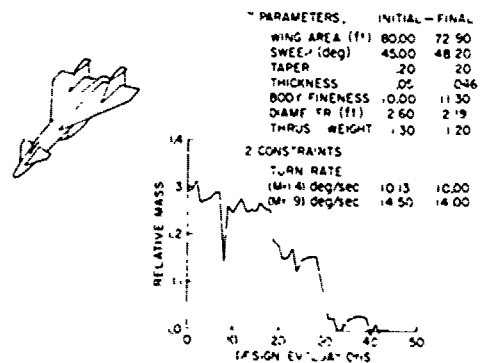


Fig. 6. Typical optimization results for a remotely piloted research vehicle.

RANGE = 5500 n mi
GROSS WEIGHT = 500 klbs

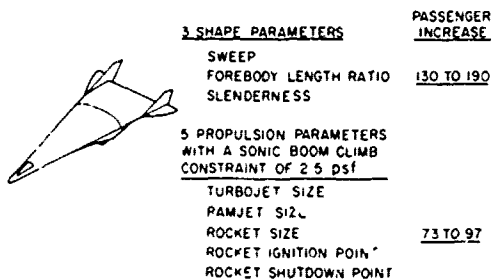


Fig. 7. Optimization results for a hypersonic aircraft.

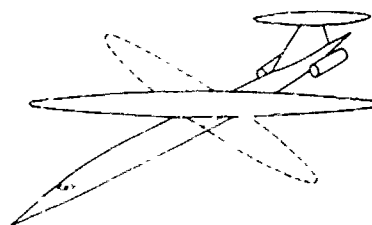


Fig. 10. Oblique transport configuration.

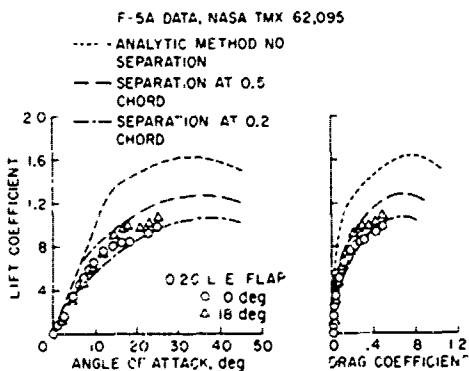


Fig. 8. High angle of attack aerodynamic estimating correlations.

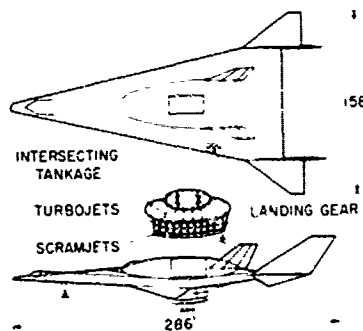


Fig. 11. Liquid hydrogen fueled airbreathing launch vehicle.

AIKRAFT - BOEING 727-200
MISSION - PAYLOAD = 163 PASS, RANGE = 2000 N M, M = 0.82

	DIMENSIONS			WEIGHTS			
	COMPUTED	ACTUAL	% ERROR	COMPUTED	ACTUAL	% ERROR	
BODY LENGTH, FT	137	131	5	AIRFRAME	54700	53700	2
BODY WIDTH, IN	139	139	0	PROPULSION	11200	12700	-12
WING SPAN, FT	110	108	2	FIXED GEAR	30000	28300	6
HORIZ TAL SAM, FT	40	36	11	FUEL	46000	42700	10
NACELLE LENGTH, FT	20.8	17.8	17	PAYLOAD	32600	32100	0
NACELLE DIAM, IN	44	45	2	GROSS	175000	170000	3
WING AREA, SQ FT	1700	1587	7				
H TAL AREA	469	376	25				
V TAL AREA	474	356	33				

Fig. 9. Overall vehicle synthesis correlation.

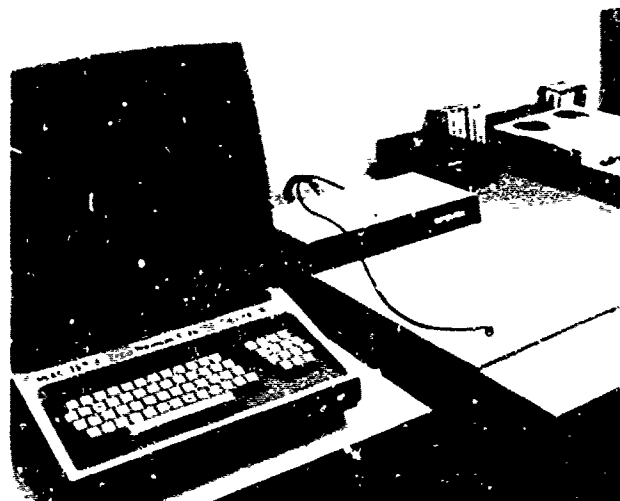


Fig. 12. Computer terminal with CRT, keyboard, and digitizing tablet.



Fig. 13. Digitized vehicle configuration.



Fig. 14. Display of engineering computations and vehicle geometry.

PRELIMINARY DESIGN TECHNIQUES FOR UNMANNED, REMOTE PILOTED VEHICLES

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1. INTRODUCTION

RPVs in preliminary design phases – where they currently are – are a great challenge for engineers. To understand these new emerging RPVs somewhat better it seems worthwhile to start from the well-known drones. Explaining the main differences Fig. 1 should give an impression of the higher complexity of RPVs in comparison with drones. If we understand drones as the mother of RPVs there are two fathers, manned A/C and missiles (Fig. 2).

At first sight there are several reasons for using RPVs in future missions (eg. fighter- Recce- and EW-missions) – in addition to saving pilots –:

- Cost reduction
- Higher effectiveness (due to maneuverability/the computer in the loop)
- Higher flexibility concerning future mission requirements (modular mission equipment)

But weighing and balancing cost, effectiveness and flexibility requires a closed loop, working in threat and mission analysis, technology, system-engineering as well as in the operational area (command/control, logistics, training ...).

RPVs, as military systems with the highest degree of automation ever developed, will probably generate resistance or at least delaying action in the part of operations personnel, which can lead to serious gaps in the defence capability especially if the opponent pushes forward RPVs with high mission effectiveness. In any case, it can be expected that 'RPVs will not do it best' in every aspect, but rather that manned/unmanned missions in combination and in mutual supplementation will be the answer.

Optimizing future manned/unmanned systems requires the closed loop cooperation mentioned above. The starting point in this loop can be at any place you like. The RPV study conducted by VFW-FOKKER, some results of which are explained in the written paper, was planned only as a part of that loop, to give some inputs to the technologist and operations personnel and to start discussions.

Objective of this study was:

Investigation of a multirole AS/CAS RPV
(Air superiority/Close air support-RPV)

based on

unified air vehicle
modular mission equipment and
specific armament

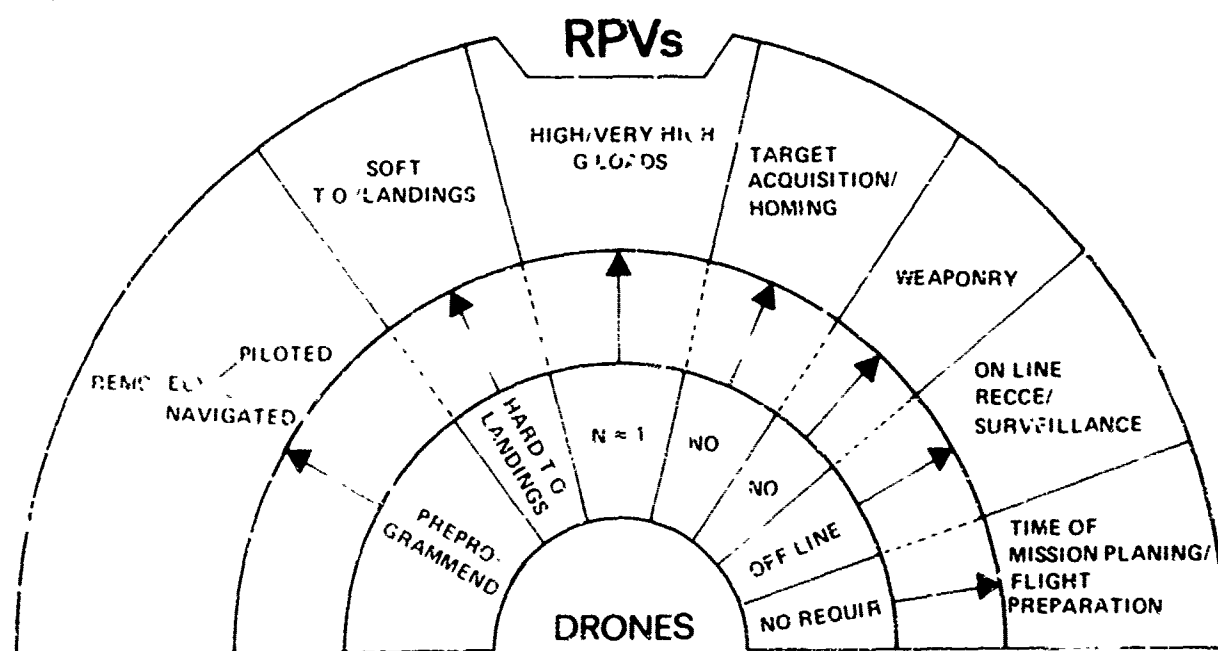


Fig. 1 RELATION RPVs-DRONES

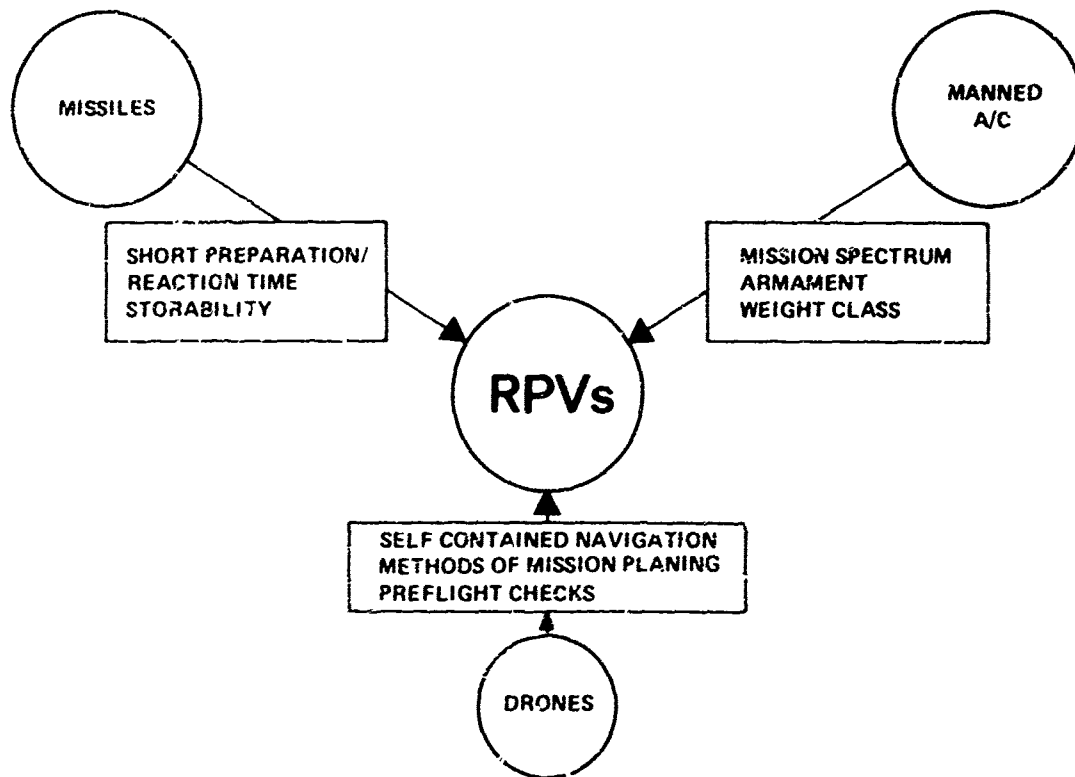


Fig. 2 RELATIVES OF RPVs

2. THE MULTIROLE RPV-STUDY

2.1 Steps in conducting the study

Figure 3 shows the different steps through which the study passed. Evaluating the FRG threat scenario, Air-Defense (AD) and CAS were found to be the predominant missions. AS-mission as the most stringent AD-mission was studied to fix the requirements for air vehicle, weaponry, and avionics. With respect to CAS, the available airfield lengths and the ground to air threat defined minimum mission requirement. A point design has worked out to present effectiveness – payload/range characteristics and to give a first impression of such a RPV-system, considering the vehicle itself, mission equipment, armament, and supporting systems.

The fallout of a point design for different missions was considered. Finally, key-problems were explained.

Typical for this study, although in the preliminary design phase, was the extensive usage of complex mathematical models concerning e.g.

Combat tactics
low level terrain-following
sensor characteristics
weapon range and effectiveness.

2.2 Results of Combat Simulations

Starting with the AS-mission, the main characteristics of RPV-vehicle, armament, and sensor system have been evaluated using a digital combat simulation program. The combat tactics used in this program have been derived for dogfights with a stern approach missile.

Combat opponent of the RPV is a manned AS-fighter in the upper performance class.

The variety of parameters investigated is given in Fig. 4. Wing loading, thrust to weight ratio and three different flap configurations characterized the vehicle itself, while armament and sensor performance were modeled by cones with variable angles and ranges. The initial conditions did not consider possible benefits of using approaches optimized and teleguided by the control station. Some results of the combat simulation regarding the vehicle parameters are shown in Fig. 5. Requiring an RPV-superiority of 80 : 20 kills (as a minimum), the necessary thrust to weight is shown as a function of wing loading for the three maneuver flap configurations. Load factor limitation is 6 g for the manned reference A/C and 10 g for the RPV.

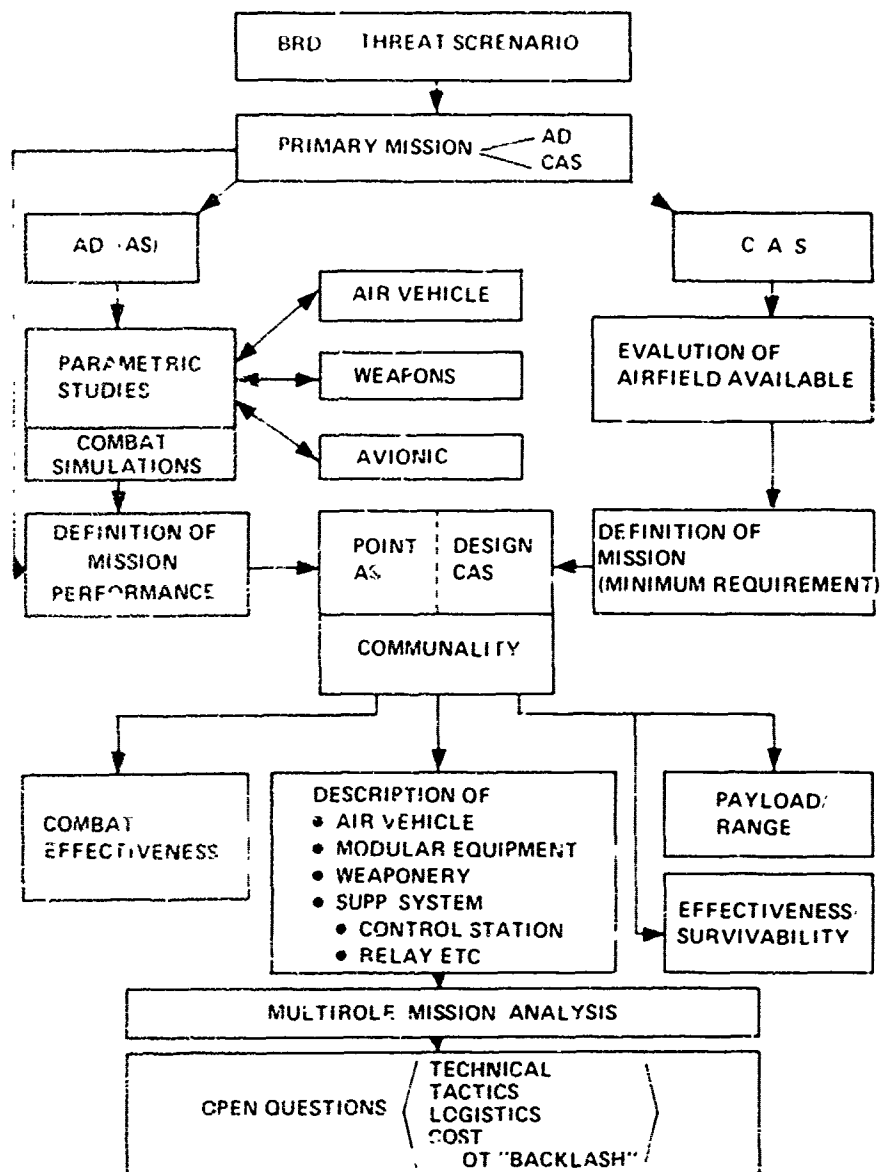


Fig. 3 STEPS OF RPV-STUDY

For the definition of a point design a maximum mach number of only 2.0 was chosen for saving weight without penalties in escape maneuvers. A thrust to weight of 1.3 and a wing loading of 240 kp m^2 (at the beginning of combat) with moderate flap performance gives the RPV the required superiority of 80 : 20 kills.

Due to the different initial conditions, the duration of combat is a stochastic quantity. Mean value of combat duration is 130 sec and 95 % of all fights are ended by a kill in 250 sec. Therefore this combat time of 250 sec was chosen for the point design.

All these results are valid under the assumption of equal weapon effectiveness, given by the characteristics of a dogfight missile limited to stern approaches. The sensitivity of the results due to changes in weapon ranges was also investigated and results maneuverability, as in case of the RPV, a remarkable reduction of missile performance is acceptable with only a small reduction of survivability (missile performance of reference A/C is held constant). Therefore a special armament for RPVs seems reasonable.

This picture would be completely changed, if a low cost weapon with head-on capability were available.

Using guns (ϵ_5 in Fig. 6) as the RPV armament, the survivability would be unsatisfactorily low.

Success of RPVs is dependent on sensor performance and sensor coverage area. Under the assumption that the combat opponent has full information about the RPV position in the combat area, the influence of a reduced RPV coverage area (given by target acquisition and tracking cone) has been evaluated and is shown in Fig. 7. The survivability of the RPV does not fall off until a sensor viewing angle of 90° . Therefore information about the enemy's position in the forward half space is sufficient. This area can be achieved using a single sensor with a bore-sight tilted upward in the aircraft and turning the RPV itself into the target plane. In this case a sensor viewing angle of 60° would be sufficient.

PARAMETERS

● WING LOADING

$$\frac{W_0}{S}$$

● THRUST/WEIGHT (MAX A/B)

$$\frac{T_M}{W_0}$$

● AERODYN CONFIGURATION

$$\begin{cases} C_L \\ C_D \end{cases}$$

VERSION

A
B
C

MANEUVER FLAPS
SINGLE SLOTTED T.E. FLAPS
NO FLAPS

● MISSILE EFFECTIVENESS CONE

$$R_s, \epsilon_s$$

● CONE OF ACQUISITION/TRACKING

$$R_E, \varphi_E$$

● INITIAL CONDITIONS

$$R_0, H_0, \bar{V}_{0Z}, \bar{V}_{0RPV}$$

Fig. 4 PARAMETRIC STUDY-AS-RPV

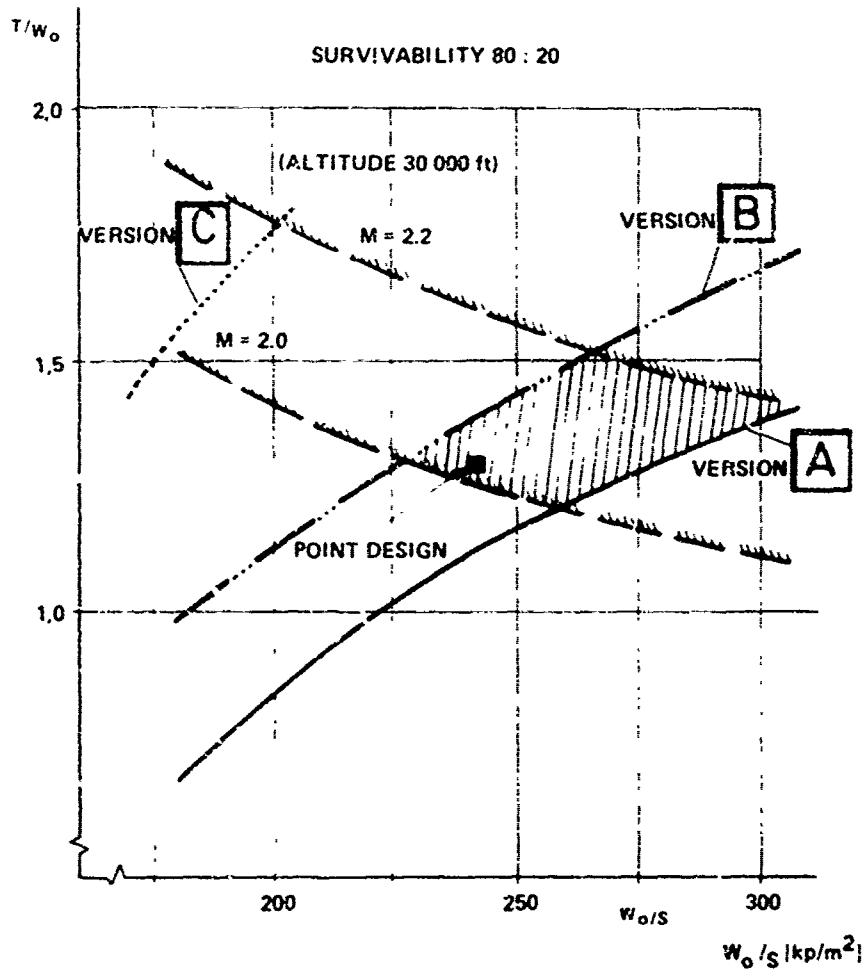


Fig 5 RESULTS OF COMSAT SIMULATION

Considering the simulation results and analyzing operational aspects the point design of the AS RPV can be summarized as follows

Thrust to weight ratio	1.3
Wing loading (T/O)	240 kp/cm ²
Maneuvre flaps	
Load factor (T/O)	10 g
combat time with a/b	250 sec
stern approach missile	
range:	2.000 m
cone angle	45°
radius of action (M = 0.9)	75 km
M _{max}	2.0
Take-off	boosted from launcher
landing	conventional

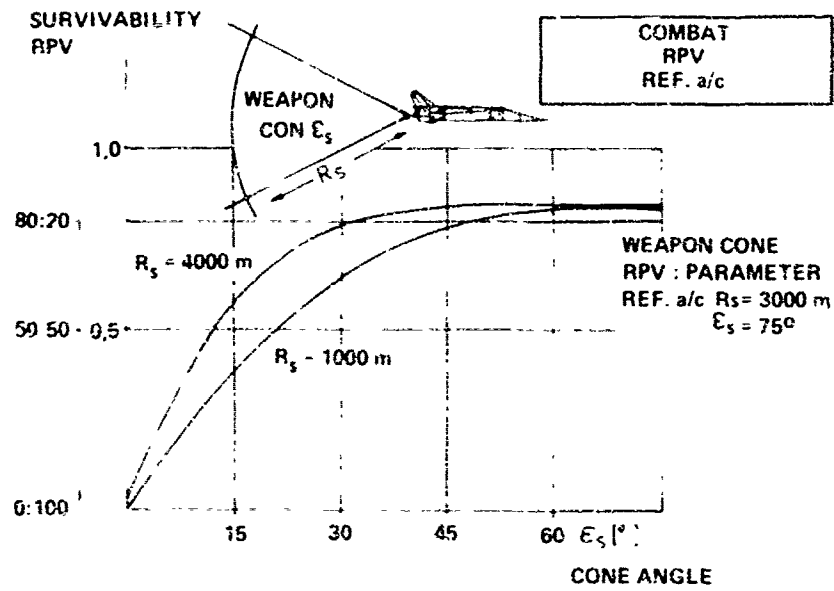
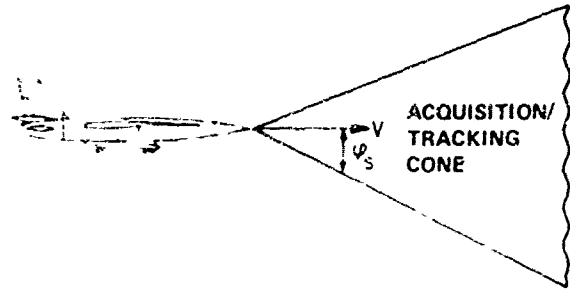


Fig. 6 WEAPON PERFORMANCE AND SURVIVABILITY



OPPONENT OUT OF SIGHT. RPV MANEUVERS
CORRESPONDING TO
THE LAST SENSOR
INFORMATION

ψ_s OF OPPONENT 190°
(NO LIMITATIONS)

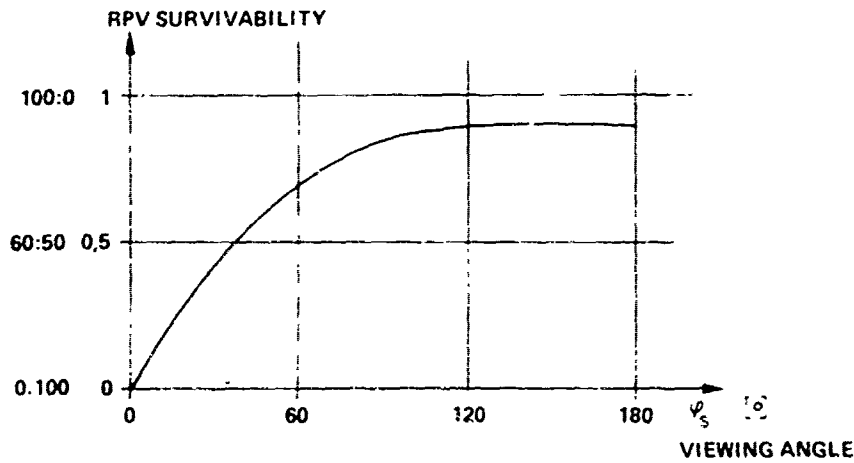


Fig. 7 INFLUENCE OF SENSOR-VIEWING ANGLE

2.3 The CAS-mission

The vulnerability of future A/C on the ground has to be reduced generally, this is valid especially in scenarios with a high threat of counter air missions as for the FRG. RPVs, as a new generation of A/C, have to meet these requirements. Therefore, dispersion on small airfields is mandatory; T/O and landing performance are predominant. The T/O/landing rolling distances are shown in the T/W - wing loading diagram Fig. 8. With a wing loading of 170 kp/m² the AS/RPV has a rolling distance at landing 370 m (with zero length T/O).

The CAS-RPV has to carry more fuel and higher military load than the AS-RPV; therefore by overloading the AS-version a conventional T/O is mandatory. Staying within take-off rolling distances below 350 m, in overloading of the AS-version by more than 65 % is possible with a load factor (T/O) of less than 6 g. The rolling performances demand a/b - take-off. In Fig. 8 superimposed q-maneuvers are included in thrust requirements. The maneuvers cover the terrain-following maneuvers on the one hand and maneuvers (jinking) superimposed for increasing the survivability against anti aircraft artillery (AAA) on the other.

This survivability by jinking maneuvers is shown in Fig. 9. With the same survivability as an A/C in level flight, a maneuvering A/C (mean value 4 g) can cross five to ten times more AAA belts. This advantage of increased survivability is of course restricted to unmanned A/C and must be made possible by higher thrust and fuel reserve.

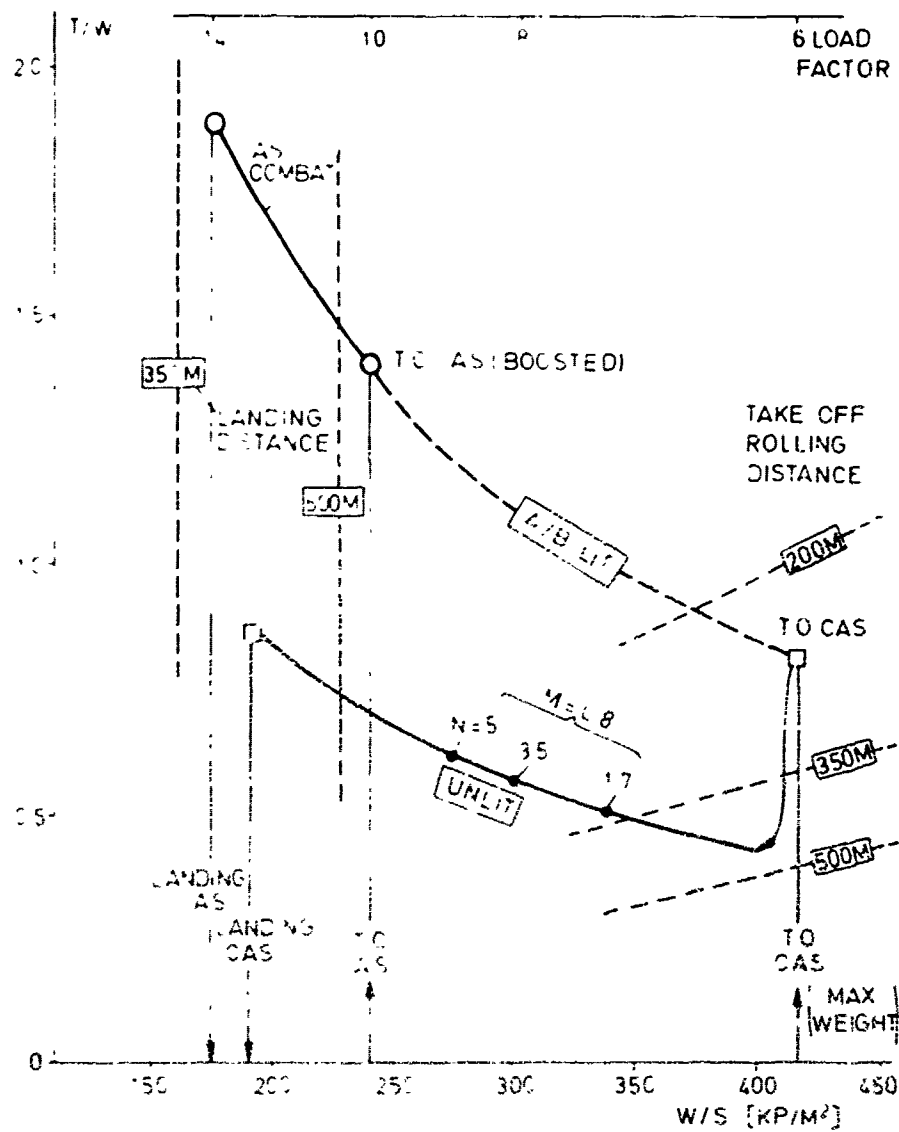


Fig. 8 RPV-POINT DESIGN T/O/LANDING REQUIREMENTS

2.4 Point Design

Figure 10 gives the configuration of the point design in the CAS version. It is a single engine design with the following main data

Overall length	10 m
Overall span	6.3 m
Wing reference	13.3 m
Sweep angle	21°
Aspect ratio	3
Engine	1 (Type J101 GE - 100)
Max. thrust ./b	
ISA, SL	4150 kp
Weight (O.W.E)	2500 kp

The T/O weight of the AS-version (including 100 kp missile and 600 kp internal fuel) is 3200 kp. The maneuver performance of this design is given by turn rates of 25 °/sec (instantaneous) and 15 °/sec (sustained) at 20,000 ft altitude.

For CAS missions the RPV point design can be considerably overloaded (Fig. 8). Military load and fuel (internal and external) up to 2800 kp still keep the load factor above 6. With the corresponding T/O weights of 5300 kp the rolling take-off distance - using a/b - is 300 m. At this order of T/O performance RPVs are able to use smaller dispersed airfields, in this way increasing on-ground survivability.

Figure 11 gives payload - range tradeoffs for the point design. In the radii of action, a 50 km distance with 3 g maneuvers (mean value) is included. 2000 lbs payload can be carried at $M = 0.85$ and 3 g. The drag of higher payloads up to 4000 lbs requires a reduction of speed or of maneuver loads. Internal fuel is limited to 600 kpl, therefore higher ranges require external fuel in considerable amount. The engine thrust - optimized for AS - seems well matched to the CAS requirements

short T/O distances

high drag of payload and external fuel tank

high induced drag due to superimposed maneuvers

In addition to the quick alert AS-mission explained above, the RPV point design can be used in air defence missions with a long loiter time. With a conventional take-off at the maximum T/O weight of 5300 kpl, a loiter time of 3 h at $H = 20\,000$ ft and $M = 0.45$ can be realized. At this performance, cruise to the combat area and a combat time of 250 sec are included.

Figures 12 and 13 summarize the main data of the point design performance for various attack and air defence missions in the order of priority as it is now seen for the FRG.

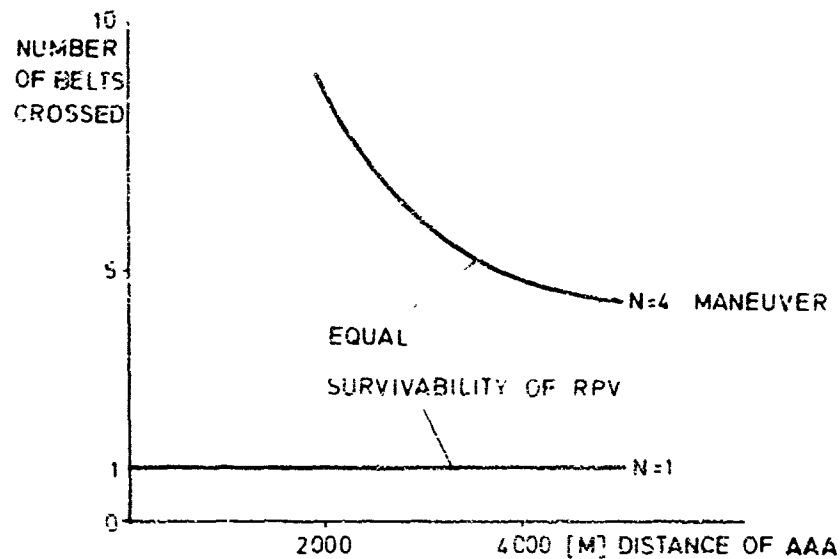


Fig. 9 CAS-RPV SURVIVABILITY AGAINST AAA IN BELTS

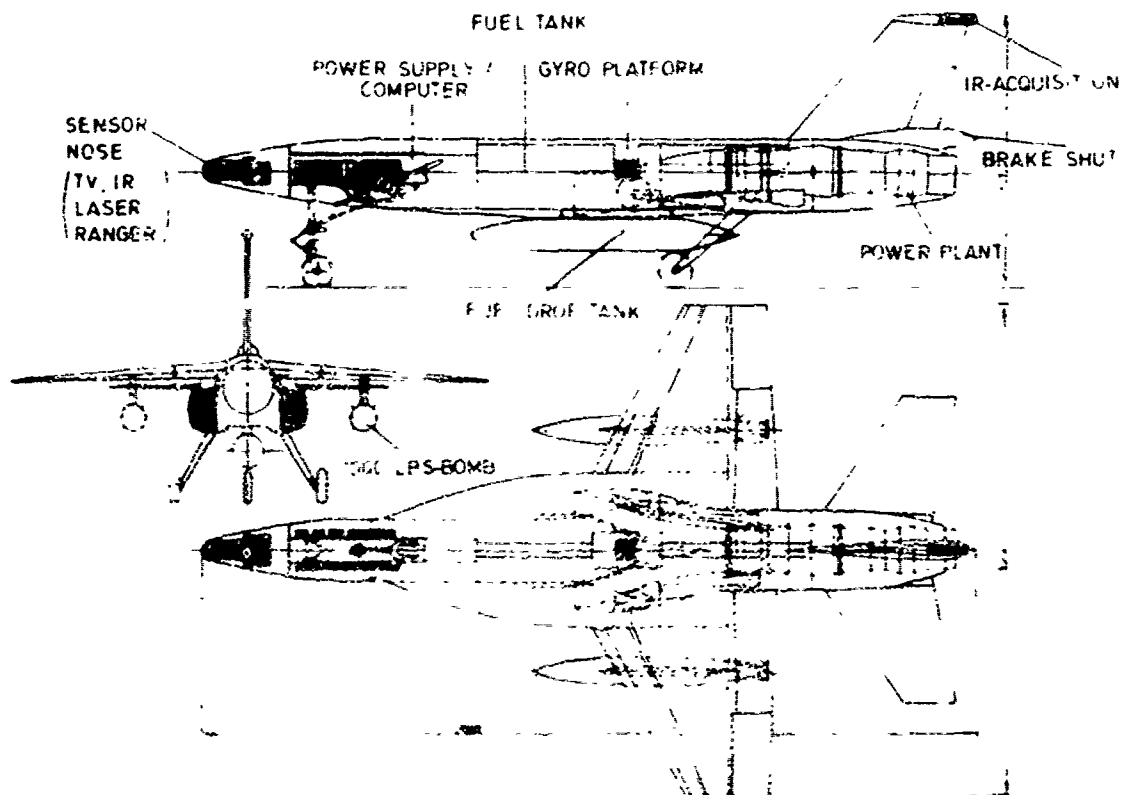


Fig. 10 POINT DESIGN

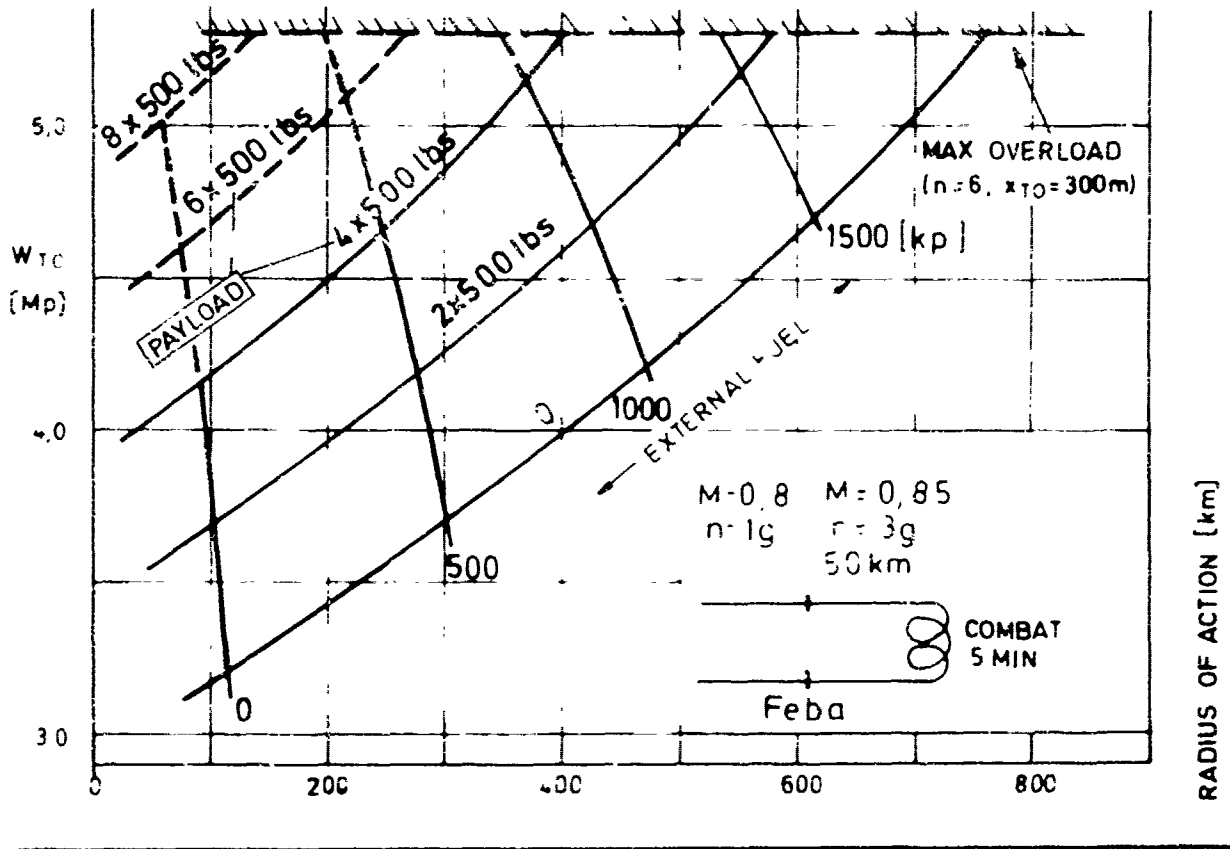


Fig. 11 PAYLOAD-RANGE, CAS-MISSION

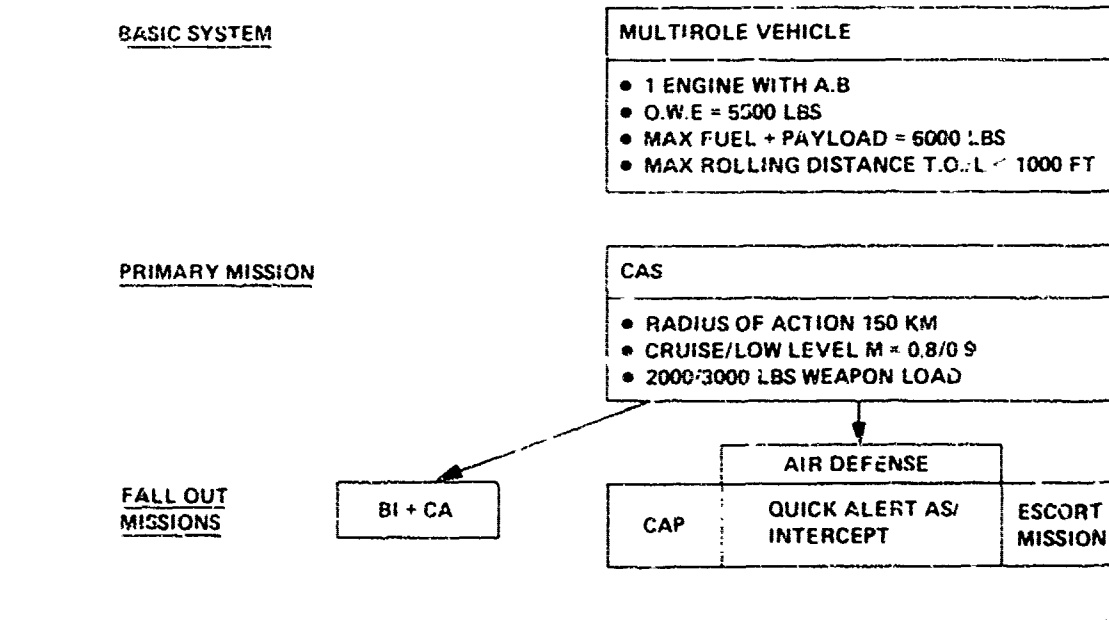


Fig. 12 RPV-MISSION

AIR DEFENSE			
BI + CA	CAP	QUICK ALERT AS/ INTERCEPT	ESCORT MISSION
<ul style="list-style-type: none"> • RADIUS OF ACTION 250 KM • CRUISE M = 0.7 • EXTERNAL FUEL • 2000 lbs WEAPON 	<ul style="list-style-type: none"> • LOITER $3h < M = 0.4$ $H = 20000 \text{ ft}$ • CONVENTIONAL T.O • EXTERNAL FUEL • TARGET ACQUISITION BY AEW/MANNED CAP 	<ul style="list-style-type: none"> • LOCATED NEAR FEBA • MOBILE TRANSPORT/ LAUNCH SYSTEM • $M_{MAX} = 2$ • TAKE OFF WITH BOOSTER • TARGET ACQUIS. BY AIRBORNE/GROUND CONTROL STATION 	<ul style="list-style-type: none"> • CONVENTIONAL T.O. • EXTERNAL FUEL • CRUISE M = 0.8/0.9 LOW LEVEL • TARGET ACQUIS. BY AIRBORNE CONTROL STATION

Fig. 13 FALL OUT MISSION

3. SURVEY ON NAVIGATION AND GUIDANCE PROBLEMS

The shortcomings of programmed flights are obvious. Fighter missions require accurate outbound navigation and target detection/designation/tracking, which need some kind of remote updating and "piloting" via a data link with a more or less high bandwidth.

The first idea - an obvious one, - to move pilot and cockpit out of the A/C to the remote control station, has been realized in first generation RPVs. In doing so, all cockpit data and visual information have to be transmitted to the control station, and pilot commands must be sent back to the A/C. This solution permitted a high degree of on-board automation, but required a high bandwidth for transmitting acceptable sensor information (video signals) and demanded a high work load of the remote pilot. The feasibility of such a method under heavy ECM environment is still open.

In any case covering lock loss phases and reducing detection probability for RPVs requires a certain self-sufficiency of on-board navigation and guidance. That is even more true, if computerized flight path optimization is needed for effective approach and combat maneuvers, including weapon delivery computations. Therefore the degree of automation on board must (and can) be the highest in air defence RPVs.

In these missions, target detection, IFF, threat evaluation, target designation and guiding the RPV to the combat area has to be done by an airborne or ground control station. After having acquired the designated target the RPV is self-sufficient.

A guidance scheme for the AS-RPV is shown in Fig. 14. After target acquisition by the RPV sensors, the control station has only monitoring tasks. Such a guidance method requires only a low data link bandwidth, but the higher degree of on-board automation is more expensive, although it seems inevitable for the high performance RPVs of the second generation.

In air to ground attack missions the situation is rather different. Under the assumption that low level flights are mandatory for survivable RPVs, the outbound navigation accuracy and target detection/designation/acquisition are severe problems.

For sufficiently accurate RPV navigation two approaches can be considered:

Method 1 is based on inertial equipment. Depending on cruise distances, inertial navigation accuracy with doppler and/or teleupdating is necessary to acquire even fixed ground targets or those designated by a FAC. Furthermore, returning to the airbase must be assured.

Method 2 uses ground based navigational equipment (e.g. Loran, Microwave landing systems) with computation on-board. This method gives sufficient accuracy as well.

A mixture of both methods has the most flexibility, but is also the most expensive.

Method 2 - most attractive for reducing on board complexity - has the highest degree of vulnerability and disturbability.

Therefore inertial navigation with teleupdating seems to be the best solution. A radar altimeter for enabling low level flights is also needed. On the other hand this system has a high commonality with the hardware equipment for AS missions explained in Fig. 14. Only the sensor package corresponding to the different targets is completely different. Using these sensors for aligning the final approaches, conventional landings can be conducted by the inertial equipment (platform/radar altimeter) without ground navigation aids - a very favourable method for using smaller dispersed airfields.

Compared with these navigation problems, the target detection and designation process in air to ground attacks is even more complicated. For the near future, only target designation concepts with man in the loop should be considered. Human detection and designation can be done directly (FAC) or by using sensor information transmitted from the RPV to the remote control station. In the former case the separately located "Designator" must illuminate the target ("Marking"). Then the RPV (and its weapons) can be guided automatically (semiactive guidance). In the latter case homing on the target must be remotely controlled. Fig. 15 gives a survey on the different methods. The penetration depths (from FEBA) are different in these methods due to limitations in positioning the FAC or in transmitting the sensor information to the control station. Estimated values are given in Fig. 15 as well. The deepest penetration can be expected utilizing a relay A/C. Compared with semiactive guidance, the remote detection and designation methods require remarkably higher bandwidth for the video transmission link and are therefore more sensitive to ECM.

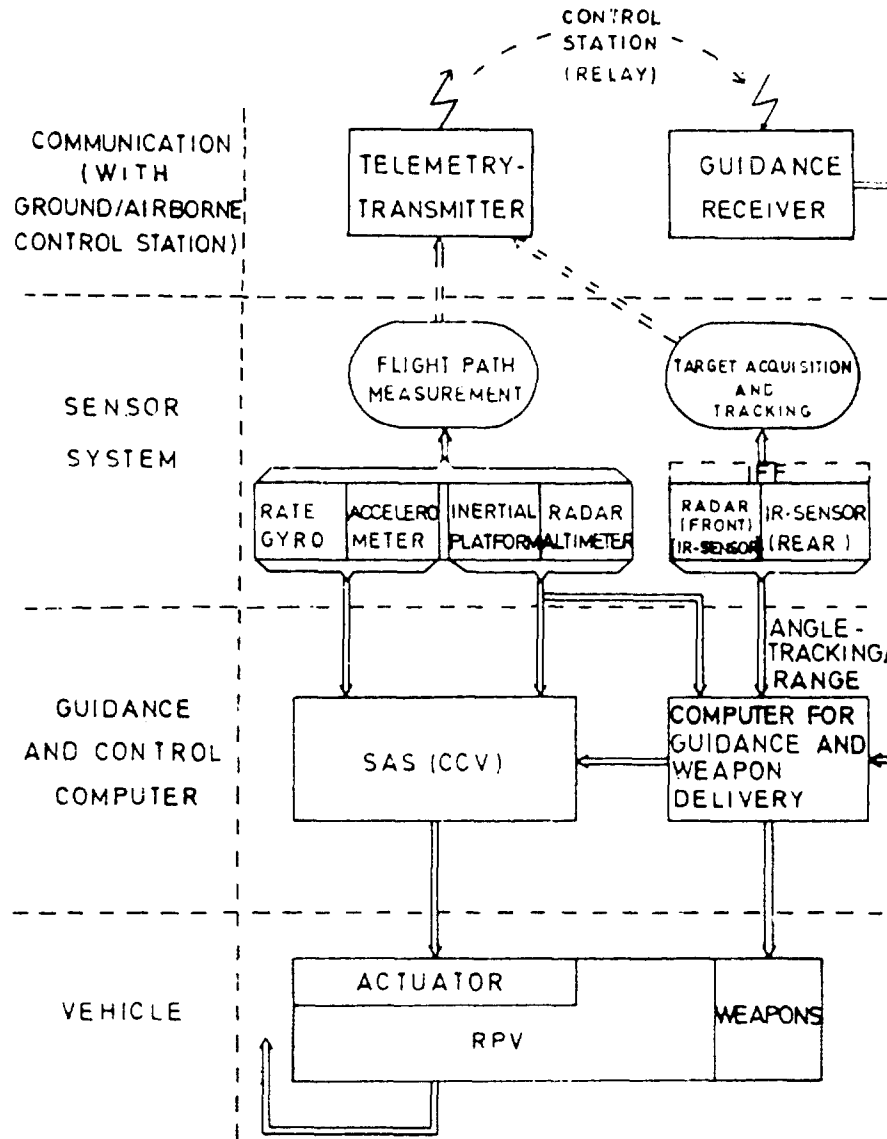


Fig. 14 GUIDANCE AND CONTROL OF AS-RPV

The tasks of (ground or airborne) control stations are the following:

- Receiving information from RPVs and from supporting systems (e.g. Search radar)
- Information processing
- Transmitting commands to RPVs and supporting systems (e.g. Relay stations)

TARGET DESIGNATION BY MAN		DESIGNATOR POSITIONING [FEBA]	RPV-GUIDANCE	MAX. PENETRATION RADIUS [KM]
DIRECTLY	UTILIZING SENSORS ABOARD			
X		CLOSE, ON GROUND	AUTOMATIC	3-5
X		CLOSE, AIRBORNE	AUTOMATIC	10-20
	X	CLOSE ON GROUND (REMOTE PILOT)	REMOTELY PILOTED	10-15
	X	AIR BORNE, FAR OFF	REMOTELY PILOTED	50-70
	X	ON GROUND, FAR OFF	REMOTELY PILOTED UTILIZING RELAIS STATION	100-150

Fig. 15 PENETRATION RADIUS OF VARIOUS GUIDANCE METHODS

The possibility of processing target information in the control station, normally a task of the pilot in the cockpit - can essentially increase the effectiveness of RPVs. Especially in low level, high speed flights detection of targets is difficult and limited to small areas around the flight paths. On-board information processing and cockpit display is restricted to simple processing. In Fig. 16 the scheme of a special display located in the control station is shown. It permits the selection and presentation of visual information under optimal viewing angles transmitted by RPVs forward looking sensors. It is a display accumulating vision stripes thereby increasing scanned area and increasing time for looking at the target area. This processing method increase the probability of target detection reducing the bandwidth of transmission in the way that redundancy of data transmission is replaced by information storage. This method can be extended to an interactive display for navigation updating and commands in the target area

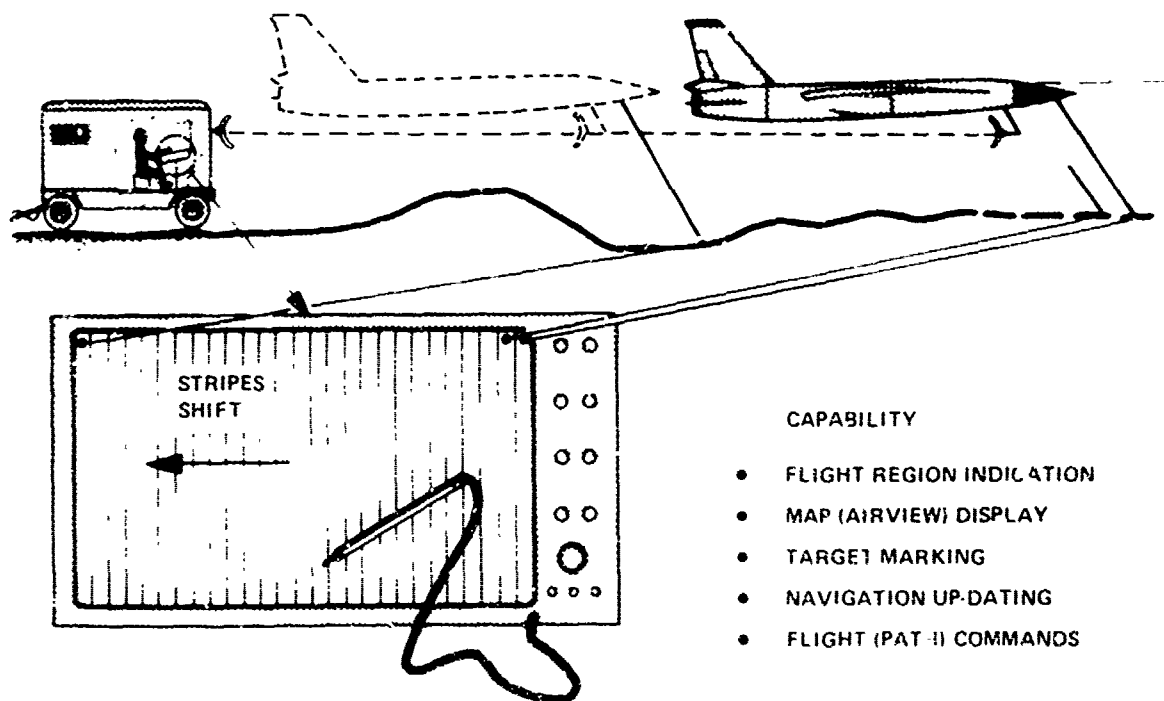


Fig 16 VISION STRIPES ACCUMULATING DISPLAY (VISAD)

SUBSYSTEM	PRIORITY	KEY PROBLEMS
STRUCTURE/KONFIG	2	CCV-CONFIGURATION MATERIALS/CONSTRUCTION REDUCED SAFETY FACTORS
ENGINE	2	REDUCED LIFE TIME MAINTENANCE/STORABILITY
BASIC EQUIPMENT	2	MODULE TECHNIQUE SYSTEMS DESIGN FOR • STORABILITY • ACTIVATION TIME
GUIDANCE SYSTEM & DATA LINK	1	SENSOR PERFORMANCE TRANSMISSION BANDWIDTH INFORMATION PROCESSING DISPLAY TECHNIQUE

PRIORITY 1 DECISIVE FOR FEASIBILITY

PRIORITY 2 IMPORTANT INCREASE OF SYSTEM
PERFORMANCES & COST/EFFECTIVENESS
REACHABLE

Fig. 17. PRIORITY OF TECHNOLOGICAL PROBLEMS

4. CONCLUSION

Thinking about RPVs for future fighter missions is a very difficult task for A/C designers. In contrast to the normal situation, they cannot start with definite military requirements. The reasons for the failing requirements are the following:

- 1) These new type of systems are essentially dependent on technology working in an operational environment, influenced by the technology itself, e.g.
 - Data transmission under ECM environment,
 - Storability and short mission-preparation time depending on basic/mission equipment,
 - Mission effectiveness depending on the supporting systems (control station, relay, mixed manned/unmanned fleet),
 - Feasibility of operation from dispersed airfields depending on logistics and infrastructure
- 2) The technical feasibility of important subsystems is open. Many of these open questions can be answered only by future experimental studies in original environment assisted by military personnel
- 3) Large changes in infrastructure (logistics, command and control) are mandatory for an effective RPV operation. The impact of such changes and their definitions can be determined only in complex simulations fed by experimental studies
- 4) The big step beyond manned systems will be taken only if the effectiveness of RPVs in future battlefield scenarios can be clearly demonstrated. Therefore modelling of scenarios with more confidence and more detail than usual is needed

Therefore, our study on multirole RPVs can be considered only a small part within such a complex loop of technological/operational paperwork/simulation/experimental studies

With respect to technological problems identified as important in this study, Fig. 17 gives a survey with a proposal concerning the priority of these problems

Navigation, guidance and data transmission are the subsystems of highest priority for the investigation in the near future, because these problems can influence the feasibility of future fighter type RPV

In comparison with these key problems the areas of basic equipment, engines and structure have somewhat smaller priority; but investigations in these fields are necessary as well to get inputs for

- Operational evaluations (e.g. maintainability, storability, vulnerability)
- Cost evaluations
- Starting long lead technological work

5. SUMMARY

RPVs for fighter type missions have a lot of uncertainties regarding operational and cost aspects, which only can be clarified by experiments and complex simulations updated from these experimental studies.

Concentrating on technical and technological areas of RPVs and analyzing the main threats for the FRG, a AS/CAS-RPV has been studied on the base of a unified vehicle, modular mission equipment, and specific armament. The characteristics of a preliminary point design have been derived from combat simulation and threat evaluation. By overloading the optimum AS design, payload, range, maneuverability, and T/O/landing performances required for CAS missions can be met. Mission effectiveness and survivability of the point design is highly superior to manned A/C. In the paper presented only a survey on navigation and guidance problems could be given. Key problems have shortly been discussed in the conclusions.

Problems with the current design process are summarized on Figure 3. They arise from the characteristics of advanced aerospace vehicles, the nature of the design process, and the big design staffs required for such vehicles. The problems stem from the increased size, sophistication, performance, and sensitivity of new vehicles which increase everything related to the design process. Of particular concern are the longer and longer times required to design an advanced vehicle. Not only does this time increase costs in many ways, it also makes the vehicle development cycle unacceptably long with the danger of introducing obsolete products into the market. Risks of poor quality and/or high costs are great.

In addition to many design requirements, conditions and criteria, and a multitude of data that must be generated, analyzed, and communicated, large numbers of people, that must work together as a team, are involved. The result is a large organization that operates in a complex manner, is difficult to manage, and which of necessity takes a long time to get its job done properly. The size of the product, the number of design conditions and the amount of data will continue to increase; therefore, time and money can be saved only by changing the process so that fewer designers can do their thing faster. This is where automation can help; the computer must be used to do all things that do not require unique human traits, faster (and better?) than man. Design flow time can be reduced with secondary benefits in reduced costs (time saved is money saved) and increased design quality. Fortunately, design technology and computer hardware and software have reached a state where much of the design process can be automated in the next decade.

The growth of automation in aerospace vehicle design - past and future - is shown in Figure 4. It traces the growth of automated structural analysis from elements to complete vehicle capability, the emergence of automated structural design and its development to a mature technology within this decade, and the prospect of automated vehicle design growing rapidly from the embryonic systems now being used. The last 20 years brought about a revolution in structural analysis through computerization. For the vantage point of 1985 or 1990, we will see that a similar revolution has occurred in design.

Automation of the total airplane design process can occur in an evolutionary manner in which the design organization and computer programs are modified in steps. The first steps will be to use the computer as a communications link between the players, gradually programing more and more human activity of a routine nature for the computer, then extending automated synthesis capability to the total vehicle. By this time it will be apparent that some fundamental changes in procedures, organization, and man-machine relationships are necessary, Reference 2. Hopefully, enough research on the design process, both its technical and social features, will have been done so that revolutionary changes can appear with a new generation of managers, designers, computers, and software. The process that results will be vastly different from what we have today in the tools, people, training, and organizational concepts, References 1 and 2.

The words design, integrated, computer-aided, and automation have been used rather freely, so I will present now the definitions I will use in this paper. I am equating automated design to computer-aided design, and using them interchangeably, to indicate that this is the best combination of men and machines for designing a product. It can involve several levels of computerization that will change with circumstances and time. The machine is the digital computer and all types of devices, apparatus, or machines that can be operated with or from it to aid the design process. The design process encompasses all activities required to generate the data needed to produce a product and therefore covers a wide scope of technical disciplines ranging, for example, from aerodynamics to noise to structures to manufacturing to economics. Integrated refers to how the many computer programs used in the design process work together. An integrated system provides for the greatest interaction and flexibility in program utilization and the highest potential for automation without loss of the insight and innovation that only a human designer can provide. In addition, an integrated system can provide for an intelligent dialog between the designer and the computer, partners that augment and complement one another in managing and accomplishing the design task.

THE DESIGN PROCESS

The process used to design any product is basically simple, Figure 5. Someone sets down a requirement, the designer finds an acceptable configuration that satisfies it, and then generates the data required to fabricate the product. The selection of the appropriate configuration is not simple, however. The designer will first select one that he thinks, from experience, will meet the need - this is the idea and innovation stage. Then he analyzes it to determine the characteristics of his product and compares these with the characteristics allowed or required of it - the analysis stage. Initially, the product will lack some essential characteristics so it must be changed - this is the decision stage which requires insight and experience. However, analysis of the effects of design changes on product characteristics can assist the designer in his decisions. Next, the designer goes through several cycles in which the product is reconfigured and reanalyzed until all required characteristics are obtained in the final configuration. Then, all data needed to fabricate the product are compiled and sent to the shop. Of course, manufacturing methods and costs were a factor in the design evolution from the beginning.

The various blobs in Figure 5 represent work to be done or tasks. The arrows represent the flow of data or information. Tasks and data are the elements of the design process which is a data management activity - the generation, flow, and processing of data are all that happens in design. Of course, this data

management activity is carried out by people and machines working in an organization. If the organization required to design a product is small, the management of the information flow is not difficult. A large design organization is another story.

Figure 6 is a cartoon of the current design procedures used for large airplanes such as a wide-body jet transport. It shows men and machines in the same information flow as in Figure 5. But this chart shows that much of the data flow only because one guy hands it to another. This is not good. The design organization for a large airplane includes more than a thousand people at its peak, involves numerous individuals that are designers, and has large numbers of analysts, draftsmen, test engineers, technicians, administrators, and other specialists. A few simple calculations of the combinations of personal contacts required in a group of this size reveal the staggering magnitude of the person-to-person communication problem in giving and receiving data and decisions in a highly interactive situation like multidisciplinary design. For example, if only 50 people on the staff must talk to each other periodically, then 1225 conversations are required in each period (with 100 people the number rises to 4950). Big design organizations, then, usually manage data flow inefficiently. They should look to automation to speed up this part of the process by reducing the number and duration of human contacts. The largest gains from automation probably will be in the big organizations with the big problems, however, any design organization can benefit. In addition, further automation of analyses and other tasks will also speed the design process, particularly in automating methods for rapidly resizing or reconfiguring a vehicle or component that does not satisfy its design requirements.

The design process has important time and phasing characteristics suggested by Figure 6 but not apparent in Figure 5. Figure 7 extends Figure 5 into the time domain to illustrate the sequencing and cycling that occurs as the design progresses. Note that each activity occurs intermittently. If each is performed by a particular specialist, his assigned task will not be completed in one work period and he will have to be reoriented each time it comes around again. Continuity of tasks is often achieved in design practice by simultaneously working tasks that ideally should be done in sequence. Then, individual specialists could be working simultaneously at different design levels on the same project with the danger that much effort may be wasted. The problem becomes more acute on multidisciplinary airplane projects wherein aerodynamics could be several configurations ahead of structures. All disciplines should work approximately in-phase to produce a technically balanced design in each cycle. Automation can help achieve this goal by reducing the time required for each task and by speeding the flow of data between activities and disciplines.

The design process passes through several stages or levels as it progresses from an initial concept to the final details. Figure 8 is a diagram of the development of an aerospace product that shows the place for design. Continuing activity in research, development, and marketing periodically identify new concepts and technology with sales potential. This idea enters a conceptual design phase to scope its characteristics and, if attractive, moves into a preliminary design phase where it is worked to greater depth. When the design is sufficiently mature, management authorizes the product go-ahead and detail design, manufacturing, and testing lead to first product delivery. Design support for the product in production is a continuing activity to cover changes and modifications in future product improvement. This total process involves many subtasks and cycles in a variety of sequences over a long period of time. The needed automated design system must encompass this total process and the multitude of functions contained therein.

Design networks can be drawn of processes depicted in Figure 8 to any level of detail required, but they soon become exceedingly complex. Figure 9 goes one step in that direction by expanding the preliminary design phase and indicating major decision points and recycling routes. The preliminary design phase is divided into sizing the product to the marketing criteria, refining it by applying more powerful analyses, and verifying it by more rigorous analyses and selected tests. The design is then reviewed to determine if construction should proceed. Note that at each decision point before go-ahead, the process may recycle to as early a level as appropriate. However, once it enters detailed design the major system parameters are fixed and any problem that arises must be solved within that subsystem. Therefore, preliminary design must produce a thoroughly satisfactory configuration for the total system adequate in all subsystem characteristics, including fabrication costs, if a successful product is to result. Not only must our automated system accommodate the requirements of each of these levels and decisions, it must also insure adequate technical depth at each level (especially in preliminary design) within the time and cost available.

Design levels have been discussed without defining them in detail; Figure 10 attempts an approximate definition. Each organization utilizes different definitions and process within its design organization so Figure 10 is an amalgamation of several sources. The levels are defined in terms of the manpower, flow time, and number of configurations examined. The accuracy of weight estimates and short descriptions of the objectives and product are included too. All numbers are approximate because of the variability of scope of a design project, ranging from a small fighter aircraft to a large supersonic bomber or transport. Again note the key role played by preliminary design; only one overall configuration goes into detail design, however, many components and parts will go through several design iterations in this design level.

DESIGN SYNTHESIS AND OPTIMIZATION

Diagrams of the design process discussed above (Figs 5, 6, 7, and 9) have shown cycles in which the design must be reconfigured because it did not meet some criteria. Resizing the total configuration or one of its parts can be done by trial and error, but systematic approaches called design synthesis are more productive. Synthesis includes optimization since the objective is to attain a maximum or minimum value of some merit function. Structural synthesis, for example, is often discussed under titles such as minimum-weight design and structural optimization. It has not been used enough in design although it has been available for over 20 years. References 3 and 4. Its use, however, is increasing.

Some general comments on design synthesis are appropriate here.

1. The synthesis approach provides a systematic way of conducting analyses used in design to reach the best configuration with a minimum of time and effort. Synthesis techniques must, therefore, be an important part of any automated design system.
2. Synthesis can, at least theoretically, be applied to the total system, as well as to small parts, but it is most useful for sizing the elements of a vehicle where less innovation is expected than in the formulation of system concepts.
3. Synthesis of most parts or systems of practical interest is too complicated for closed-form solutions. The use of a computer is essential. Therefore, synthesis is essential in automated design and synthesis of complex parts is impractical without automation.

The computer finds the optimum solution to a synthesis problem in either a direct or indirect way, References 5-7. The direct approach, exemplified by mathematical programming, makes an intelligent and systematic search among the design variables to locate the minimum value of its objective which may be weight, cost, or other considerations. This method is general but is usually expensive if large numbers of variables or complex analyses are involved. The indirect approach uses optimality criteria that are expected to produce the desired result, at least within engineering accuracy. Such procedures are relatively fast, they can handle large numbers of variables readily and are practical for design of large structures. Both approaches are needed for automated designs of vehicle systems; however, considerable additional development is required for application to multidisciplinary situations.

The application and strategy of optimization in aircraft design synthesis is summarized in Figure 11 for the three design levels. In the conceptual and initial preliminary design phase (sizing) the total system or configuration can be optimized with respect to a few key parameters. Subsequently, optimization must be limited to subsystems. However, subsystem input to configuration optimization must be in sufficient technical depth to accurately reflect the effects of subsystem changes on configuration parameters. The types of parameters optimized in each level are indicated for wing design as an example.

The number of parameters increases greatly as the designer goes into more detail. Judicious choice of the parameters to be optimized in each cycle or level is important. The designer cannot afford to dissipate his resources on optimization of secondary parameters. To use an extreme example, why optimize rivet size and spacing if the wing aspect ratio is still being varied?

The total system is designed at the conceptual level where ideas and innovation are essential whereas elements are of concern in the detailed levels. On the other hand, parameter optimization in the conceptual phase may be inaccurate because the analyses used are not of sufficient depth to include detailed design parameters that could affect some of the primary parameters. I am thinking, in particular, of the flutter problem which involves the integrated effect of many structural and aerodynamic parameters, some of which may be frozen before a comprehensive flutter analysis can be made.

Optimization strategy also varies with design level and three approaches are indicated. The mathematical approach uses the processes discussed above; it is useful at all levels. Trade-off studies explore a range of solutions around a local optimum to determine sensitivity to changes before committing to a specific arrangement. Sequential refining is the application of more accurate analytical modeling without formal mathematical optimization processes, but these refined analytical models can be subjected to mathematical optimization and trade-off studies, also.

Incorporation of optimization procedures into an automated design system imposes several requirements in addition to the basic technical, search, and mathematical capabilities. The user must have great flexibility in the way he sets up each particular task. He must be able to specify the merit function and the constrained and independent design variables he desires. He must have great freedom of choice in the execution sequence and the particular technical and optimization methods employed. He must have complete control over the computations so that he can stop the search, inspect intermediate results, modify the procedure, and start again. All this dictates that the design system must be very fast, flexible, and versatile; but only an automated system can accomplish this in a practical way.

PROGRESS TOWARD INTEGRATED, COMPUTER-AIDED DESIGN

From the discussion so far, you might get the idea that little has been done to integrate disciplines and do computer-aided design. On the contrary, computers are used extensively to do all kinds of design tasks and some rather sophisticated systems have been assembled and used. The point to be made is that we can go much, much farther.

Figure 12 lists a sampling of the code names and originators of programs in operation or under development. The list is not exhaustive, simply representative, References 8-14. Several programs with a company as originator were developed under U.S. Air Force sponsorship, for example, Reference 14. Conceptual vehicle synthesis codes are widely used. They have a highly computerized, broad, multi-disciplinary base but relatively little technical depth in some disciplines such as structures. Such comprehensive systems are not now available for the subsequent design phases (preliminary and detailed) of a complete vehicle. The other examples shown are representative of progress in structural analysis and design. Some design programs size individual elements of a prescribed structural arrangement under given loads, using optimality criteria for minimum weight. Other programs integrate loads and structures, including aeroelasticity considerations, some doing primarily analysis while others emphasize structural sizing. Although NASTRAN, a large NASA program, Reference 15, is listed under structural analysis, plans for future additions include a fully stressed design module to provide some sizing capability. Similarly, other programs may be undergoing increases in scope.

The aerospace industry is not the only one that is automating the design process, Figure 13. The U.S. Army is considering a comprehensive automated design system for each class of commodity it uses. The U.S. Navy has been steadily increasing its use of computer-aided ship design and the U.S. Maritime Administration has been encouraging accelerated use of computers in commercial shipbuilding and design. In architecture and civil engineering, extensive international activity in automated design is being coordinated and fostered through technical societies, Reference 16. All this is further evidence that integrated, computer-aided design is an emerging technology, the benefits of which are widely accepted.

In general, programs for computer-aided design have been built to provide capability in one design level only. Figure 14 shows the approximate design level applicability of several programs listed in Figure 12. Note that the IPAD program, which will be discussed later, is the only one intended to cover all design levels in a single-integrated system.

The IPAD program will be described in the next section to illustrate the type of integrated, computer-aided, system design programs that can be developed today and the benefits that can be achieved. But first, let's examine two representative structures programs (IDEAS and ATLAS) that have demonstrated substantial improvements already. The structure of an aerospace vehicle contains more parts and details that require exacting design than any other subsystem. The airframe of a wide-body jet transport, for example, contains more than 1,000,000 parts. Consequently, structural analysts and designers of airplanes, missiles, space vehicles, ships, and buildings are leading proponents of computer-aided design.

Figure 15 is a simplified layout of the Integrated Design Analysis System (IDEAS) developed by the Grumman Aerospace Corporation, starting in 1967, Reference 17. IDEAS is a highly organized system, of more than 70 computer program modules interrelated by data packages from a central data bank that stores all calculated data. Sequencing and execution of these modules enables the design team to plan, schedule, coordinate, and control a stream of analyses that provide internal loads, deflections, and temperatures for many subsequent analyses by several engineering groups. IDEAS is primarily used at the detail level and the flow of information in the first and subsequent IDEAS cycles is an integral part of the Grumman design organization and process. IDEAS was used to design the F-14, requiring over 2000 computer hours. This application verified their initial predictions of substantial reductions in the time and engineering man-hours required to accomplish the same tasks by conventional procedures. It is a major step forward in computer-aided analysis and demonstrates that real gains are indeed attainable. However, it currently contains only a limited, but growing, capability for design, that is, for determining allowables, margins of safety, and resizing.

The development of the IDEAS program concentrated on building interfaces between accepted analysis procedures and deliberately avoided new technology. A different approach is being taken in The Boeing Commercial Airplane Company in developing an integrated structural analysis and design system (ATLAS), Figure 16. Its objective is to integrate related structural disciplines in a common framework, applicable to most design levels, with emphasis on automated control of program flow and data communications between modules, utilizing all available computer resources to achieve "optimal" processing efficiency. Thus, ATLAS has a control module that functions like a design manager to make analysis path decisions and monitor their execution. Module development, a technically oriented language for task definition, and emphasis on automatic input data generation, are among its new technology features. The development of the analysis loop for strength design is operational and that for stiffness is nearing completion. Future additions planned include flutter design modules to resize for the next analysis cycle. ATLAS has been applied to several analysis tasks and a comparison of the time and resources required by ATLAS and by conventional methods to do the same structural analysis job on a supersonic commercial transport at the preliminary design level showed that both manpower and flow time were cut in half.

INTEGRATED PROGRAMS FOR AEROSPACE-VEHICLE DESIGN (IPAD)

The next step toward design automation in the aerospace industry is to assemble a computer-aided design program for complete vehicles or for an entire transportation system. NASA is studying the feasibility of such a system called Integrated Programs for Aerospace-Vehicle Design (IPAD), Figure 17, Reference 18. The basic software in IPAD could be used on other design projects too. The U.S. Air Force, Army, Navy, and industrial companies are participating in these studies. IPAD in the design situation will provide the software for conducting the design process with people and computers. The major software elements are the Executive, the Data Base Manager, the Utilities, and the Operational Modules. The Operational Modules are the computer codes that perform particular analysis or design functions. These programs are, for the most part, now in use but, in most present design activity, they are linked together by humans. In IPAD, programs can be linked together in the computer in any sequence desired by the designer through the Executive, which is the manager that interfaces with the user, provides him control of the process, and provides instructions for carrying out each part of the design task. The Data Base Manager and Utilities are the Executives' staff assistants that collect, organize, store, distribute, and display information, computational activity, and task sequences for effective operation and control of the process. The primary NASA goal in IPAD is the development of the IPAD core - the Executive, Data Base Manager, and Utilities. NASA will develop, also, some Operational Modules, as appropriate, but most of them will be programs already available to the IPAD user. IPAD will be constructed so that modifications required to fit existing Operational Modules into the IPAD system will be minimized.

Figure 18 gives a different overview of IPAD, shows the interrelationships between the major components and illustrates the engineering usage philosophy. The technical management and engineering capability utilized reside in the people (managers and users) and the library of data and automated modules they have developed. They exploit their capability through a variety of computer hardware and software. The host computer has a variety of peripheral devices to facilitate user input and output and his interaction with the computer. The IPAD framework software supports and augments the capabilities of the operating system software and the automated analysis, design, drafting, and management tools of the user.

The objectives of IPAD are given in Figure 19. NASA envisions a versatile and open-ended, multidisciplinary design tool that can handle a wide variety of design situations and be responsive to the needs of the designer. The objectives are not directed toward fundamental changes in the current design process but toward better and more extensive use of the computer in existing organizations. IPAD will be structured to do any group of tasks the designer chooses with complete automation possible on tasks for which appropriate operational modules are available. The basic idea is to integrate design activity through the computer to speed up the process. Automation, in the sense discussed in this paper, will then grow as the IPAD system is developed.

Since the IPAD data bank can contain all data on the status of a design project, it can contain records and logs of all activities that have occurred and can display them and compare them with plans. Its capability in this area can greatly reduce the demand on men to maintain so many routine records. Therefore, it introduces new control features between the people and the systems within an organization and a way to achieve more accurate information and more positive communication. Thus, engineering project management may more efficiently plan, control, and communicate design information, Reference 2.

The status of IPAD is given in Figure 20. Two feasibility studies are being conducted for NASA by The Boeing Commercial Airplane Company and General Dynamics/Convair Aerospace Division with completion scheduled in September 1973. The IPAD system development plans resulting from these studies will be evaluated by NASA with assistance of the aerospace industry in the months that follow. Development of the core software of an acceptable system will begin then at a pace determined by the available resources. When a functioning IPAD system is developed, it will be released to the U.S. aerospace industry for checkout and design use. The dates on Figure 20 are the current estimate of the time required for the first level of IPAD to become operational in U.S. industry. IPAD will be a tool for U.S. industry primarily and the U.S. Government secondarily.

Operational modules will be collected and modified to function in IPAD. Some of them may be NASA computer programs (for example, NASTRAN, Ref. 15) developed in a continuing research and technology program; others may be developed to fill particular technology gaps in the IPAD approach to design. The status of computerization of the technical capability that resides in such operational modules for preliminary design of large supersonic and subsonic commercial transport aircraft is summarized in Figure 21. It shows that we have a long way to go to achieve the full potential of computerization and that a significant fraction cannot be coded with the present state of the art. Examination of the detailed and final design phases and of this or other vehicles would show a similar picture. Therefore, many gaps must be filled before we can have a fully effective automated design capability. It is also clear, however, that some analyses used in the design process and the extensive testing required will never be completely automated.

A basic objective in the implementation of an IPAD system is the exploitation of the capabilities of a subordinate computing system to enhance the design productivity of a project engineering team. The design environment associated with IPAD will enrich individual participation, encourage more team activity, and encourage greater user creativity. The total work environment will be improved with its

principal outside manifestation in a number of IPAD work and management centers or rooms. The particular room illustrated in Figure 22 is an IPAD executive room for engineering and management reviews that is equipped with a variety of remote terminals and display devices to enhance the evaluation of technical and administrative data. Other rooms with different arrangement of equipment would be used as IPAD workrooms in which interdisciplinary teams could create, review, and change design concepts and details in trade-off and optimization studies as well as in individual specialized design activities. The number and type of rooms required depends on the type and level of projects underway at a particular time.

Figure 23 is a photograph of the Mission Control Center at the Lyndon B. Johnson Space Center in Houston, Texas. It uses many of the features expected in an IPAD executive or workroom. Therefore much software and hardware experience applicable to IPAD is available now.

IPAD BENEFITS

The primary benefit expected from IPAD will be an increase in designer productivity from the utilization of system software and design methods that increase technical capability and creativity and reduce cost and flow time. Reinvestment of time and cost savings can provide better products sooner and cheaper and thus insure greater technical depth before product fabrication. The potential benefits of IPAD have been evaluated by studying the time and labor utilized in the design process, by determining the savings experienced with currently available systems such as those listed on Figure 12, and by estimating the extension of such savings on small tasks to the whole system design task in a large organization. The results are presented in Figure 24 where a range of values are given because characteristics of design projects differ. Manpower costs are classified as technical management, technical judgment, and technical routine with the latter two divided into subgroups. The current distribution of effort and that in IPAD are given along with the estimated cost savings. Note that, as expected, the largest savings accrue in technical routine with no cost savings anticipated in technical management. Total cost savings are estimated at 20% to 60%, depending on the project, with 25% to 90% savings in flow time.

Design productivity trades available to the IPAD user are illustrated in Figure 25. Cost, flow time, and design quality are plotted on three orthogonal axes with an arrow projecting from the origin to indicate a particular combination selected for a design project using current methods. If IPAD could be used, the chief designer would have the additional options indicated by the three other arrows. For the same cost and time, he could increase design quality (vertical arrow) and thereby reduce his company risk in the development of his product. Alternatively, he could reduce cost or flow time for the same design quality (right and left arrows) or he could select some other combination. Thus, IPAD can provide new opportunities to increase the productivity of a design staff in the highly competitive, tight budget environment that exists today.

The development of systems such as IPAD will continue for many years but leave room for even more automation. In addition, the availability of such tools will change the nature of the design process and identify other needs and opportunities for improvement. To prepare ourselves we should conduct research on the design process itself as well as research on design technology. We must learn more about combining men and machines in organizations that will produce the best design quickly and economically. Research on the design process must consider both technical and social factors if we are to reach our optimum design goal. The technical side of new analyses, synthesis methods, and computer hardware and software, is clear-cut. The social part is only partly understood today. A large design organization is a dynamic social system that cannot be managed well without knowledge of the social forces that constantly buffet it. Such knowledge will become more important in the future as members of the design team become increasingly concerned about competition with the computer. More technology is available today for design automation than the social side will accept, References 1 and 19.

CONCLUDING REMARKS

Great progress has been made in the computerization of analysis. Computerization of design is underway and the time and technology are right for exploiting this emerging field. However, design automation must emphasize computer application for enhancing communications and management as well as calculations. Speeding the flow of information (data and decisions) is the next contribution that will enable big design organizations and their man-computer teams to design better, faster, and cheaper. NASA-industry studies have identified an IPAD system that can be developed from existing technology to provide the desired benefits. Additional research on the design process is needed to tell us how to arrange future man-to-man and man-to-computer interfaces to accomplish even more. Surely, automation will increase and gradually change the characteristics of the design process and the designers involved. Certainly any organization that plans to be a leader in design of advanced vehicles and systems of the future must be the leaders in the development and application of automated design processes.

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ACKNOWLEDGMENT

The author wishes to acknowledge the contributions of many individuals that made the writing of this paper possible. In particular, I wish to recognize the efforts of my colleagues in the Structures Directorate at the NASA Langley Research Center and the IPAD study teams led by R. E. Miller, Jr., at The Boeing Commercial Airplane Company and C. A. Garroq at General Dynamics/Convair Aerospace Division.

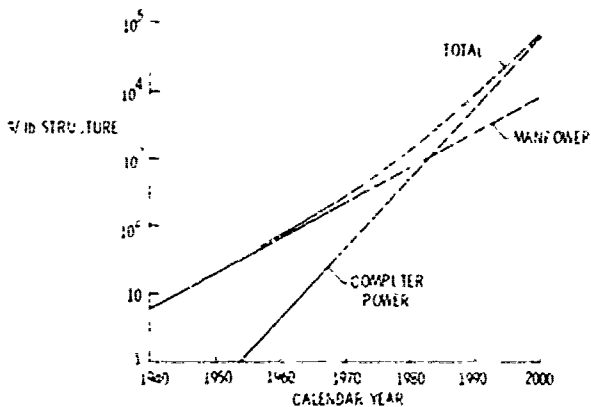


Figure 1. Aircraft structural design costs per pound.

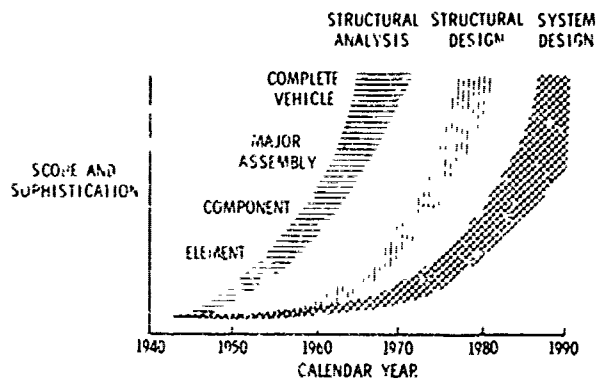


Figure 4. Growth of analysis and design automation.

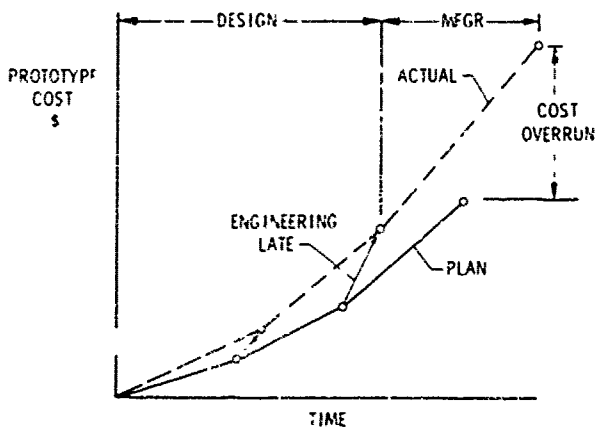


Figure 2. Cost impact of untimely engineering.

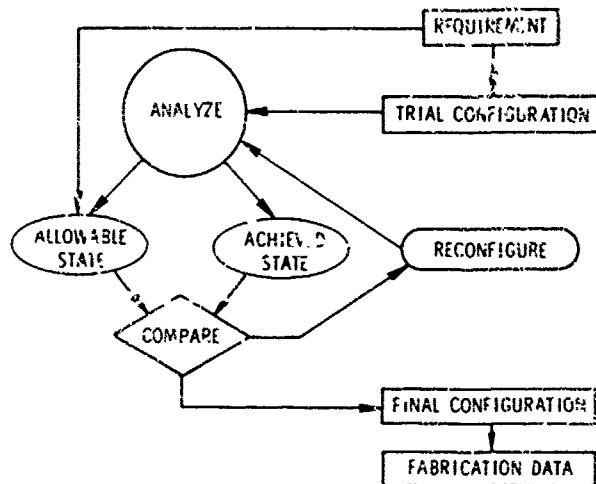


Figure 5. The basic design process.

CHARACTERISTICS OF ADVANCED AEROSPACE VEHICLES

- BIG, COMPLEX, SOPHISTICATED, EXPENSIVE
- INCREASED PERFORMANCE AND ECONOMY
- SENSITIVITY TO STRUCTURAL WEIGHT, ETC.

NATURE OF DESIGN PROCESS FOR LARGE VEHICLES

- COMPLICATED, EXPENSIVE, TAKES A LONG TIME
- MULTITUDE OF DESIGN CRITERIA, DATA, CALCULATIONS
- HIGH RISK OF POOR QUALITY, COSTLY CHANGES

CHARACTERISTICS OF BIG DESIGN STAFFS

- NUMEROUS PEOPLE, INTERFACES, COMMUNICATIONS
- MANAGEMENT COMPLEX AND DIFFICULT

Figure 3. Problems of the design process.

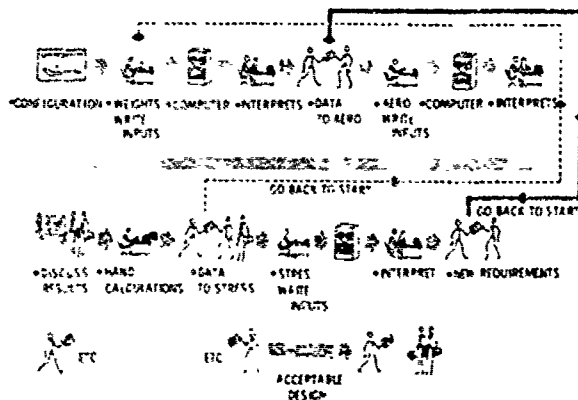


Figure 6. Current design procedures.

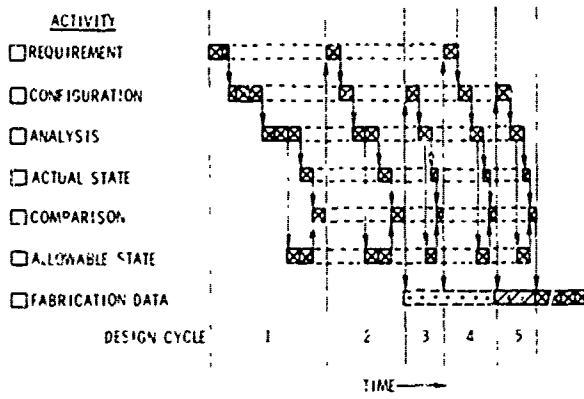


Figure 7. Design process in the time domain.

	CONCEPTUAL	PRELIMINARY	DETAILED
• MANPOWER	1 - 5	10 - 200	200 - 2000
• FLOWTIME	1 - 4 WK	3 - 12 MO	1 - 5 YR
• CONFIGURATIONS EXAMINED	30	150	1
• WEIGHT ACCURACY	< 92	92 - 98	> 98%
• OBJECTIVE	DEFINE MARKET POTENTIAL	SELECT ACCEPTABLE AIRPLANE CONFIGURATION	PREPARE MANUFACTURING & TEST PLANS
• OUTPUT	POTENTIAL AIRPLANE CONCEPT	VERIFIED AIRPLANE CONFIGURATION	SHOP DATA AND DRAWINGS

Figure 10. Definition of design levels.

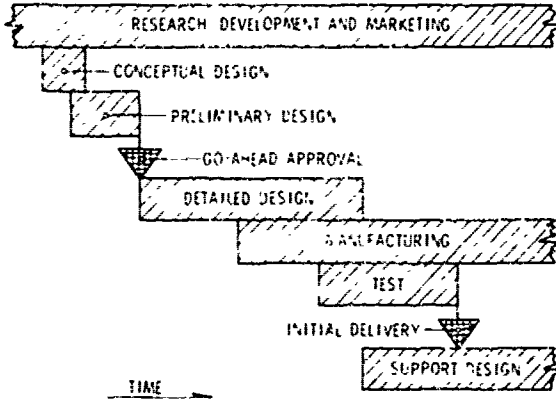


Figure 8. Aerospace product development.

	DESIGN LEVEL		
	CONCEPTUAL	PRELIMINARY	DETAILED
APPLICATION			
• CONFIGURATION OPTIMIZATION	[Hatched bar spanning Conceptual and Preliminary]		
• COMPONENT OPTIMIZATION		[Hatched bar spanning Preliminary and Detailed]	
TYPICAL PARAMETERS			
• WING	AREA	SPAR SPACING	RIVET SPACING
STRATEGY			
• MATHEMATICAL	[Hatched bar spanning all three levels]		
• TRADEOFFS	[Hatched bar spanning all three levels]		
• SEQUENTIAL REFINING		[Hatched bar spanning Preliminary and Detailed]	

Figure 11. Optimization application and strategy.

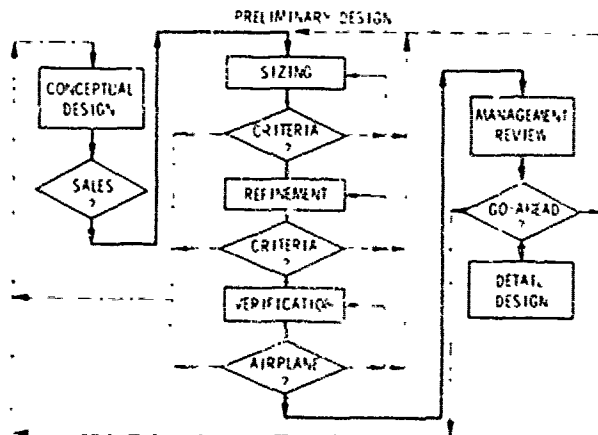


Figure 9. A design decision network.

CONCEPTUAL VEHICLE SYNTHESIS				COMPUTERIZED STRUCTURAL ANALYSIS	
ACVSYN	NASA	AMES		NASTRAN	NASA LANGLEY
ADAM	McDONNELL	DOUGLAS		ASKA	U OF STUTTGART
AVSYN	RYAN	TELEDYNE		FORMAT	USAF FCI
CPM	McDONNELL	DOUGLAS			
CPD	BOEING			AUTOMATIC STRUCTURAL SIZING	
ICAD	USAF	P&D		ASD	McDONNELL DOUGLAS
CDN	AEROPHYSICS	NASA	USAF	ASOP	GRUMMAN
SYNAC	GENERAL DYNAMICS			FADES	NASA - LANGLEY
				SPAR	LOCKHEED
INTEGRATED LOADS AND STRUCTURES					
ANALYSIS			SIZING		
ATLAS	BOEING		ARROW	McDONNELL DOUGLAS	
FAMAS	LOCKHEED		RRAPID	NORTH AMERICAN	
IDEAS	GRUMMAN		DAWNS		
			SAVES		
			SWIFT		
			WIDOWAC	NASA - LANGLEY	

Figure 12. Progress toward automated design in aerospace.

U. S. ARMY

- INTEGRATED WEAPONS SYSTEMS SYNTHESIS MODEL (IWSSM)
INCLUDES SEVEN COMMODITY CLASS (GUN, VEHICLE ETC) VERSIONS

U. S. NAVY

- COMPUTER-AIDED DESIGN ENVIRONMENT (COMRADE)
- INTEGRATED SHIP DESIGN SYSTEM (ISDS)

U. S. MARITIME ADMINISTRATION

- COMPUTER AIDS TO SHIPBUILDING

ARCHITECTS

- ARCHITECTURE MACHINE (URBAN 5)
- COMPUTER-AIDED MODEL FOR ARCHITECTURAL DESIGN (SYNARC)
- SPATIAL ALLOCATION IN DESIGN AND PLANNING (ALOKAT)

CIVIL, BRIDGE & STRUCTURAL ENGINEERS (US & UK)

- COMPUTER-AIDED BUILDING DESIGN SYSTEM
- COMPUTER SYSTEMS FOR BUILDING PLANNING AND DESIGN
- INTEGRATED CIVIL ENGINEERING SYSTEM (ICES)

Figure 13. Progress toward automated design outside aerospace.

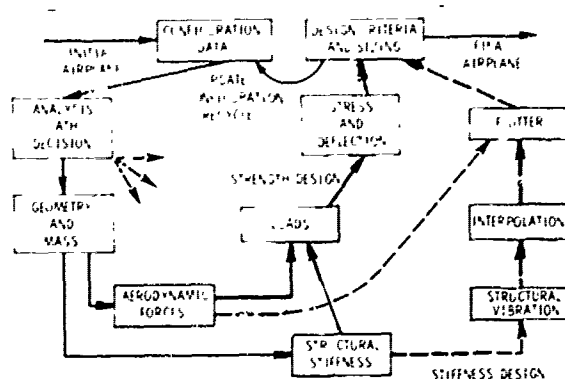


Figure 16. Boeing Integrated Analysis System (ATLAS).

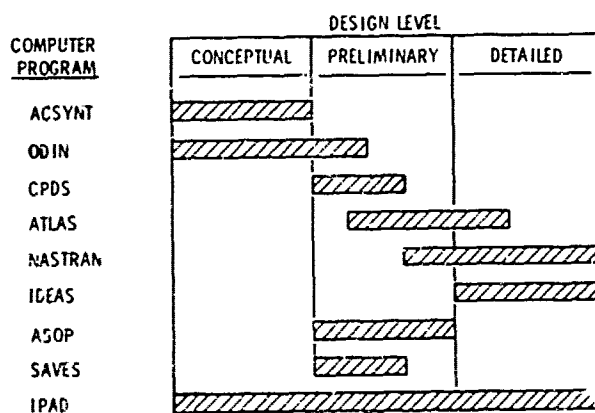


Figure 14. Relationship of aerospace computer programs to design levels.

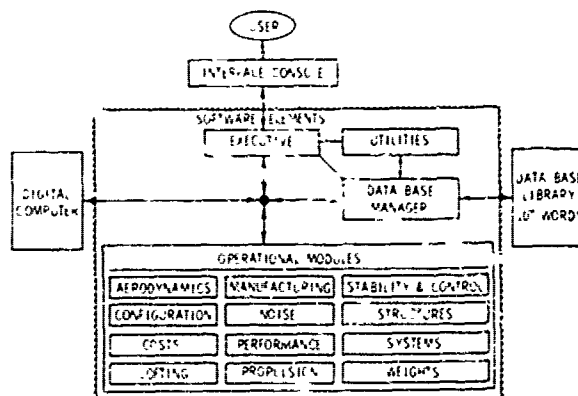


Figure 17. NASA Integrated Programs for Aerospace-Vehicle Design (IPAD).

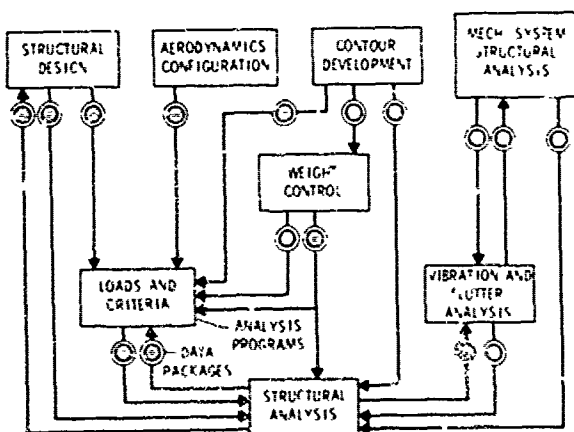


Figure 15. Grumman Integrated Design Analysis System (IDEAS).

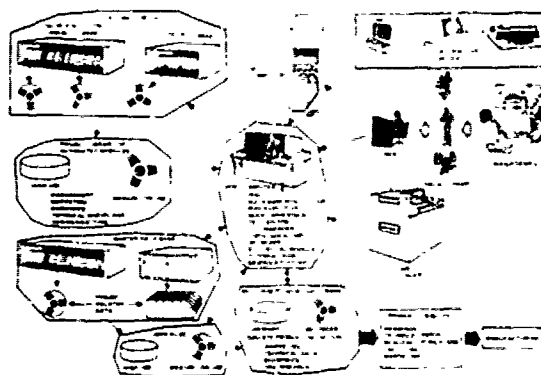


Figure 18. IPAD overview.

- MULTIDISCIPLINARY ENGINEERING DESIGN TOOL TO INCREASE THE EFFECTIVENESS, ECONOMY AND SPEED OF THE DESIGN PROCESS AND ITS MANAGEMENT
- FLEXIBLE, VERSATILE AND OPEN-ENDED SYSTEM OF INTEGRATED MODULAR COMPUTER PROGRAMS FOR DESIGN
 - OF A VARIETY OF VEHICLES OR SYSTEMS
 - AT VARIOUS LEVELS, SCOPES AND DEPTHS OF DESIGN
 - WITH USER-ORIENTED LANGUAGE CONTROL AND RESPONSE
- REASONABLE INITIAL CAPABILITY IN A COMPUTER-AIDED TOOL WITH GROWTH POTENTIAL TO A COMPREHENSIVE DYNAMIC SOFTWARE-HARDWARE SYSTEM

Figure 19. Objectives of IPAD.

SYSTEM DEFINITION

- TWO CONTRACTUAL FEASIBILITY STUDIES (MARCH 1972 - SEPT 1973)
 - DEFINE A FEASIBLE SYSTEM
 - DEFINE A COMPUTER SOFTWARE SYSTEM DESIGN
 - ASSESS COSTS, SCHEDULES, BENEFITS, AND IMPACTS
- EVALUATE SYSTEM WITH INDUSTRY PARTICIPATION (1973-74)

SYSTEM DEVELOPMENT

- DEVELOP CORE SOFTWARE (1975-80) (EXECUTIVE, DATA BASE MANAGER, UTILITIES)
- COLLECT AND DEVELOP SELECTED OPERATIONAL MODULES (1970-)
- CHECKOUT AND RELEASE TO INDUSTRY (1980)

Figure 20. Status and plans of IPAD.

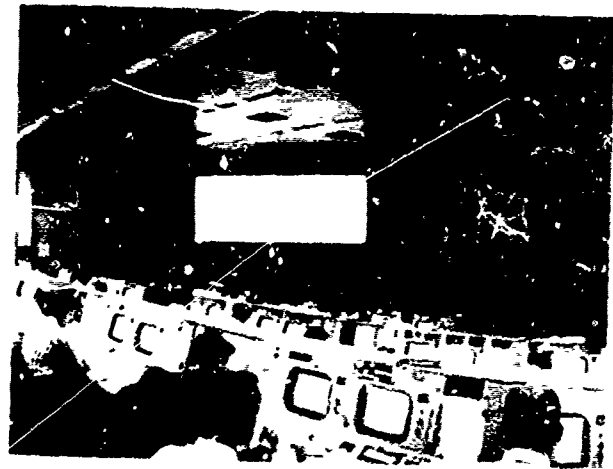


Figure 23. Mission control at NASA-JSC.

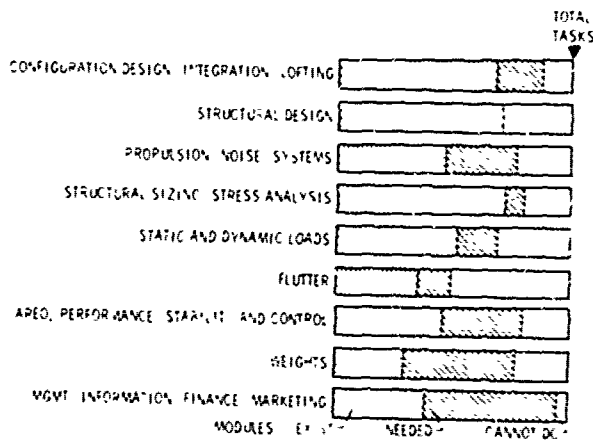


Figure 21. Technical capability for automated preliminary design of subsonic and supersonic commercial transports.

MANPOWER COST	DISTRIBUTION OF EFFORT		IPAD SAVINGS
	CURRENT	IPAD	
□ TECHNICAL MANAGEMENT	6	10 - 15	0%
○ TECHNICAL JUDGEMENT	34	35 - 65	20%
• PROCEDURE DEVELOPMENT	12		
• CALCULATIONS	6		
• RESULTS EVALUATION	13		
□ TECHNICAL ROUTINE	60	55 - 20	25 - 90%
• INFORMATION EXCHANGE	26		
• DATA PREPARATION	34		
◇ TOTAL	100	100	20 - 60%
FLOWTIME			25 - 90%

Figure 24. Potential savings from IPAD.

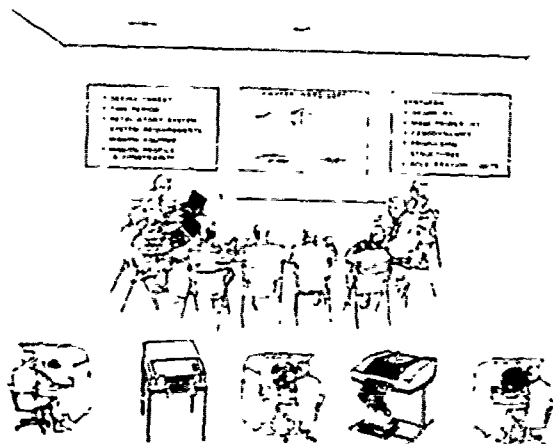


Figure 22. IPAD executive room.

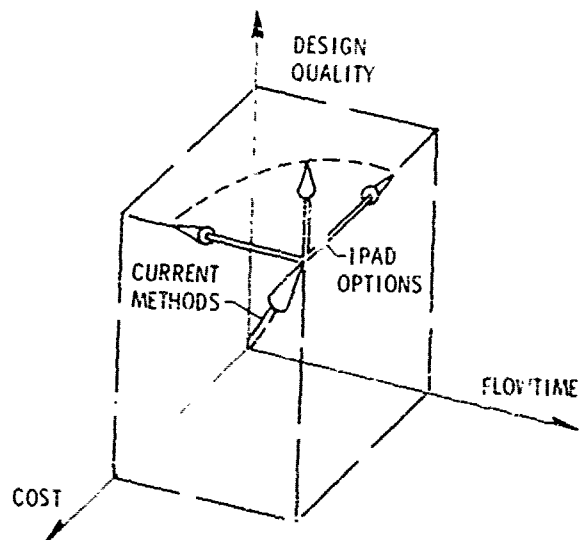


Figure 25. IPAD design productivity trades.

PROJECT WEIGHT PREDICTION BASED ON ADVANCED STATISTICAL METHODS

by

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SUMMARY

This lecture is intended to give a survey of the possibilities of mathematical statistics for engineering evaluation of reliable data sets for design weight estimates of first level accuracy

In order to choose quasi orthogonal weight parameters, the application of partial correlation analysis, which is based on extensive idealization of the physical relationships between weight and design variables is proposed

Proceeding from a classical form of regression analysis, several statements specially adapted for finding weight prediction formulas will be described

- constraint regression for development formulas which give physically interpretable weight trends, using methods of quadratic optimization
- non linear regression statements which will be solved by the aid of iterative computer routines

For the purpose of comparing the developed formulas with existing procedures, several methods of error probability theory will be discussed

Some practical examples will show the possibilities of applying statistical evaluations.

RESUMÉ

L'emploi de méthodes statistique avancées pour déterminer le poids du projet.

L'exposé donne un aperçu des possibilités offertes par la statistique mathématique dans le domaine de l'exploitation par les techniciens de collectes de données au sujet du calcul du poids d'un projet du 1er degré de précision

Pour les choix des paramètres quasi orthogonaux ayant une influence sur le poids, il est proposé d'employer l'analyse partielle en corrélations, basée sur une idéalisation extensive des rapports physiques entre le poids et les paramètres du projet

A partir d'une forme classique de analyse régressive, différentes dispositions adaptées spécialement au calcul de formules d'estimation du poids seront décrites quant à leur utilisation pratique

- régression "forcée" pour le développement de formules donnant des tendances de poids interprétables physiquement, en utilisant des méthodes d'optimisation quadratique
- dispositions en régression non linéaires, résolues à l'aide de routines d'itération programmées.

Différentes méthodes de la théorie des probabilités d'erreur seront discutées dans le but de comparer les formules développées avec les procédés connus

Des exemples pratiques permettront de déterminer les possibilités et limites d'application de méthodes d'exploitation statistique.

NOTATIONS AND ABBREVIATIONS

A	matrix of coefficients for the normal equation system of the multiplicative regression formulation	C(j)	exponents or coefficients of various regression statements
a, a_{1,2}	error scatter parameters for error mean and standard deviation	C_s	shear web coefficient
S_{Yq}	"coefficient of partial determination"	d	local wing box depth
b	vector of the estimates for β (problem of non-linear regression) vector of the normal equation (multiplicative regression statement)	E	statistical expectation
b	wing span	e	constraint coefficient for elastic problem
b_R	average geometrical wing rib spacing	f	error mean for a statistical estimating function
C	solution vector of the classic regression problem	F_B	required cross section of the wing bending material
C_d	compression panel coefficient	F'_B	proportion of F _B which is determined by the loading
C_i	general constants	F_{BD}	required cross section of the compression panel
		F_{BZ}	required cross section of the tensile panel
		F_i	planform area of various secondary structure components

\overline{F}_{RF}	effective rib flange cross section	n_{Br}	ultimate load factor
\overline{F}_{RS}	effective rib shear web cross section	O	zero matrix
F_s	required cross section for the wing shear webs	P	statistical probability measure for the validity range of a statistical estimating function
F'_s	proportion of F_s which is determined by the loading	P	number of weight influence parameters
F_{th}	wing reference area	P	matrix of all discrete weight parameters values in a sample
$f(i)$	individual error	P'_F	effective wing flange load
G	general notation for weight	Q_{max}	max. wing shear load
g	vector of all observed weights	$\frac{Q}{h_b^2}$	structural index for shear loading
$g(i)$	logarithm of an individual weight estimate	R	statistical valuation parameter
G_{Amax}	max. take-off weight	R	matrix of the two-dimensional correlation coefficients
G_B	stress cross design weight	r	correlation coefficient
G_{BI}	weight of the bending material	S	error standard deviation for a statistical estimating function
G_{FL}	wing weight	SAC	sum of deviation squares
G_{RF}	weight of the wing rib flanges	t	statistical probability distribution
G_{RS}	weight of the wing rib shear webs	t_{R0}	minimum rib thickness
G_{SEK}	weight of the wing secondary structure	U	vector of N LAGRANGE multipliers
g_j	specific weight (weight per unit area)	V_B'	bending material proportion determined by the loading
h	effective wing box depth	V_s'	shear material proportion determined by the loading
i	index running over all observations in a sample	W	vector of additionally introduced unknown variables (WOLFE method)
I	unit matrix	X	general vector
J_{nb}	mass relief coefficient (bending) $J_{nb} = J_{ns} \cdot J_s$	X, Y	general variables
J_{ns}	mass relief coefficient (shear)	$Z^{1,2}$	vector of additionally introduced unknown variables (WOLFE method)
J_s	shear reduction factor	α_j	exponent of various regression statements
j, k	index running over all weight influence parameters	β	solution vector for the problem on non-linear regression
K, K_j	general constants	γ	density confidence level
k_{ba}	lift distribution coefficient	δ	wing thickness ratio
k_{ib}	integration factor (bending)	$\bar{\delta}$	equivalent thickness ratio $\bar{\delta} = \frac{3\delta_w + \delta_a}{4}$
k_{is}	integration factor (shear)	ϵ	vector of the individual deviations between estimated and actual weight for a data sample
k_{NO}	non-optimum factor	ϵ	wing box chord ratio $\epsilon = \frac{l_{HK}}{l}$
k_s	shear web depth coefficient	Θ	vector of all unknown variables for the problem of non-linear regression
L_0	effective wing rib spacing $L_0 = \frac{\Delta b_R}{\sqrt{\epsilon}}$	κ	rib spacing ratio $\kappa = \frac{\Delta b_R}{l}$
l	local wing chord	Λ	wing aspect ratio $\Lambda = \frac{b^2}{F_{th}}$
l_{HK}	local wing box chord	λ	wing taper ratio $\lambda = \frac{l_a}{l_i}$
l_u	aerodynamical mean chord of the wing	μ	mean value of a normal distribution
m_{RB}	rib bending stress ratio	σ	standard deviation of a normal distribution
m_s	rib shear stress ratio		
N	number of observations in a sample		
n	number of weight influence parameters		

- σ_{B_0} equivalent wing bending stress
- σ_{D_0} basic compression stress
- $\sigma_{D_{zul}}$ allowable compression stress
- σ_{RB_0} equivalent rib bending stress
- $\sigma_{Z_{zul}}$ allowable tensile stress
- τ_{s_0} basic shear stress
- $\tau_{s_{zul}}$ allowable shear stress
- φ_{50} sweep angle of the wing box center line
- Φ objective functions in an optimization process
- χ^2 statistical probability distribution

ABBREVIATIONS

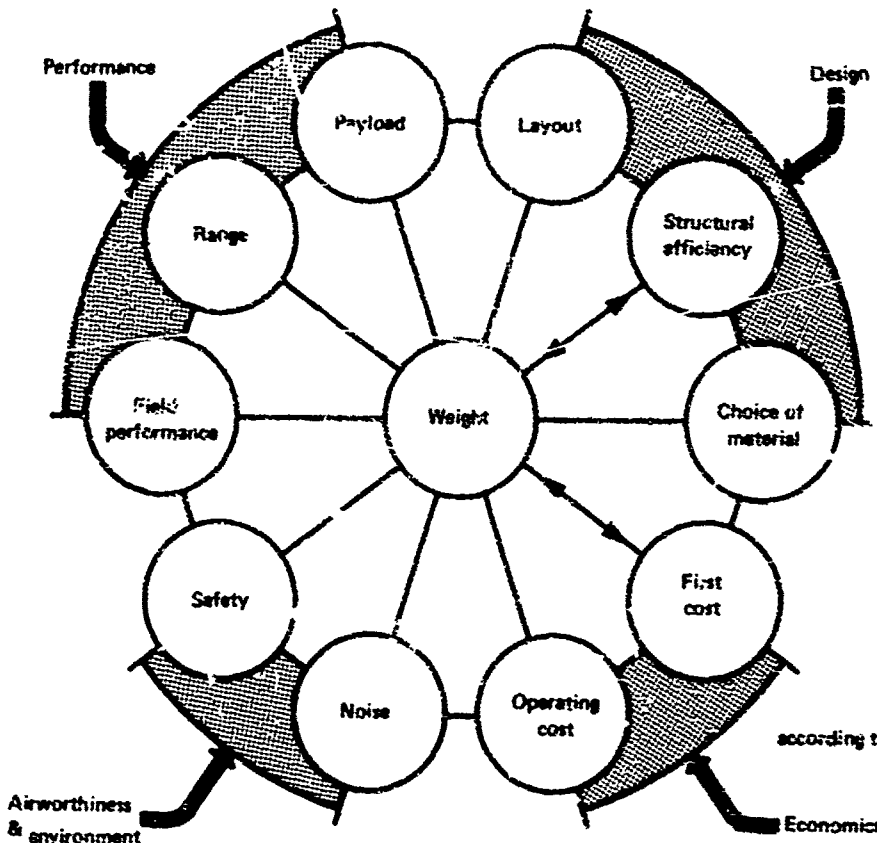
- T wing tip
- I cross section in the aircraft symmetry area
- x dimensions in the structural coordinate system
- W wing root

1. INTRODUCTION

The general efforts to minimize the achievable weight are one of the basic prerequisites for the realization of efficient aircraft design. The weight of an aircraft is one of the most important efficiency parameters, since the criteria

- cost
- structural efficiency
- flight performances
- safety and reliability

are in direct correlation to the weight (see Fig. 1).



according to: A.A. BLYTHE, HSA
SAWE-Paper No. 996

Fig. 1 THE HUB OF THE WHEEL

Due to these relationships, the aircraft engineer is compelled to predict the weight of the vehicle to be designed and constructed with high accuracy as early as the preliminary stage. The variety of present weight estimating procedures can be broken down into two major groups:

Analytical methods following the sequence:

- Definition of the outer geometry (in accordance with payload requirements, aerodynamics, engine installation etc.)
- Estimating of the critical loads or of numerical values for other design criteria (e.g. power requirement for the electrical system)
- Fixing of the structural layout in principle
- Prediction of the critical cross sections for the primary components in accordance with existing strength, stiffness or operational requirements
- Integration of the material concentration locally necessary, taking into consideration existing manufacturing restrictions.

Statistical evaluations of weight data of comparable existing aircraft.

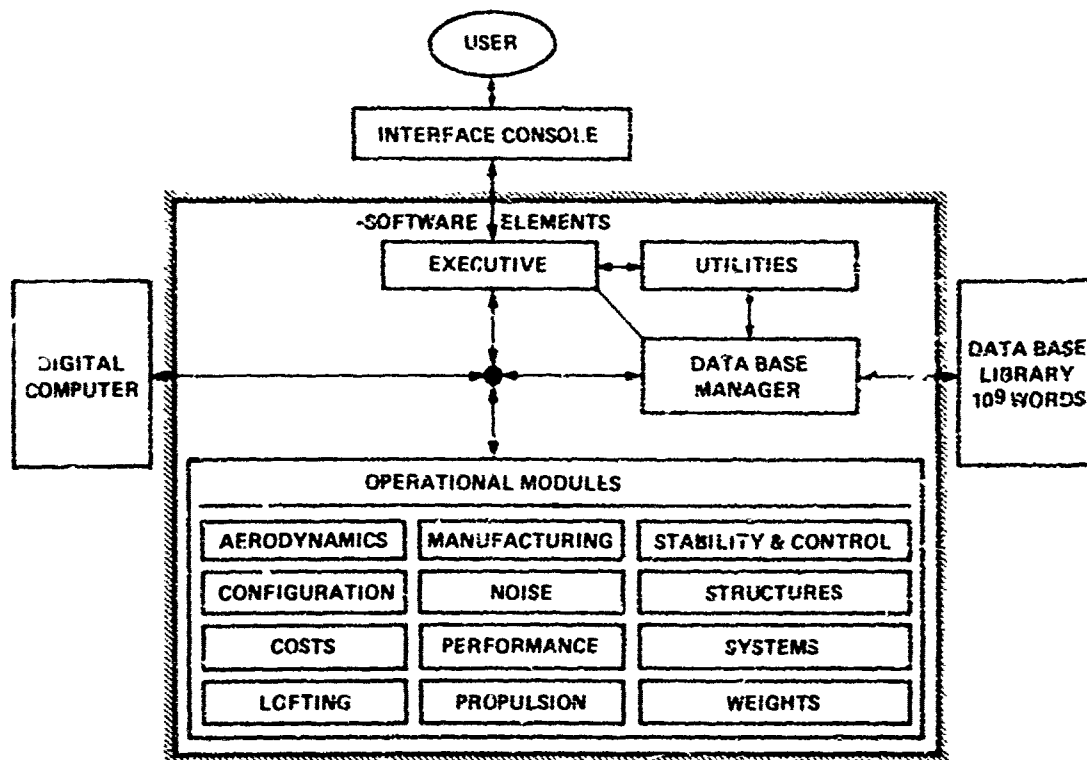
The first methods are not practicable in the earliest design phase (the concept phase) because the majority of the required input data are still unknown and estimated can be carried out only in part. In addition, the expense necessary for the numerical calculations is very high, so that it is not possible to get initial results very quickly, even where a high speed computer is available.

The second group of procedures has its own restrictions too, but it allows a bypassing of the complex analytical problems in practice, if all possibilities of mathematical statistics are used very consistently by correlating the numerical values of design parameters and weights of contemporary aircraft.

At the present time, such types of estimating formulas are a fundamental tool of the project engineer in many technical fields. Since very often, much empirical experience is included in the methods for developing such "rules of thumb" themselves, they have so far been published very seldom.

This lecture is intended to give a rough survey of the advanced possibilities of statistical evaluation methods using high speed computers to increase the cost effectiveness of the design process. Furthermore, the purpose of this paper is to show that very often essential mathematical assumptions of statistical analysis are disregarded in practical applications, resulting in considerable errors in many cases.

In addition, this paper should highlight the problem area of module development for computer-aided design synthesis programs. Although the weight prediction subroutines are only a small part of the total process, this lecture should illustrate to what extent all detailed problems must be investigated to get results of reasonable accuracy. If it is taken into consideration that the presented methodology only offers the possibility of obtaining estimating functions of first level accuracy, then it will be obvious what restrictions still exist for applying computer-aided design programming to the automation of the preliminary design process. Especially for the investigation of unconventional designs it seems necessary to combine automated computer routines with human engineering experience. At present it is not possible to allow for all complex non-linear influences on the design iteration process (e.g. aeroelastic effects and instationary aerodynamics). The flow chart of a typical aircraft design program as described by R.R. Heldenfels during this symposium is shown in Fig. 2.



According to: R. R. Heldenfels:
Integrated Computer-Aided Design of Aircraft AGARD-Paper, F/M-Symposium in Oct. 1973 at Florence, Italy

Fig. 2 NASA INTEGRATED PROGRAMS FOR AEROSPACE-VEHICLE DESIGN (IPAD)

2. BASIC ASSUMPTIONS AND CONCEPTION OF THE METHODOLOGY

The aims of this investigation are:

- to work out a new methodology for the development of statistical weight prediction formulas of first level accuracy for the preliminary aircraft design stage to improve existing methods,
- to create the possibility of deriving empirical correction coefficients for analytical weight calculation procedures, taking into consideration influences which cannot be covered analytically.

For this purpose the following prerequisites must be fulfilled:

- an extensive collection of reliable data of comparable existing aircraft must be available,
- the set of statistical estimating formulas to be developed will be applied only for new designs which correspond to the contemporary technological standard
- the application of the developed formulas for weight trend evaluations is limited to minor partial changes of the various weight parameters,
- for control of the numerical work, a high speed computer must be available.

It should be emphasized explicitly that it is necessary to expend much effort in preparing the basic data set since even very sophisticated statistical methods cannot compensate for the lack of non-representative data samples

The proposed methodology can be broken down into three main categories:

- Selection of significant weight influence parameters
- Prediction of the estimating functions
- Objective valuation of alternative formulas

A more detailed arrangement is shown in Fig. 3.

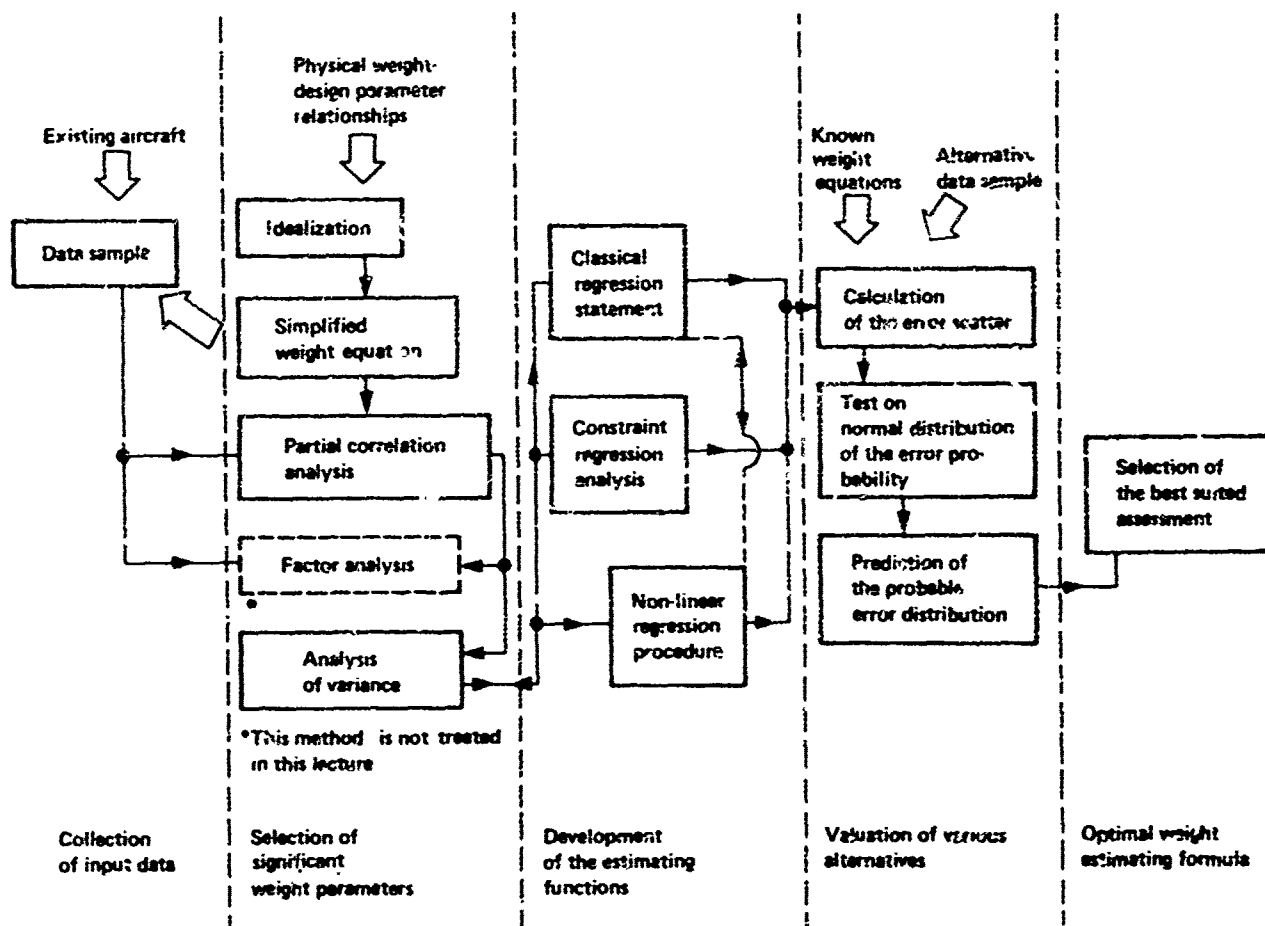


Fig. 3 FLOW CHART OF THE PROPOSED METHODOLOGY FOR DEVELOPING STATISTICAL WEIGHT ESTIMATING FORMULAS

3. SELECTION OF SIGNIFICANT WEIGHT PARAMETERS

Before we look for suitable formulations for the estimating formulas containing the design variables which are available in the preliminary stage, the ones which have the most significant influence on the weight must be selected.

3.1 Graphical Methods

In the past, it was common practise to seek weight relationships in an empirical manner, plotting the weight of the investigated component versus various single or combined parameters. Such time-consuming trials have only very seldom been successful, since just a few variables can be taken into consideration simultaneously in a graph. Therefore, the multiple functional connections cannot be analyzed in this way.

Furthermore, physically incorrect tendencies will be observed in trend analysis for such empirically developed formulas. The reason is, that along these trend curves the variations of other important parameters are suppressed. As an illustration, the wing weight of existing transport aircraft is plotted versus the wing root thickness ratio in Fig. 4. The structural aspect ratio is taken as an additional influence parameter. For reasons of comparison, a weight curve for constant Λ_s planform dimensions and loading has been predicted using an analytical method. It is obvious that the comparable empirically developed trend curve has incorrect local gradients and a physically unexplainable minimum location. This is caused by the neglected influence of other important parameters, e.g. the load level and the geometrical configuration for data point 25 are completely different from those for point 4.

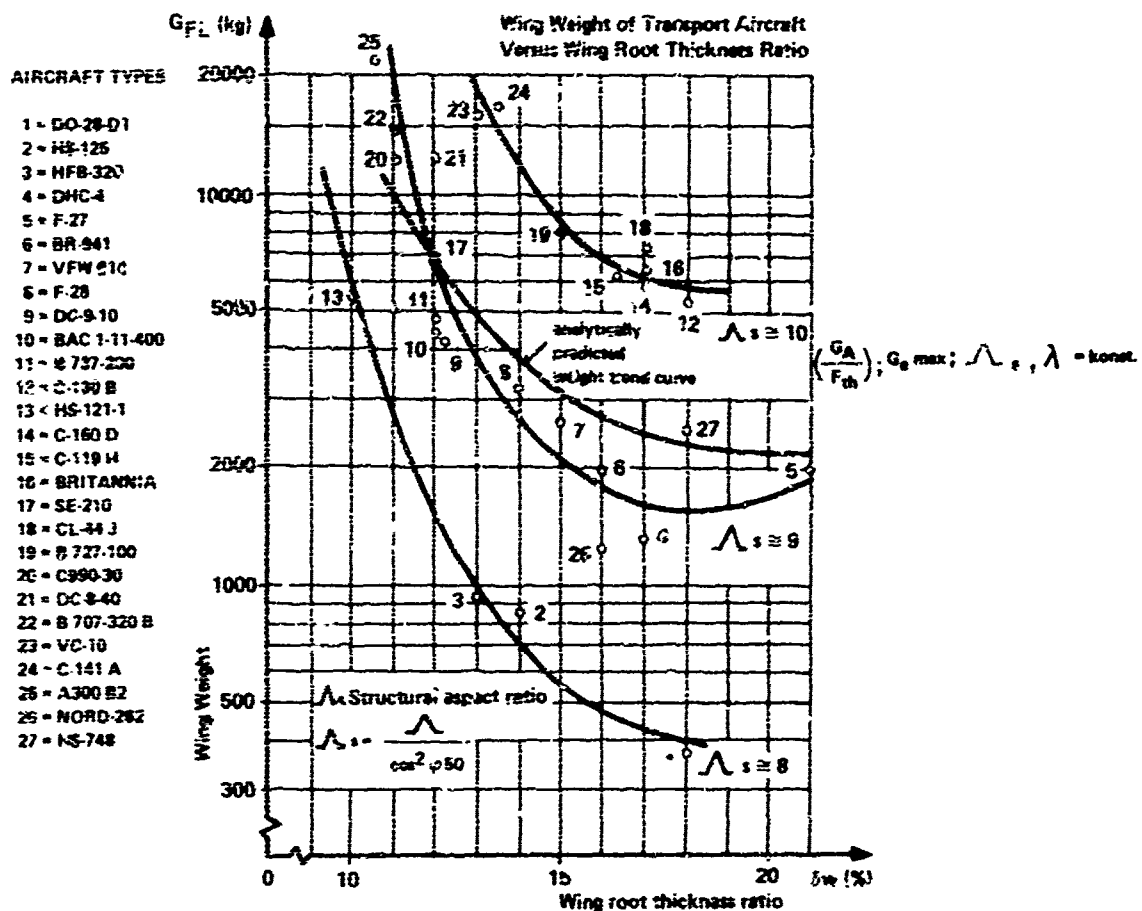


Fig. 4 EMPIRICALLY DEVELOPED WEIGHT RELATIONSHIP

3.2 Physical-analytical parameter selection

An alternative method is the physical-analytical parameter selection. For this purpose, the investigated weight group will be idealized in such a manner that the relationships between the weight and various parameters can be approximated by simple functions. All influences which cannot be covered directly will be expressed by unknown exponents and coefficients to be predicted in the following regression analyses.

Since a general procedure cannot be suggested the proceeding will be explained in principle through the example of the wing weight estimate. The wing weight is assumed as the total of main 5 components:

$$G_{FL} = G_{BI} + G_S + G_{RF} + G_{RS} + C_{Sek} \quad (1)$$

- G_{BI} Bending material (wing box cover panels);
 G_S Shear material (box webs)
 G_{RF} Rib flanges
 G_{RS} Rib shear webs
 G_{Sek} Secondary structure (flaps, leading edge etc.)

In the next step, simplified relationships will be developed for all these terms containing only variables, which are available in the preliminary design phase.

As an illustration, a brief description of the estimate of shear and bending material is given here:

The required cross section of the shear load transferring box webs at root position is:

$$F_{S_w} = \frac{Q_{max,w}}{\tau_{S_{zul}}} \quad (2)$$

Furthermore:

$$Q_{max,w} = \frac{1}{2} \cdot J_{RS} \cdot J_S \cdot n_{BR} \cdot G_B \quad (3)$$

($n_{BR} \cdot G_B$) is the design load level (max. load factor multiplied by stress gross design weight), J_{RS} takes into consideration the wing mass relief and J_S is a measurement for the shear load reduction in the webs due to the box depth taper. The allowable shear stress depends primarily on the shear structural index as shown in Fig. 5. The relationship between shear stress and structural index can be linearized very simply by dividing the various abscissa values in Fig. 5 by the abscissa values in Fig. 5 by the appropriate allowable shear stresses:

$$\frac{F_S}{h_s^2} = \frac{Q/h_s^2}{\tau_{S0}} + C_S \quad (4)$$

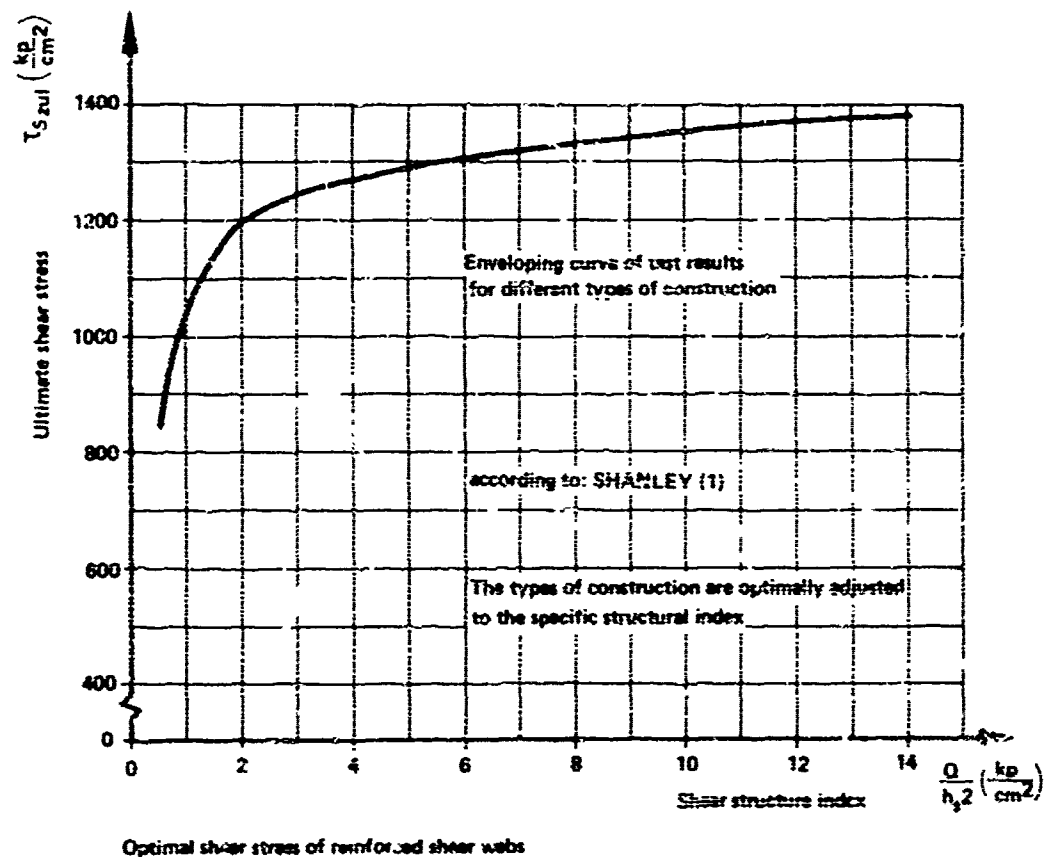


Fig. 5 PHYSICAL-ANALYTICAL WEIGHT PARAMETER SECTION

C_s and τ_{s0} in Eq. (4) are typical constants. The required wing root shear web cross section will then be:

$$F_{S_w} = 2 \cdot C_s \cdot k_s^2 \cdot d_w^2 + \frac{J_{ns} \cdot J_s \cdot r_{Br} \cdot G_B}{2 \cdot \tau_{s0}} \quad (5)$$

Assuming a suitable cross section distribution function, the shear material weight results from the integration along the structural span:

$$G_s = \gamma \cdot k_{NO} \cdot \left\{ 2 \cdot C_s \cdot k_s^2 \cdot \delta_s^2 \cdot l_{\mu_s} \cdot F_{th} + \frac{n_{Br} \cdot G_B}{2 \cdot \sigma_{Ba}} \cdot b_s \cdot \frac{J_s \cdot J_{ns} \cdot k_{is}}{2 m_s} \right\} \quad (6)$$

k_{NO} , the so called "non-optimum" factor, represents the average ratio of the shear web weights, calculated for existing aircraft with Eq. (6) ($k_{NO} = 1.0$) to the actual weights. k_{is} , the integration factor, contains the ratio of the average integral value of the actual lift distribution to that of constant loading per span unit.

In a similar manner the bending material cross section at the root can be expressed by:

$$F_{B_w} = P F' \cdot \left(\frac{1}{\sigma_{Z_{ul}}} + \frac{1}{\sigma_{Z_{dl}}} \right) \quad (7)$$

The allowable stress levels can be estimated as:

$$\text{tensile: } \sigma_{Z_{ul}} \approx 0.8 \cdot \sigma_{Z_{Br}} \quad \text{compression: } \sigma_{D_{Z_{ul}}} = f \left(\frac{N}{L_0} \right) \quad (8)$$

Applying an enveloping curve of test data (similar to Fig. 5) the required cross section for compression follows from:

$$F_{B_{d_w}} = c_d \cdot L_0 \cdot I_{HK_w} + \frac{P F_w}{\sigma_{D_0}} \quad (9)$$

Both allowable stress levels can be combined by definition into an "equivalent bending stress" σ_{Ba}

$$F_{B_w} = 2 \frac{P F'}{\sigma_{Ba}} \quad \sigma_{Ba} = \frac{2 \sigma_{D_0} \cdot \sigma_{Z_{ul}}}{\sigma_{D_0} + \sigma_{Z_{ul}}} \quad (10)$$

The effective flange load can be expressed as:

$$P F_w = \frac{1}{4} \cdot \frac{K_{Da}}{K_e} \cdot J_{nb} \cdot \frac{b}{d_w} \cdot n_{Br} \cdot G_b \quad (11)$$

The total bending material cross sections:

$$F_{B_w} = F_{B_{d_w}} + F_{B_{Z_w}} \quad (12)$$

Using the Eq.'s (9) to (12) the result is:

$$F_{B_w} = c_d \cdot I_{HK_w} \cdot \frac{\Delta b_R}{\sqrt{e}} + \frac{1}{2} \cdot J_{nb} \cdot \frac{K_{Da}}{K_e} \cdot \frac{b/2}{d_w} \cdot \frac{n_{Br} \cdot G_B}{\sigma_{Ba}} \quad (13)$$

In a similar manner (for details see SHANLEY [1]), the root cross sections for the rib shear webs and the rib flanges can be predicted as:

$$\overline{F_{RSW}} = \overline{I_{R_0}} \cdot \frac{d_w \cdot I_{HK_w}}{\Delta b_R} + \frac{n_{Br} \cdot G_B \cdot I_{HK_w} \cdot l_w}{2 F_{th} \cdot \tau_{s0}} \quad (14)$$

$$\overline{F_{RFW}} = \frac{n_{Br} \cdot G_B \cdot I_{HK_w} \cdot l_w}{4 \cdot d_w \cdot F_{th} \cdot \sigma_{RB_s}} \quad (15)$$

The weight of the secondary structure is estimated by:

$$G_{SEK} = g_{SEK} \cdot F_{th} \quad \text{where:} \quad g_{SEK} = \sum_i g_i \cdot \left(\frac{F_i}{F_{th}} \right) \quad (16)$$

The various g_i are typical specific unit weights (kg/m²) of the secondary structural components (trailing edge flaps, ailerons, leading edge etc.).

The following prediction steps can be simplified by introducing several dimensionless ratios:

$$\delta_S = \frac{d}{l \cdot \cos \varphi_{50}}; \quad \epsilon = \frac{l_{HK}}{l}; \quad \kappa_S = \frac{\tau_{S0}}{\sigma_{Ba}} \dots \dots m_s = \frac{\tau_{S0}}{\sigma_{Ba}} \quad (17)$$

The total bending material weight then results from the integration of the locally required cross section. For further simplifications a reasonable distribution of the load depending on the cross section area is assumed and several variables are replaced by typical mean values:

$$\bar{v}_S = \frac{3\delta_{SW} + \delta_{Sa}}{4} \dots \quad V_{Bges} = \frac{1}{3} K_{ib} \cdot FBW \cdot b_S \quad (18)$$

Combining Eq. (12) (13) and (18) the wing box bending material is given by:

$$G_B = \gamma \cdot k_{no} \cdot \left\{ C_d \cdot \bar{\epsilon} \cdot \frac{\bar{\kappa}_S}{\sqrt{\bar{\epsilon}}} \cdot l_{\mu_S} \cdot F_{th} + \frac{n_{Br} \cdot G_B}{2 \cdot \sigma_{Ba}} \cdot b_S \cdot \left[\frac{J_{nb} \cdot K_{ib} \cdot K_{ba}}{3 \cdot K_e} \cdot \frac{b_S/2}{d_w} \right] \right\} \quad (19)$$

The total wing weight follows from the addition of all five components:

$$G_{Fi} = \gamma \cdot K_{NO} \cdot \left\{ \left[(2c_s \cdot K_s^2 \cdot \bar{\delta}_S^2 + c_d \cdot \bar{\epsilon} \cdot \frac{\bar{\kappa}_S}{\sqrt{\bar{\epsilon}}}) \cdot l_{\mu_S} + \frac{\bar{\delta}_S \cdot \bar{\epsilon}}{K_S} \cdot \bar{r}_{Ro} \right] \cdot F_{th} + \frac{n_{Br} \cdot G_B}{2 \cdot \sigma_{Ba}} \cdot \left[\left(\frac{J_s \cdot J_{ns} \cdot k_{is}}{2m_s} + \frac{J_{nb} \cdot K_{ib} \cdot K_{ba}}{3 \cdot K_e} \cdot \frac{b_S/2}{d_w} \right) \cdot b_S \right. \right. \\ \left. \left. + \bar{\epsilon} \cdot \left(\frac{1}{m_s} + \frac{\bar{\epsilon}}{2m_{RB} \cdot \delta_S} \cdot l_{\mu_S} \right) \right] \right\} + g_{SEK} \cdot F_{th} \quad (20)$$

This wing weight equation can be approximated by:

$$\hat{G} = C_1 \cdot p_1^{\alpha_1} + \dots + C_j \cdot p_j^{\alpha_j} + \dots + C_n \cdot p_n^{\alpha_n} \quad (21)$$

The unknown C_j and α_j cover the effects of the idealization. The p_j are not necessarily single variables, but can include the functional interdependence of several variables such as:

$$\left. \begin{aligned} P_1 &= (K_1 \cdot k_s^2 \cdot \bar{\delta}_S^2 + \bar{\epsilon} \cdot \bar{\kappa}_S) \cdot l_{\mu_S} \cdot F_{th} \\ P_2 &= \frac{\bar{\delta}_S \cdot \bar{\epsilon}}{K_S} \cdot F_{th} \\ P_3 &= \frac{n_{Br} \cdot G_B}{\sigma_{Ba}} \cdot \left(\frac{J_s \cdot J_{ns} \cdot k_{is}}{m_s} + \frac{J_{nb} \cdot K_{ib} \cdot K_{ba}}{3 \cdot K_e} \cdot \frac{b_S}{d_w} \right) \cdot b_S \\ P_4 &= \bar{\epsilon} \cdot \left(\frac{1}{m_s} + \frac{\bar{\epsilon} \cdot l_{\mu_S}}{2m_{RB} \cdot \delta_S} \right) \cdot \frac{n_{Br} \cdot G_B}{\sigma_{Ba}} \\ P_5 &= g_{SEK} \cdot F_{th} \end{aligned} \right\} \quad (22)$$

$0 < \alpha_j < 2.0$

This statement is not used for regression curve fitting in contemporary practise because its application produces considerable numerical difficulties.

An alternative approximation of the physical weight (equations (20)) is:

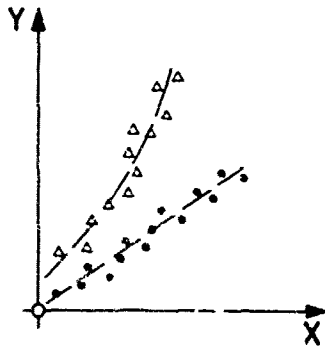
$$\hat{G} = \bar{C}_1 \cdot a_1^{a_1} \cdot p_2^{a_2} \cdot \dots \cdot p_n^{a_n} \quad (23)$$

Using an expansion into a TAYLOR series for the various terms ($a_i^{a_i}$) it is apparent that at least for the most important parts conformity exists between the approximation (Eq. (23)) and the simplified physical relationship, (Eq. (20)).

3.3 Correlation analysis

Supplemental to this procedure the application of the correlation analysis is proposed:

This method has been used very seldom in this form in the technical field. Using the correlation coefficient r , the quality of a linear functional relationship between two variables Y and X can be expressed as:



$$r_{yx} = \frac{\sum_{i=1}^N (x_i - \bar{x}) \cdot (y_i - \bar{y})}{\sqrt{\sum_{i=1}^N (x_i - \bar{x})^2 \cdot \sum_{i=1}^N (y_i - \bar{y})^2}} \quad (24)$$

Most of the weight design parameter functions are non linear, but can be approximated very well by exponential functions. Linearization by taking the logarithm before the prediction of r is therefore necessary.

$$y = C_0 \cdot x^\alpha \quad \tilde{y} = \ln y = \ln C_0 + \alpha \cdot \ln x = K_0 + \alpha \cdot \tilde{x} \quad (25)$$

Accordingly, r does not describe the kind of functional relationship between X and Y, but rather clarifies whether interdependency exists at all.

Consideration of the total (2-dimensional) correlation coefficient r_{xy} alone however is not sufficient, because both X and Y are influenced by other variables x_j according to the mutual partial correlation is also influenced. By predicting the partial correlation coefficient the functional relationship between the objective variable (the weight) and a weight influence parameter is freed from the influence of the other design variables. In Eq. (26) the variables behind the point in the index of r designate the eliminated variables.

$$r_{y x_j \cdot x_1 \dots x_{j-1} \cdot x_{j+1} \dots x_{n-1}} = \frac{r_{y x_j \cdot x_1 \dots x_{n-2}} - r_{y x_{n-1} \cdot x_1 \dots x_{n-2}} \cdot r_{x_j x_{n-1} \cdot x_1 \dots x_{n-2}}}{\sqrt{(1 - r_{y x_{n-1} \cdot x_1 \dots x_{n-2}}^2) \cdot (1 - r_{x_j x_{n-1} \cdot x_1 \dots x_{n-2}}^2)}} \quad (26)$$

The following matrix operation simplifies the numerical calculation:

The symmetrical matrix R of all 2-dimensional correlation coefficients is created, whereby $r_{jk} = r_{kj}$. The diagonal of this matrix is built up by ones, since in accordance with the definition of r , the correlation of a variable with itself is equal to one.

$$R = \begin{bmatrix} 1 & r_{12} & r_{1k} & \dots & r_{1n} \\ r_{21} & 1 & & & \\ r_{j1} & & 1 & & \\ r_{n1} & & & & 1 \end{bmatrix}$$

R must be inverted in the following step. For the numerical computation the method of CHOLESKY [2] is proposed. If the matrix R has larger dimensions, a digital computer must be used, but many software routines are available for this purpose. Using Eq. (27) the various partial correlation coefficients can be predicted from the elements of the inverted matrix R^{-1} .

$$R^{-1} \cdot R = I \quad R^{-1} = \begin{bmatrix} r'_{11} & & \\ & r'_{jj} & \\ & & r'_{yy} \end{bmatrix} \quad r_{y x_j \cdot x_1 \dots x_{n-1}} = \frac{r'_{jy}}{\sqrt{r'_{jj} \cdot r'_{yy}}} \quad (27)$$

With this method even highly multiple relationships can be analyzed qualitatively. For the practical optimum parameter selection, the following procedure can be proposed:

- the idealized physical weight equations are formulated to obtain a survey of the relevant weight coefficients
- variables with low variance over the data sample are replaced by typical constants
- after calculating the partial correlation coefficients for the individual variables, universal parameter combinations are created in such a manner that those variables are combined which
 - have a high correlation to the objective variable (weight)
 - and are mutually interdependent to a high degree
- Using the partial correlation coefficients for the new "overall parameters" it can be checked whether sufficient independency exists between these new variables (orthogonality test)
- individual variables with a strong correlation to the weight, for which the discrete numerical values cannot be provided very easily, will be replaced by other parameters which can be obtained without difficulty
- the calculated correlation coefficients have to be checked for significance, i.e. all values of r , smaller than a certain limit which depends on the magnitude of the sample, must be interpreted as a random deviation of $r = 0$
- in several cases it is not possible to assess simplified weight equations (e.g. for power systems, electronic equipment). For this, the correlation analysis offers new possibilities

A practical application of the procedure is presented in Fig. 6.

VFW -- PROJECT WEIGHT ANALYSIS FOR AIRCRAFT

PARTIAL CORRELATION ANALYSIS OF 8th DEGREE

WING WEIGHTS OF TRANSPORT AIRCRAFT

INFLUENCE PARAMETERS: GB, FTH, DIW, CF150, B, SKLFL, VD, KLAM, NBR

COMBINATION	ELIMINATED PARAMETERS	R-VALUE
GFL .GB	- ALL OTHER VARIABLES	0.5616
GFL .FTH	- ALL OTHER VARIABLES	0.4892
GFL .DIW	- ALL OTHER VARIABLES	- 0.3915
GFL .CF150	- ALL OTHER VARIABLES	- 0.4850
GFL .B	- ALL OTHER VARIABLES	0.2950
GFL .SKLFL	- ALL OTHER VARIABLES	0.1218
GFL .VD	- ALL OTHER VARIABLES	0.5138
GFL .KLAM	- ALL OTHER VARIABLES	0.1314
GFL .NBR	- ALL OTHER VARIABLES	0.1023
FTH .B	- ALL OTHER VARIABLES	0.4175
B .VD	- ALL OTHER VARIABLES	- 0.4499
GB .FTH	- ALL OTHER VARIABLES	0.1815

SIGNIFICANCE LIMIT: / 0.3853 /

Fig. 6 OUTPRINT OF A COMPUTER PROGRAM FOR CALCULATION OF PARTIAL CORRELATION COEFFICIENTS

4 DEVELOPMENT OF THE STATISTICAL ESTIMATING FUNCTIONS

The basis of the so-called regression analysis is the "principle of least deviation squares". The unknown variables of the approximation statement used are calculated from the condition equations for minimizing the deviation squares between observed and estimated weights.

4.1 The classical regression statement

First description of a classical regression formulation:

$$G(i) = C_0 + P(i,1)c(1) + P(i,2)c(2) + \dots + P(i,j)c(j) + \dots + P(i,n)c(n) \quad i = 1 \dots N, \quad j = 1 \dots n \quad (28)$$

To simplify the numerical calculation the basic statement is linearized by taking the logarithm. Index i running from 1 to N characterizes the individual observation (an existing aircraft). Therefore, Eq (29) represents a system of N equations:

$$\begin{aligned} \lg \hat{G}(i) &= \lg C_0 + c(1) \cdot \lg P(i,1) + \dots + c(j) \cdot \lg P(i,j) + \dots + c(n) \cdot \lg P(i,n) \\ \hat{g}(i) &= K_0 + c(1) \cdot p(i,1) + \dots + c(n) \cdot p(i,n) \end{aligned} \quad (29)$$

The next step is the elimination of the constant K_0 :

$$\hat{g}(i) \cdot \bar{g} = c(1) \cdot [p(i,1) \cdot \bar{p}(1)] + \dots + c(j) \cdot [p(i,j) \cdot \bar{p}(j)] + \dots + c(n) \cdot [p(i,n) \cdot \bar{p}(n)] \quad (30)$$

for which:

$$\bar{p}(j) = \frac{1}{N} \sum_i p(i,j) \quad \bar{g} = \frac{1}{N} \sum_i \hat{g}(i)$$

The unknown exponents can be computed from:

$$\frac{\partial \sum_i f(i)}{\partial C(k)} = 0 \quad k = 1 \dots n$$

$f(i)$ are the individual errors of the data sample. Index k runs over all unknown variables.

$$f(i) = [\hat{g}(i) \cdot \bar{g}] - \sum_j c(j) \cdot [p(i,j) \cdot \bar{p}(j)] \quad (32)$$

The practical calculation can be rationalized through introducing the matrix calculus:

$$A \cdot c = b \quad (33)$$

Eq (33) can be written in more detail:

$$A \cdot c = \begin{bmatrix} a_{11} & a_{12} & \dots & a_{1j} & \dots & a_{1n} \\ a_{21} & a_{22} & & & & \\ a_{j1} & & & & & \\ a_{n1} & & & & & a_{nn} \end{bmatrix} \cdot \begin{bmatrix} c(1) \\ c(2) \\ \vdots \\ c(j) \\ \vdots \\ c(n) \end{bmatrix} = \begin{bmatrix} b_1 \\ b_2 \\ \vdots \\ b_j \\ \vdots \\ b_n \end{bmatrix} = b \quad (33a)$$

The elements of the system matrix are derived from Eq (31):

$$\begin{aligned} a_{jk} &= a(j,k) = \sum_i p(i,j) \cdot p(i,k) \cdot \frac{\sum_i p(i,j) \cdot \sum_i p(i,k)}{N} \\ b_k &= b(k) = \sum_i \hat{g}(i) \cdot p(i,k) \cdot \frac{\sum_i \hat{g}(i) \cdot \sum_i p(i,k)}{N} \end{aligned} \quad (34)$$

Using one of the available computer routines for matrix inversion, the solution for C is:

$$c = A^{-1} \cdot b \quad (35)$$

The necessary condition for the existence of a solution is the non-singularity of A

$$\det A \neq 0$$

The unknown constant K_0 can then be predicted from:

$$K_0 = \bar{g} - \sum_j c(j) \cdot \frac{\sum_i p(i,j)}{N} \quad (36)$$

Retransformation:

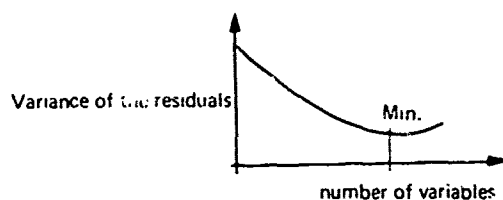
$$C_0 = e^{K_0}$$

The experience gained with this procedure can be summarized as follows:

- The number of weight influence parameters should be limited (max. 8) to avoid difficulties with the inversion of A (problem of quasi-singularity)
- The number of data points must be at least 3 times the number of weight parameters
- Only truly independent observations should be used
- Due to the remaining functional coupling between the variables, which cannot be eliminated completely by such simple statistical models, the developed estimating formulas sometimes show physically incorrect tendencies

4.2 Analysis of variances

Applying the described combination of physical parameter selection and correlation analysis a certain number of weight influence variables can be suggested, but it has still to be determined which combination of these parameters gives the optimal regression formulation. With the aid of a method for analyzing the variance about the regression function, it can be shown, that in many cases a smaller number of variables results in a smaller standard deviation of the estimating formula (see Fig. 11).



The basis for this additional selection of parameters is the modified correlation matrix \bar{R} which is reduced to the significant variables. STUDENT's t-test [3] is used here as significance test. The inverse matrix \bar{R}^{-1} is then stepwise further reduced by one variable respectively:

$$\hat{r}_{jk} = r_{jk} - r_{jq} \frac{r_{qk}}{r_{qq}} \quad (37)$$

The condition for eliminating a parameter is the minimum of the so-called "degree of partial determination" of this variable with regard to the objective variable

$$B_{yq \cdot 1,2, \dots, (q-1)(q+1) \dots n} = \frac{r_{qy}^2}{r_{qq} \cdot r_{yy}} \quad X_q \cong \text{Min} \{ B_{yq \cdot 1,2, \dots, (i-1)(i+1) \dots n} \} \quad (38)$$

This sequence of reduction is finished after p cycles, if the variance of the residuals for the estimating function with (n-p) variables is larger the first time than in the previous step:

$$\begin{array}{l} \text{After the reduction (step p).} \\ S_{Res}^2 = \frac{SAQ_{Res}^0}{|N-(n-p)-1|} \end{array} \geq \begin{array}{l} \text{before: (p-1)} \\ S_{Res}^2 = \frac{SAQ_{Res}}{|N-(n-p)-2|} \end{array} \quad SAQ_{Res} = \sum_{i=1}^N (y_i - \hat{y}_i)^2 \quad (39)$$

The $\hat{y}_i(p)$ ensue from a regression process

4.3 Constraint regression analysis

As the result of the relatively severe simplifications, a residual interdependency between the various weight parameters in the classical weight estimating formulation remains. This causes incorrect tendencies, if these formulas are used for partial weight trend investigations, a very important part of the project engineer's daily work. To prevent these effects, the unknown variables in the regression statement must be "pressed" in certain numerical ranges. For this purpose the method of constraint regression analysis is proposed. The minimization problem described by Eq. (31), supplemented by a set of restrictions, is the basis of this procedure. The constraints are

$$C_u(j) \leq C(j) \leq C_0(j) \quad (40)$$

Practical calculations have shown, that these very severe restrictions result very often in unacceptable error scatters. Therefore they will be simplified, whereas only the "correct" sign for the exponents is required.

$$c(j) \geq 0 \quad ; \quad \text{if } c(k) < 0 \rightarrow \bar{c}(k) = -c(k) \quad (40a)$$

The numerical process for solving this problem is rather complex. A non linear optimization procedure is proposed as a suitable method.

The basic formulation according to Eq. (29) or (30) expressed in matrix terms is

$$g = P \cdot \beta + \epsilon \quad (41)$$

g is the column vector of the observed sample weights, $(P \cdot \beta)$ represents the estimates and ϵ is the vector of the deviations. The sum to be minimized is:

$$\epsilon' \cdot \epsilon = (g - P \cdot \beta)' \cdot (g - P \cdot \beta) = \text{Min} \quad (42)$$

The additional linear constraints are

$$P \cdot \beta \geq g \quad \beta > 0 \quad (43)$$

The next step is to create a generalized LAGRANGE function $\Phi(\beta, u)$, which is the quadratic objective of this optimization problem. U is the column vector of the N LAGRANGE multipliers.

$$\Phi(\beta, u) = d' \cdot \beta + \beta' \cdot E \cdot \beta + u' \cdot (P \cdot \beta - g) \quad (44)$$

d and E are abbreviations

$$d' = -2g' \cdot P \quad E = P' \cdot P \quad (44a)$$

These variables are known.

The optimization conditions are given by the KUHN-TUCKER Theorem [4] and differ considerably from those of the linear optimization:

$$\frac{\partial \Phi}{\partial \beta} = v \leq 0 \quad \beta \geq 0 \quad \frac{\partial \Phi}{\partial u} = 0 \quad \beta' v \geq 0 \quad (45)$$

For the numerical solution, WOLFE's method [5] is used, which is a modification of the wellknown SIMPLEX-procedure for solving linear optimization problems. In this process the sum of the components of two additionally introduced variables which transform the inequalities into equations, are minimized to 0 in two steps. Thereby, the basic solutions are iteratively improved. The use of a high speed computer is necessary for this. The method is applied in all those cases, where the classical regression statement fails. Sometimes, the price for these advantages - reasonable weight trends - is a larger variance of the predicted estimating formula.

4.4 Non-linear regression

Investigating physical relationships between the weight and the other design parameters, it is obvious that a non-linear regression statement of additive connected exponential functions (see Eq. (21)) probably gives the best results.

This fact has been wellknown for a long time. The application of this formulation has been avoided, since a linearization creates certain difficulties. The new basic statement in matrix terms is:

$$g = f(P, \Theta) + \epsilon \quad (46)$$

The vector Θ is created by the unknown variables C_j and α_j . The essential condition for the prediction of the optimal regressors is once again the minimization of the sum of deviation squares (Eq. (42)).

The sum of the deviation squares (SAQ) in this particular case is

$$SAQ(\Theta) = \sum_{i=1}^N \epsilon_i^2 = \sum_i \{g_i - f(p_i, \Theta)\}^2 \quad (47)$$

In contrast to the formulations described so far, the following normal equations for this problem are non linear, which can easily be checked through differentiation of Eq. (21) with regard to one of the unknown variables.

$$\sum_{i=1}^N \{g_i - f(p_i, \hat{\Theta})\} \cdot \left[\frac{\partial f(p_i, \Theta)}{\partial \vartheta_i} \right]_{\vartheta_i = \hat{\vartheta}_i} = 0 \quad (n=1, 2, \dots, 2n) \quad (48)$$

For an iterative solution of this problem, a linearization can be attained by applying an extension into a TAYLOR series.

$$f(p_i, \Theta) = \underbrace{f(p_i, \Theta_0)}_{f_i^0} + \sum_{l=1}^{2n} \underbrace{\left[\frac{\partial f(p_i, \Theta)}{\partial \vartheta_l} \right]_{\vartheta_l = \vartheta_{l0}}}_{Z_{li}^0} \cdot \underbrace{(\vartheta_l - \vartheta_{l0})}_{\beta_l^0} \quad (49)$$

For this, it is necessary to estimate a solution in advance. In this particular case it is possible to determine that exponents $\alpha_j > 2,0$ are mostly unacceptable. Therefore:

$$\Theta_0 = \{\alpha_{10}, \alpha_{20}, \alpha_{n0}; c_{10}, c_{20}, \dots, c_{n0}\} \quad (50)$$

The other unknown variables (the coefficient C_j) result then from a linear regression analysis (as previously described)

The linearized system can again be expressed in matrix terms.

For every individual observation:

$$y_i = g_i - f_i^0 = \sum_{l=1}^{2n} \beta_l^0 \cdot Z_{li}^0 + \varepsilon_i \quad (51)$$

As a system of equations:

$$Y_0 = Z_0 \cdot h_0 + \varepsilon \quad (51a)$$

b_0 is an estimate of β_0 . It follows from Eq. (49):

$$b_0 \equiv \beta_0 = (\Theta - \Theta_0) \quad (52)$$

Using the general minimization once more:

$$\varepsilon' \cdot \varepsilon = \text{Min} \quad (42)$$

the following issues:

$$b_0 = (Z_0' \cdot Z_0)^{-1} \cdot Z_0' \cdot Y_0 \quad (53)$$

The iterative improvement of the solution vector results from:

$$\Theta_{m+1} = \Theta_m + b_m = \Theta_m + (Z_m' \cdot Z_m)^{-1} \cdot Z_m' \cdot Y_m \quad (54)$$

The iteration is completed, if Θ no longer differs from the former estimate and the corresponding least squares sum has reached its minimum.

$$\left. \begin{aligned} |\vartheta_{l(m+1)} - \vartheta_{lm}| < \delta \quad \delta = 10^{-5} \cdot \text{Min} \{ \vartheta_{lm} \} \\ \text{SAQ}(\Theta_{m,2}) > \text{SAQ}(\Theta_{m,1}) < \text{SAQ}(\Theta_m) \end{aligned} \right\} \quad (55)$$

As an alternative to the proposed linearization procedure a method developed by KRAWCZYK [6] can be mentioned. This procedure provides a direct solution for the non-linear system of the normal equations (see Eq. (49)) by means of a quadratic converging iteration process. The method of steepest descent, often used for similar problems, cannot be recommended as the required computer time is too long.

The experience gained from the first-mentioned linearization technique can be summarized as follows:

- In about 90 % of all applications convergence has been reached. However it is necessary that the first estimate for the solution does not differ too much from the final solution.
- The required computer time is larger than for the classical multiplicative regression formulation but of approximately the same magnitude as for the constraint regression analysis.
- The predicted estimating functions have very often a smaller error scatter than the known weight equations used at present.
- The magnitude of the data samples must be greater than that of the other statistical models.

5. THE OBJECTIVE VALUATION OF ALTERNATIVE ESTIMATING FUNCTIONS

For the majority of the known weight estimating formulas the possible error variation is given, but often this scatter is considerably exceeded in practical applications, e.g. checking the formulas by calculating the known weights of existing aircraft. One of the most important reasons for this is that the prerequisites for the specified error tolerances are not mentioned nor are the theoretical assumptions taken as the basis for the error distribution investigation fulfilled, e.g. a check should be made as to whether it can be assumed that the error of the discussed weight estimating procedures follows a normal distribution according to GAUSS. For the purpose of objective comparisons between various newly developed formulas and classical methods, several error probability investigations must be carried out.

5.1 Test on normal distribution

In this test the considered data sample (here: the totality of all individual differences between predicted and actual weights characterizing a particular estimating procedure) is checked to see whether it originates from a normally distributed parent population. Only if this test gives a positive result, can the formulas of the GAUSS-distribution be used to describe the error scatter of an investigated method. As a check the so called "X²-test" according to PEARSON [7] is applied:

For this purpose the error scatter of the sample is subdivided into several intervals. The corresponding probabilities - using the hypothetical normal distribution - are then compared with the relative class frequencies of the observed procedure errors. Only if the sum of the differences (the X²-value) is under a certain limit can the hypothesis that the sample originates from a normally distributed parent population, be accepted.

The error of an estimating formula for every individual observation (an existing aircraft) is:

$$f(i) = \frac{g'(i) - g(i)}{g(i)} \quad (56)$$

The error probability of the normal distribution according to GAUSS is given by:

$$F(f) = \frac{1}{\sigma\sqrt{2\pi}} \int_{-\infty}^f \frac{1}{2} \left(\frac{v-\mu}{\sigma} \right)^2 dv \quad (57)$$

For the normal distribution the error mean can be estimated from:

$$\mu \cong \bar{f} = \frac{1}{N} \cdot \sum_i f(i) \quad (58)$$

The standard deviation, the error interval about the mean including 68 (%) of all individual "events", follows from another estimate:

$$\sigma \cong s = \sqrt{\frac{\sum_i (f_i - \bar{f})^2}{N \cdot p - 1}} \quad (59)$$

5.2 The probable error distribution

Assuming the 'X²-test' has a positive result, one should check whether the investigated sample is representative of the parent distribution - the totality of all possible applications of the developed formula. In reality the value of the error mean will be in an interval about the sample mean - the so-called confidence interval. Assuming a high statistical confidence level ($\gamma = 95\%$) the confidence intervals for error mean and standard deviation can be calculated as follows:

mean.

$$\bar{f} - a \leq \mu \leq \bar{f} + a \quad (60)$$

a is predicted applying STUDENT's t-distribution [7]:

$$f(t) = \frac{1}{2} (1 + \gamma) \quad a = S \frac{\hat{t}}{\sqrt{N}} \quad (61)$$

Standard deviation:

$$a_1 \leq \sigma^2 \leq a_2 \quad (62)$$

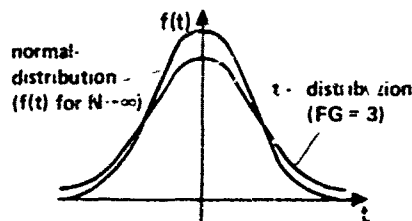
$a_{1,2}$ follows from:

$$f_{1,2}(c_{1,2}) = \frac{1}{2} (1 + \gamma) \quad a_{1,2} = \frac{(N-1) \cdot S^2}{c_{1,2}} \quad (63)$$

in which $f(c)$ is the χ^2 -distribution

The "t-distribution" is shown in the figure opposite. If N tends to be an indefinite large number, i.e. a very expanded sample, then the difference between normal and t-distribution disappears.

An extension from the $(1 - \sigma)$ -deviation (68 % of all observations) to 90 % seems to be reasonable.



$$K \cdot \sqrt{a_1} \leq k \cdot \sigma \leq K \cdot \sqrt{a_2} \quad (64)$$

In this case the constants k and K are functions of the confidence level γ and the measurement of the probability frequency integration range p .

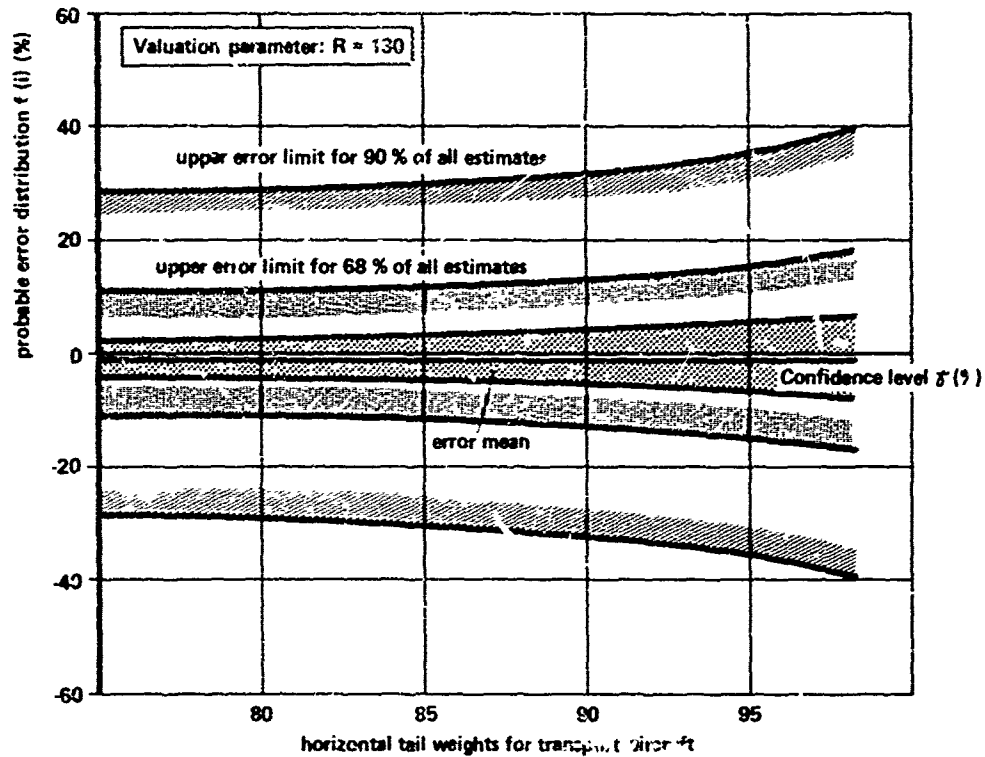
The total error scatter can be formulated as:

$$\bar{f} - K \cdot s \leq f(t) \leq \bar{f} + K \cdot s \quad (65)$$

Applying this information, the following valuation parameter for objective procedure comparisons is proposed:

$$R = ((\bar{f} + a) \cdot K \cdot s) \quad (66)$$

The dependency between the error tolerance parameters and the assumed confidence level is plotted in Fig. 7 and 8 for two different weight prediction formulas. Fig. 7 is valid for a formula developed with one of the described methods. Fig. 8 represents the accuracy which can be expected of a well-known classical formula.



$$G_{HLW} = 533,2 \cdot (n_{Br} \cdot G_B)^{0,52} \cdot (F_{STR})^{0,31} \cdot i_{HLW}^{0,12} \cdot r_{HLW}^{-0,44} \cdot b_{NP}^{0,4} \cdot \sigma_{H_w}^{-0,59} \cdot i_w^{0,24} \cdot (1 + \cos 250)^{-0,31}$$

Fig. 7 VALUATION DIAGRAM FOR A NEWLY DEVELOPED FORMULA

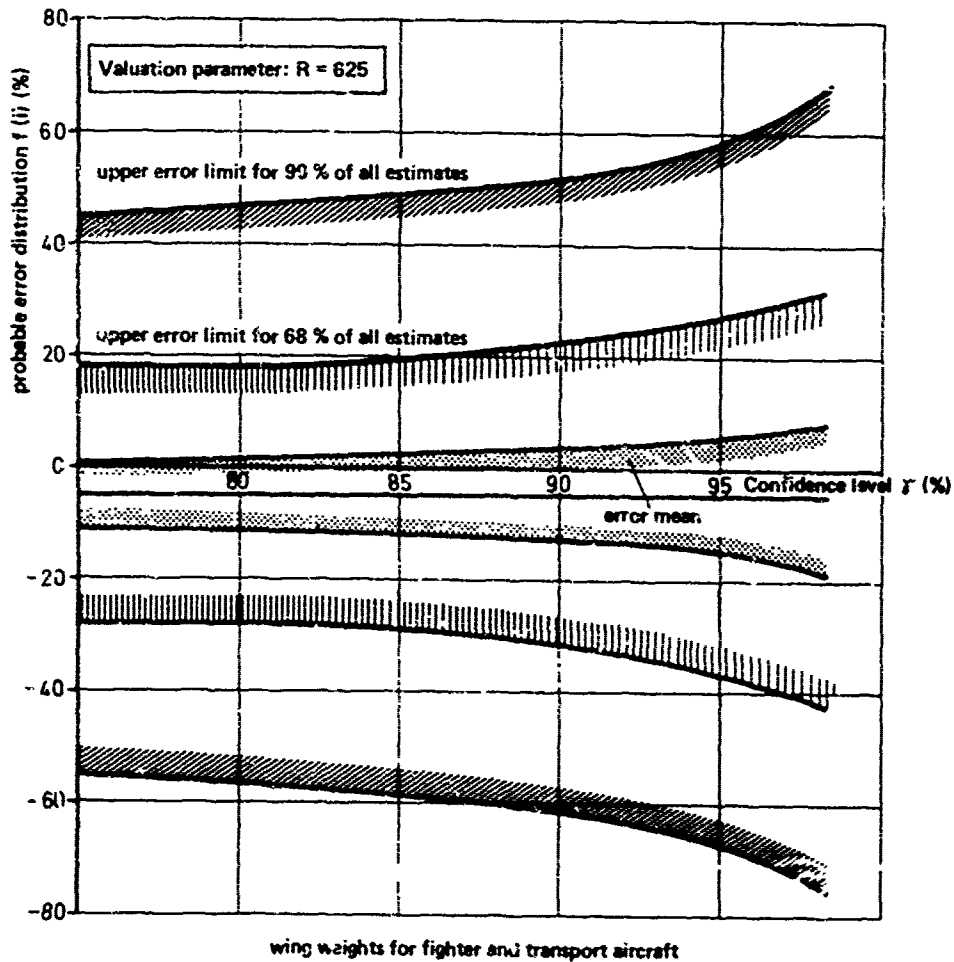


Fig. 8 VALUATION DIAGRAM FOR A CLASSICAL ESTIMATING FORMULA

6. POSSIBILITIES AND RESTRICTIONS FOR APPLYING STATISTICAL EVALUATION METHODS

Finally the presented methodology can be summarized as follows:

- The selection of semi-orthogonal weight influence parameters can be improved by combining correlation analysis and idealization of the fundamental physical relationships.
- Applying the multiplicative regression formulation and a reliable data collection weight estimating functions of first level accuracy can be developed.
- In cases, where this method results in incorrect weight trend tendencies, the application of the constraint regression analysis brings improvements.
- If larger data samples are available, non-linear regression procedures result in estimating formulas with smaller error tolerances.
- For the valuation of newly developed formulas the prerequisites of the error probability theory must be taken into consideration. Normal distribution cannot be assumed automatically.
- The main application of this methodology is the project weight analysis of first level accuracy. The investigated weight groups must represent the common technological standard.
- All described methods can be applied to problems in other fields too (e.g. evaluation of aerodynamical test data, empirical load assumptions etc.).
- For trend analysis the use of statistically derived equations should be limited to small intervals. Applying these to optimization procedures is very dangerous! The application of analytical techniques is preferred here.
- For numerical calculations a computer must be available.
- If all the individual methods are computerized it is possible to develop estimating formulas which are easy to handle. Changes to the formulated problems can be taken into consideration very quickly if an appropriate data sample is created. In this way the aircraft design process can be accelerated in an effective manner.

7. EXAMPLES OF APPLICATION

The results of two practical applications are shown in Fig. 9 and 10. These plots and the diagrams in Fig. 7 and 8 show that the application of sophisticated statistical models brings improvements but only to a certain extent.

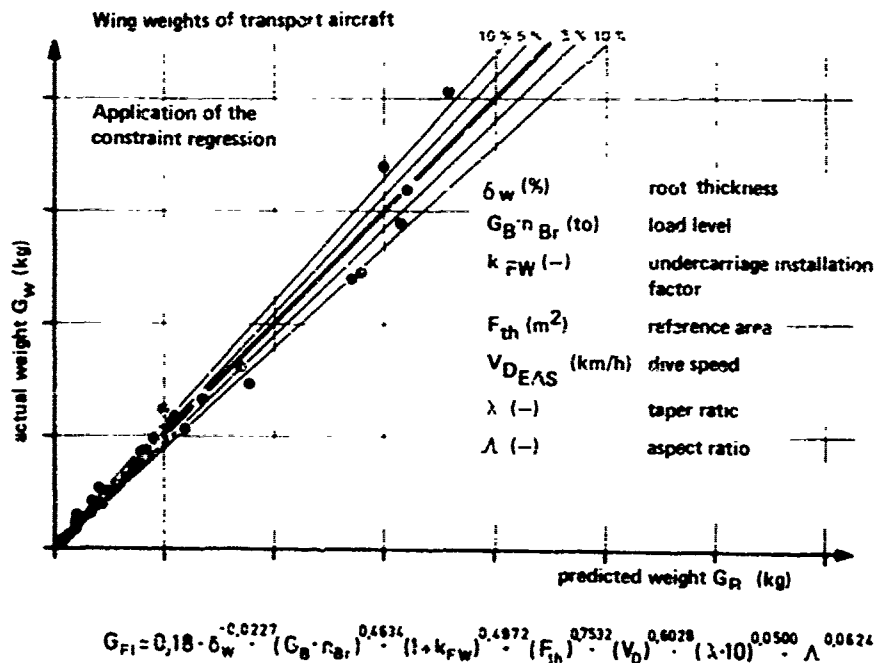
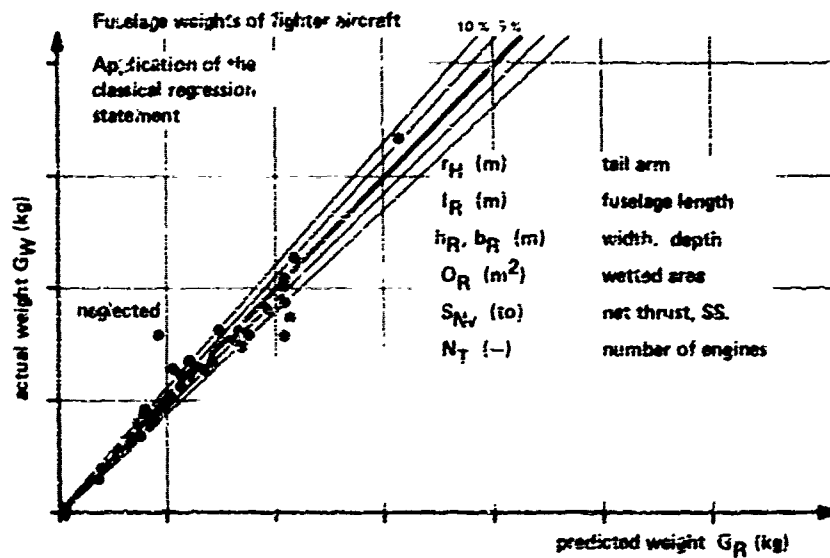


Fig. 9 PRACTICAL APPLICATION: WING WEIGHTS OF TRANSPORT AIRCRAFT



$$G_R = 99,323 \cdot (G_E \cdot n_E)^{0,2545} \cdot \left(\frac{10 \cdot l_H}{l_R}\right)^{0,58164} \cdot \left(\frac{V_D}{100}\right)^{0,26084} \cdot \left(\frac{l_R}{h_R \cdot b_R}\right)^{0,25514} \cdot \left(\frac{O_R}{\pi \cdot l_R}\right)^{0,92694} \cdot S_{NV}^{0,14035} \cdot N_T^{0,23418} \text{ (kg)}$$

Fig. 10 PRACTICAL APPLICATION: FUSELAGE WEIGHTS OF FIGHTER AIRCRAFT

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POTENTIAL PAYOFF OF NEW AERODYNAMIC PREDICTION METHODS

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SUMMARY

The development of a new aircraft requires an extensive series of trade studies and design compromises to optimize the system in order to achieve a well integrated configuration. The selected arrangement of wing, fuselage, tail surfaces and propulsion system may introduce subtle aerodynamic flow interactions which are not well understood or not recognized early in the design phase. The paper reviews several examples involving major aircraft programs where this has happened.

New aerodynamic analysis techniques now in development may hold the key to the development of better, more efficient aircraft if they are used throughout the development to guide the evolving design. The development of new design methods for analyzing the spin susceptibility and flight characteristics near the stall departure boundaries is also reviewed in the paper. Results of studies conducted to determine criteria for predicting departure from controlled flight at near-stall angles of attack, based on lateral-directional static stability are summarized. The use of the directional divergence, $C_{n\beta, \text{dynamic}}$, and a lateral control divergence parameter, LCDP, as criteria to design new high performance aircraft with inherent spin resistant qualities shows promise for avoiding the serious loss of control problem which has caused high losses on some of our older fighter aircraft.

SYMBOLS

Γ = Vortex Strength
 a = Distance Between Vortex Centers
 C_n = Yawing Moment Coefficient
 C_l = Rolling Moment Coefficient
 β = Angle of Sideslip (Degrees)
 α = Angle of Attack (Degrees)
 δ_a = Aileron Deflection (Degrees)
 LCDP = Lateral Control Divergence Parameter, $C_{n\beta} - C_{l\beta}(C_{n\beta}/C_{l\beta})$
 $C_{n\beta, \text{dynamic}}$ = Directional Divergence Parameter, $C_{n\beta} \cos \alpha - (l_z/l_x)C_{l\beta} \sin \alpha$

INTRODUCTION

The development of a new aircraft or missile is usually preceded by an extensive series of trade-off studies where many configuration parameters are varied over a broad range of numerical values. At this stage, aerodynamic prediction methods must consider the total configuration and many details such as flow patterns and local interference effects are not evaluated. The cost in manpower and computation time to study these effects at this phase of the program would be prohibitive and wasteful since only a few of the large family of configurations will be given serious consideration. When the candidates are narrowed to two or three configurations, it becomes important to evaluate potential problem areas and to investigate specific flow interference effects due to the arrangement of the wing, fuselage, tail surfaces and propulsion system. Unfortunately, there is a strong tendency to overlook potential problem areas at this stage for a number of reasons.

1. Time is not available to make an in-depth study.
2. Insufficient manpower on the program.
3. Wind tunnel data is lacking.
4. Aerodynamic analysis methods are not accurate enough.
5. Inexperienced staff.
6. Strong competition for the program.

There are undoubtedly many other reasons. One fact remains, the selected configuration may, and often does, have serious aerodynamic problems which will have to be solved before the development is completed.

Although the design will not be frozen for 12 to 18 months after the development has been started, the general arrangement of the major components is usually firmed up quite early. The aerodynamic interactions between the wing, fuselage, tail surfaces and propulsion unit are not well known at this time; however, it is assumed that these details will be worked out during the wind tunnel program.

There is really nothing wrong with this process if the different design groups are well integrated and controlled. The problems usually show up when the idealized configuration is closely examined and certain deficiencies show up. These may be caused by one or several restraints such as weight, structural design, internal component layout, balance, avionics requirements, crew station design and many others. The aerodynamics design group may be placed under considerable pressure to relax their requirements to help solve the deficiency. There is inexorable pressure to move ahead and not delay the program for

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further study. The aerodynamic prediction methods may not be adequate to completely analyze the problem and wind tunnel data to back up the prediction may not be obtainable in a short period of time. The decision must be based on the best available information and the design groups must move ahead or the schedule will slip. If the right decision can be made, the payoff can be very large over the lifetime of the system in terms of future operating costs, performance and mission effectiveness.

Military systems have become so complex and expensive to build and operate that life cycle costs may soon become the most important parameter to be considered before a final configuration is selected for development and even throughout its development. Performance may have been the most important factor in past developments, but, it is not necessarily so any longer.

There is a very significant payoff for good integration of the complete configuration at an early stage in the development. Refined aerodynamic prediction methods, backed up by high quality wind tunnel data are needed in order to make the best design decisions. These can have a far reaching effect on life cycle costs as well as performance.

THE DESIGN PROCESS

Although most aerodynamic design problems are discovered and corrected early in the development phase, it is clearly evident that many of our major aircraft programs have experienced airframe and propulsion problems early in the flight test program. The result has often been a costly redesign or delay in the production which can have a most serious impact on the program. Table I shows only a partial list of the problems that were found after flight testing began on a number of USAF aircraft. In some cases, such as the B-47 aileron reversal, structural flexibility was the primary reason. However, in many of these cases, the problem did show up in the wind tunnel tests, but, was not recognized as a serious problem or was blamed on the low Reynolds Number of the test. Of course, stall and spin investigations require special tests such as free flight or spin model tests and this was not always accomplished in our older programs. Most of our current aircraft programs, even large aircraft such as the B-1, now include tests at high angle of attack well beyond the stall. Spin model tests in the wind tunnel and radio controlled free flight models are used in these investigations. The three items marked with a star on this Table are typical problems which often do not receive sufficient attention before flight tests are started. They will be reviewed in further detail in this paper.

B-47 Aileron Reversal & Pitch-up	TF-102 Canopy Separation
B-58 Engine Out Control	F-4 Loss of Control at Stall/Spin
B-66 Tip Tank Buffet	*RF-4C Nose Separation
C-123 Rudder Lock	F-111 Secondary Air System/Inlet
C-130 Low Long. Stability-Landing	AGM-69 Yaw-Roll Coupling
C-133 Poor Stall Characteristics	*AWACS Radome Pitching Moment
*C-141 Fuselage Drag & "Shibbole"	C-141 Wing Shock Location

TABLE I - PROBLEMS FOUND DURING FLIGHT TEST

Studies have indicated that while the lessons learned in solving the many problems encountered can and should be passed along to benefit the next generation of aircraft, each new aircraft invariably brings with it a set of problems (1). They may be similar to old problems, but, new and unique solutions may be required because the new aircraft may have to fly over a much larger envelope of altitudes and airspeeds.

The process normally pursued in the development of aircraft systems is by nature an empirical, iterative process. Development and operational test data are recycled back into the design to complete the design and development of the system. If this can be accomplished rapidly without major design changes, the program progresses smoothly, schedules are met and production is uninterrupted. This ideal cannot be attained unless particular attention is given to the many interface areas in a complex system and the design is very carefully integrated between the airframe and propulsion systems. There is no doubt that many aerodynamic design problems can be traced to subtle flow interactions which may not be apparent in the many series of wind tunnel tests performed on a wide variety of model configurations which are used to develop some part of the total configuration. Some models represent only a small portion of the total configuration and important flow interactions may not be present on these models, particularly at transonic speeds.

Wind tunnel test techniques have not been standardized among different facilities and possibly they should not be. However, if the results from several facilities are inconsistent or show differences which are larger than the expected data accuracy, serious aerodynamic problems may be covered up or at least not thoroughly investigated. In the last 5 years there has been considerable improvement in wind tunnel test procedures as a result of special correlation tests conducted on a model of the C-5A in three major transonic wind tunnels in the United States (2). Hopefully this progress will continue and the uncertainty of test data accuracy will not be a problem to the aerodynamicist in the future.

There is still the problem of extrapolating model scale results to the full scale aircraft. This has worried wind tunnel and aerodynamics engineers for years. A large number of theoretical and experimental research investigations have been accomplished to determine the effects of Reynolds Number on small scale model data. In fact, there is considerable controversy now going on whether or not Reynolds Number effects persist up to values of 30 to 50 million or more. Apparently enough test data has been found that supports both conclusions to keep this argument going for some time. There is no doubt that flow separation regions, shock wave-boundary layer interactions and wing shock wave locations may be strongly affected by Reynolds Number within the normal test range of transonic wind tunnel facilities.

Recent studies of drag prediction techniques for subsonic and transonic aircraft have shown that good correlation between low scale tunnel results and full scale flight data can be obtained if very careful test procedures are followed as outlined by Paterson, MacWilkinson and Blackerby (3). The use of standard scaling methods is adequate in most cases. The C-5A cruise drag at a Reynolds Number of 55 million (MAC)

was predicted to within 1 to 3 percent using wind tunnel data at approximately 4 million Reynolds Number. This favorable correlation was obtained only by a very extensive study of factors such as flexibility effects, roughness drag and in-flight thrust measurements which must be accurately known. The use of potential flow methods and boundary layer theories also proved to be a valuable tool in predicting airfoil profile drag and interpreting low Reynolds Number test data.

If the tunnel results are not carefully evaluated, problem areas may be hidden or dismissed as not too important. There have been numerous cases where a stability or drag problem discovered in the wind tunnel program was not corrected because it was judged to be an anomaly due to the low Reynolds Number of the test. Unfortunately, in many cases the problem was still present when the flight tests started. By that time it is difficult to justify a major design change because of the serious impact on production schedules and cost. In recent years, new procurement concepts such as "fly-before-buy", "milestone reporting" and "prototype programs have been initiated to avoid the cost of re-design during the production phase. It is quite evident that improved analytical prediction techniques to supplement standard tools such as the wind tunnel and computer flight simulation should provide a large payoff through better aerodynamic design early in the development.

WING-FUSELAGE DRAG

Large cargo aircraft are usually designed for maximum utility of the cargo compartment and ease of loading and unloading. To obtain an uninterrupted floor area, a high wing location is often used with an upswept fuselage shape to accommodate the large cargo door and rear loading provisions. The C-130, C-141 and C-5 aircraft are typical of this design trend. The aerodynamic design and integration of the wing, fuselage and landing gear pods to achieve low cruise drag and good handling characteristics is a large challenge to the designer because of the complex flow patterns on these configurations. Figure 1 shows the type of flow pattern present on the C-141 fuselage. Strong vortices are present adjacent to the fuselage afterbody which start in the wing-fuselage juncture and around the landing gear pod. During early flight testing of the C-141, Lockheed flight crews became aware of an oscillatory motion which was named "shibble", a combination of the two words, 'shake' and 'nibble'. It was characterized by random directional oscillations of the aircraft in the frequency range of four to seven cycles per second. The amplitude of this oscillation at the cockpit varied from negligible to as much as 0.07 of an inch at the highest air speeds. It was not always apparent to the pilot, but it did create considerable interest and concern.

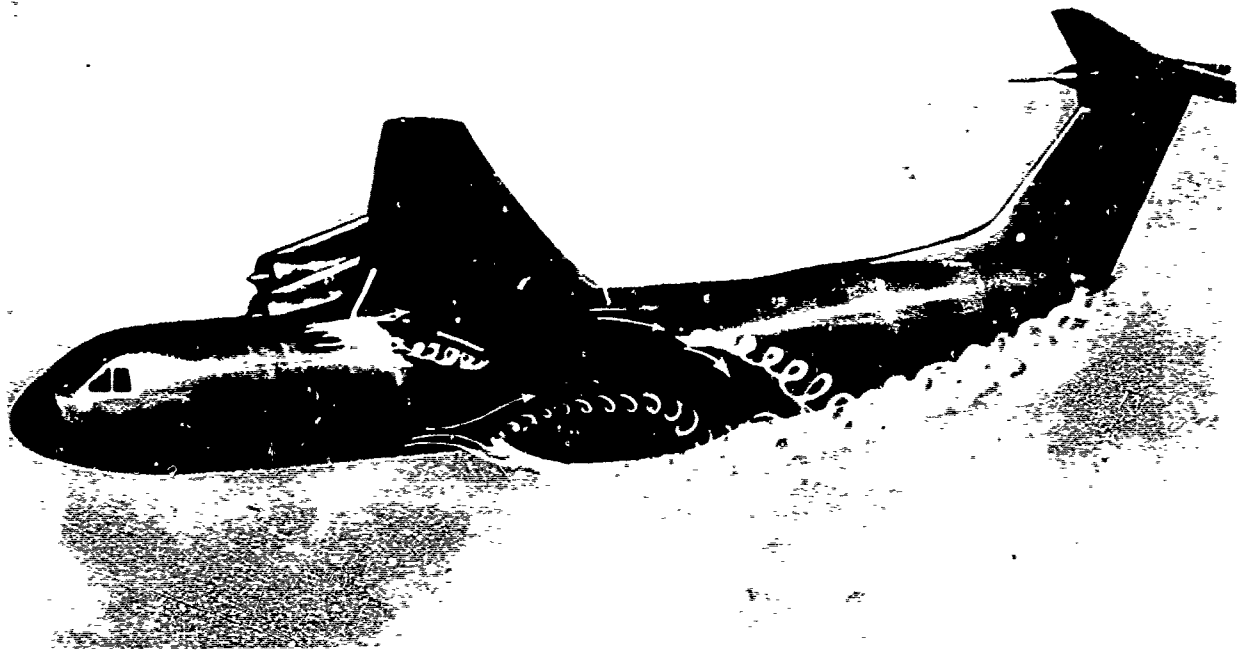


FIGURE 1 - C-141 VORTEX FLOW PATTERN

Lockheed initiated a series of oil flow visualization tests in the Georgia Institute of Technology Wind Tunnel to study the flow patterns around the C-141 to identify the cause of the problem and find a solution. It was thought that intermittent shifting of the vortices could be creating the aerodynamic driving force for the shibble oscillations. It was found that the strongest vortex emanates from the region of the wing fuselage juncture. The two vortices on each side of the fuselage at the wing root form a vortex pair of equal strength, but opposite rotation. The mutual interference between these two vortices induces a downward velocity behind the wing root and down over the fuselage. The downward velocity at the mid-point between the two centers is

$$w = \frac{2\Gamma}{a}$$

Γ = the vortex strength

a = distance between centers.

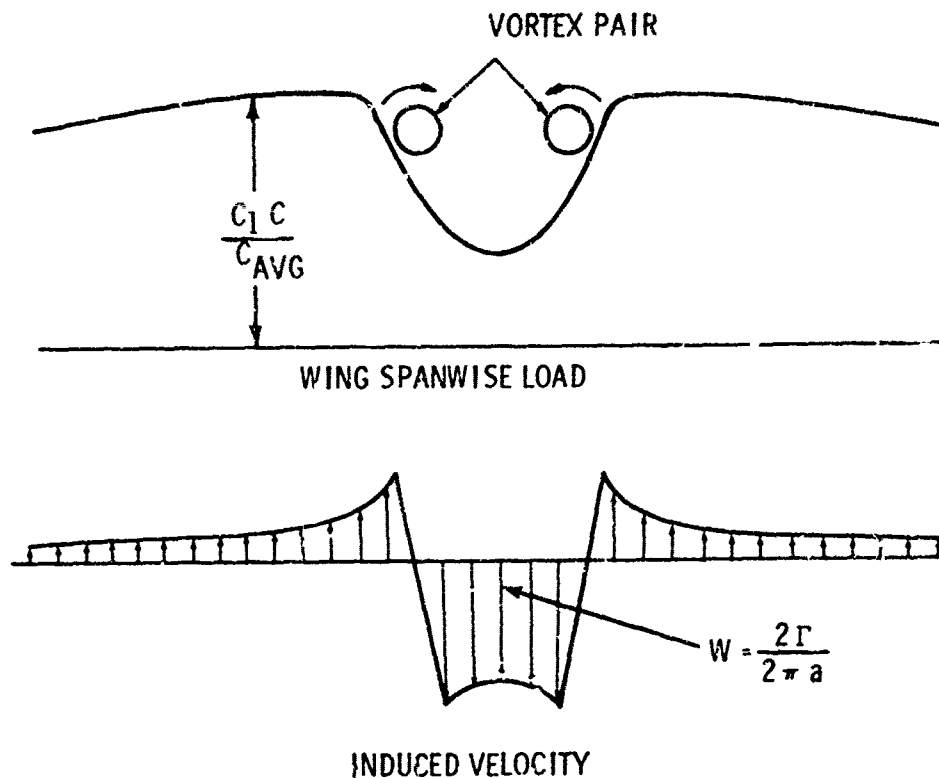
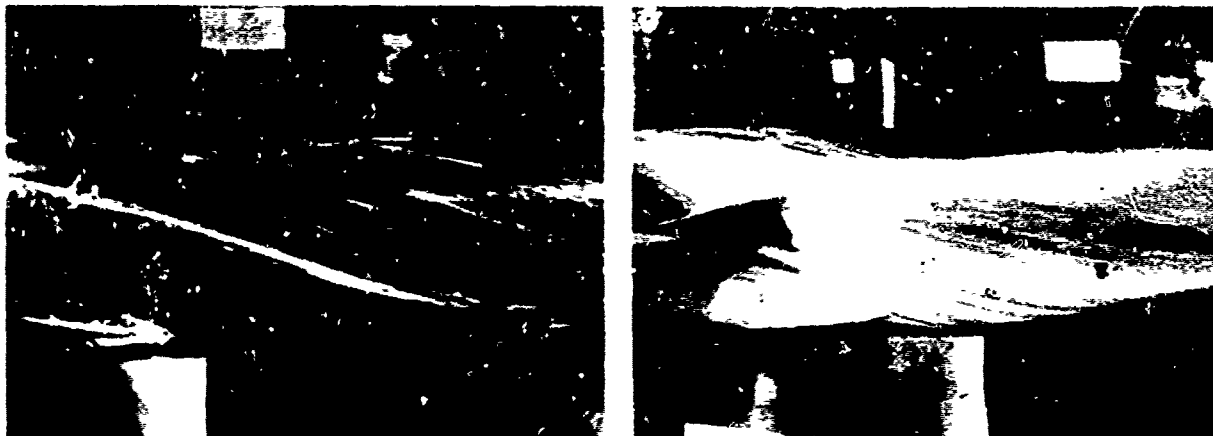


FIGURE 2 - EFFECT OF WING-FUSELAGE VORTEX

Figure 2 shows the velocity induced by the vortices and the effect on the wing lift distribution. The hole in the distribution causes a significant increase in the wing induced drag in the cruise region. The flow studies indicated that the vortices impinged on the rear fuselage which was the most likely cause of shibble. The C-141 flight crews also reported that the noise level appeared higher just aft of the wing trailing edge. A larger wing fillet was developed in the Georgia Tech. tunnel which substantially improved the flow around the wing-fuselage juncture and reduced the flow angularity and the strength of the vortex pair.



Standard Wing Fillet

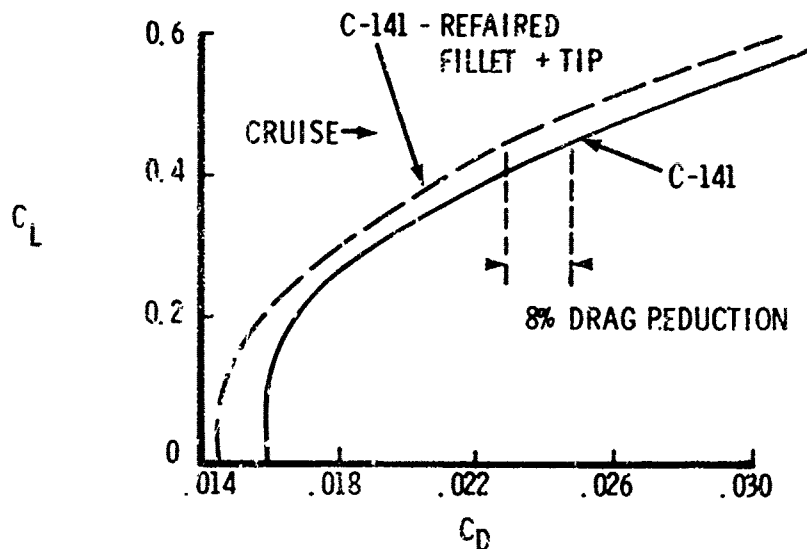
Improved Wing Fillet

FIGURE 3 - C-141 OIL FLOW PATTERNS

The oil flow photographs on Figure 3 show the improved flow patterns and the significant change in the flow angularity behind the wing. Wind tunnel force tests indicated a drag reduction of 5 percent in the cruise region was obtained with the improved fillet. There was also an improvement in fuselage lift carry over which moved the wing center of pressure inboard. It was found that a six-foot wing tip extension could be added by taking advantage of the reduction in wing bending moments obtained with the improved fillet. This produced an additional reduction in cruise drag of 3 percent.

It was unfortunate that this improved wing fillet was not developed earlier in the C-141 program. It was decided not to incorporate this modification since production was well along and the range performance was adequate to accomplish the airlift missions.

Figure 4 shows that the 8 percent cruise drag reduction can be translated into an 8 percent reduction in cruise fuel for each aircraft in the C-141 fleet. Based on current and projected future operation, a total saving of 82 million gallons per year appears possible. The modification cost for 277 aircraft has been estimated to cost approximately 11 million dollars, roughly equal to the cost of one year's fuel saving. This is a significant potential saving since the C-141 fleet will be flying for many years. It should be noted that the fillet and wing tip modification would not have increased the wing stress levels because of the favorable effect of the fillet on span load distribution.



FUEL SAVING FOR C-141 FLEET -

82,000,000 GALLONS PER YEAR

ESTIMATED COST OF MODIFICATION -

277 A/C \$11,000,000

FIGURE 4 - C-141 DRAG REDUCTION

Early wind tunnel tests of the C-5A showed a drag increase of 33 drag counts over the predicted cruise drag value of 250 counts. Again the problem was found to be associated with the wing-fuselage fillet. The fuselage drag was also found to be adversely affected by the fairings around the nose gear and main gear pods. Lockheed engineers took advantage of the knowledge gained on the C-141 to redesign the critical areas on the C-5A and were able to completely eliminate the large drag increment. This happened early enough in the C-5A development to redesign the configuration before the schedule was impacted. The lessons learned on the C-141 were very effectively applied in the design of the C-5A. The result is an aircraft with a high degree of aerodynamic cleanliness which retains the advantages of the upswept fuselage design for cargo handling. This is a truly remarkable achievement.

RF-4C RECONNAISSANCE AIRCRAFT

The RF-4C is an F-4C aircraft modified to carry a variety of cameras and other reconnaissance equipment in the fuselage. A camera bay housing was added to the lower surface which has an abrupt ramp with a 33-degree nose angle. After the aircraft was placed in service it became evident that something was happening to the photographic resolution at high subsonic Mach numbers. A short series of flight tests by McDonnell Douglas of a 23-degree ramp indicated that the primary cause of image degradation was probably due to either an aerodynamic disturbance or flow field phenomena outside of the camera window, but, it was also noted that an additional loss in resolution was being produced by some unknown mechanical vibration within the camera. There was considerable variation in resolution between photographs which added to the mystery.

In 1969, the Aeronautical Systems Division started a series of photographic resolution tests in two transonic wind tunnels and laboratory vibration and acoustic tests to find the cause of the problem and hopefully a practical solution which could be applied to the RF-4C fleet. Full scale three dimensional models of the lower forward section of the RF-4C aircraft extending from the forward end of the radome to just aft of the second camera station were fabricated for use in the wind tunnel. Fabrication was simplified by utilizing a fiberglass mold from an actual RF-4C aircraft. Other model sections were constructed to provide a configuration in which the camera bay ramp was fully faired into the radome and also a configuration incorporating a camera bay ramp of 23 degrees which was similar to one flight tested by McDonnell. An 18-degree ramp was also tested at AEDC. The first series of tests were performed at the Cornell Aeronautical Laboratory 8-foot Transonic Wind Tunnel by mounting the large model on the floor of the tunnel. A collimator containing a light source and a tri-bar resolution target was mounted through the ceiling of the wind tunnel and aligned with a camera in the model. Both the camera and collimator were rigidly mounted to minimize relative motion due to tunnel vibration. In addition to the optical data, 23 static pressure measurements were made along the centerline of the model. Tunnel blockage proved to be a severe obstacle at Mach numbers above 0.80 in the Cornell Tunnel and it was decided to conduct additional tests on the same model at the Arnold Engineering Development Center (AEDC) 16-foot Transonic Wind Tunnel. Since the two series of tests were in good agreement only the AEDC results are presented. Figure 5 shows the results of the photographic resolution tests on the original 33-degree bay ramp and the modified 23-degree bay ramp. These results confirmed the large degradation in resolution above a Mach number of 0.70 with the 33-degree ramp configuration and show the large improvement obtained with the 23-degree ramp. Also shown for comparison are the results of flight tests which were flown shortly after these wind tunnel tests were completed. Even though the wind tunnel tests were performed with a smaller camera the

agreement was good, indicating that valid simulation of flight conditions was achieved in the optical wind tunnel tests conducted at Cornell and AEDC. This type of testing should be seriously considered in the development of future reconnaissance aircraft.

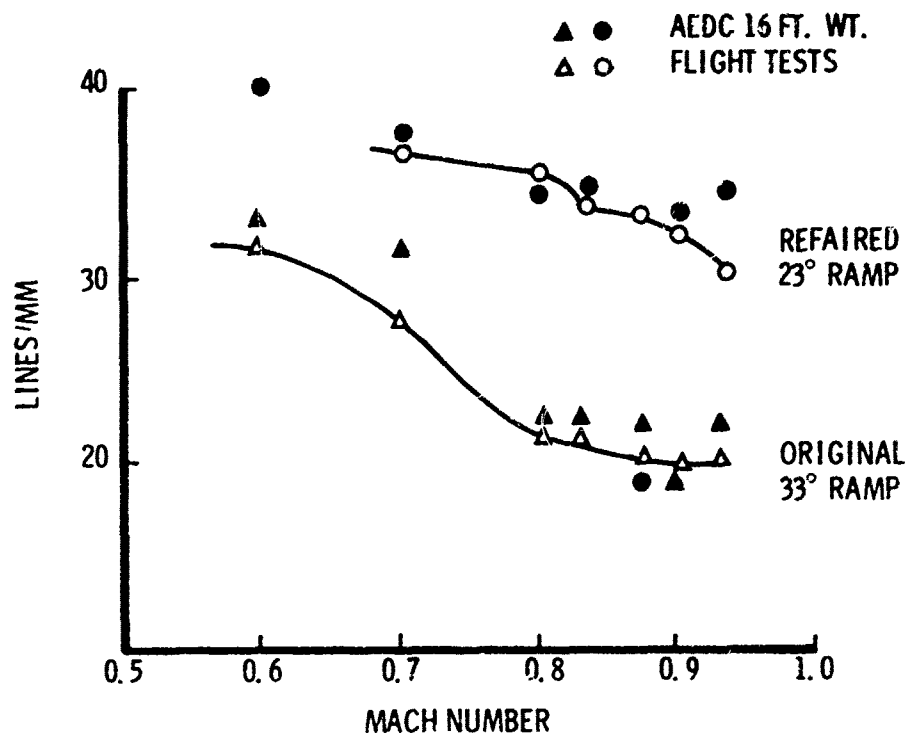


FIGURE 5 - RF-4C CAMERA RESOLUTION

The cause of the severe loss in resolution is shown in Figure 6. A strong normal shock is positioned directly over the center of the camera window at high speeds. At a Mach number of 0.87 the shock is positioned along the optic axis of the camera. The normal shock resulting from the large increase in local velocities over the 33-degree ramp is so strong that the flow is completely separated over the camera window. The small sketch on this figure was produced from the Schlieren photographs taken in the AEDC wind tunnel (4). The local pressure distribution confirms the extensive local flow acceleration, shock location and flow separation. Dynamic pressure sensors were also installed at eight locations. The upper part of the figure shows the root-mean-square values of the fluctuating pressures divided by the free stream dynamic pressure ($C_{p,rms}$). Maximum values of the fluctuating pressures were of the order of 12 percent of free stream dynamic pressure. Figure 7 indicates that the normal shock is still present over the 23-degree ramp, but, flow separation is no longer evident in the Schlieren photographs. The pressure distribution also shows the improved pressure recovery over the camera window. Also, the peak fluctuating pressures were reduced to about four percent of the free stream pressure. At a Mach number of 0.87, the normal shock is still positioned along the optic axis of the camera as in the case of the 33-degree ramp. This shock causes a distortion of the optic wavefront entering the camera aperture due to the large difference in refractive index which exists on either side of the shock wave. This explains the large loss in resolution at $M = 0.87$ with the 23-degree ramp shown on Figure 5. This was evident in flight tests as a narrow blurred band running across the photograph, but, was not objectionable since only a small area of the photograph was affected. Since no vibration of the optic system occurred during the wind tunnel tests, the source of the degradation could be established as the index of refraction variations in the separated air flow. This was confirmed by a detailed edge gradient analysis of the flight test and wind tunnel photographs by Mazurowski (5).

As a result of these investigations, a study was conducted to determine the costs of modifying the RF-4C fleet with a 23-degree ramp camera bay. Nine new aircraft were purchased in fiscal year 1970 with the modified camera bay for an additional cost of \$540,000. It was estimated that modification kits for 386 aircraft would cost 29,000 dollars each for a total of 11,194,000 dollars. The decision was made not to retrofit the fleet because of the cost. This is another case where improved aerodynamic analysis techniques combined with a rather low cost wind tunnel program could have resulted in much better mission performance, if the results had been available at an early stage before production was started.

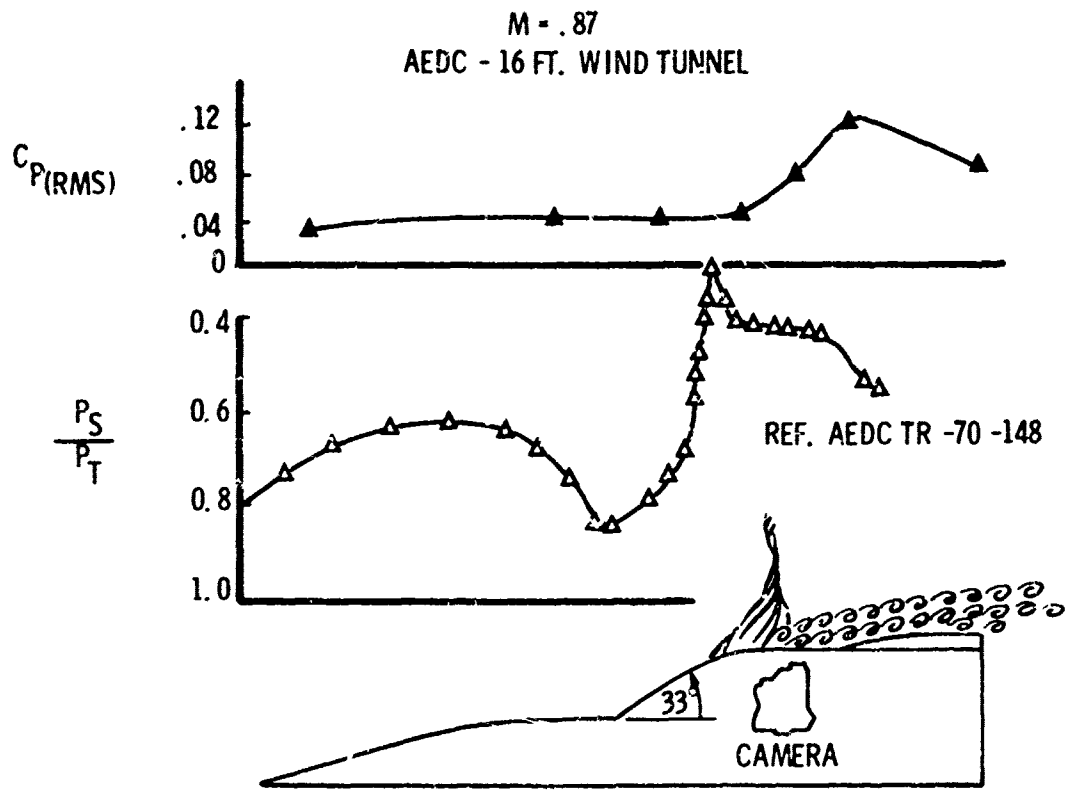


FIGURE 6 - RF-4C STANDARD NOSE

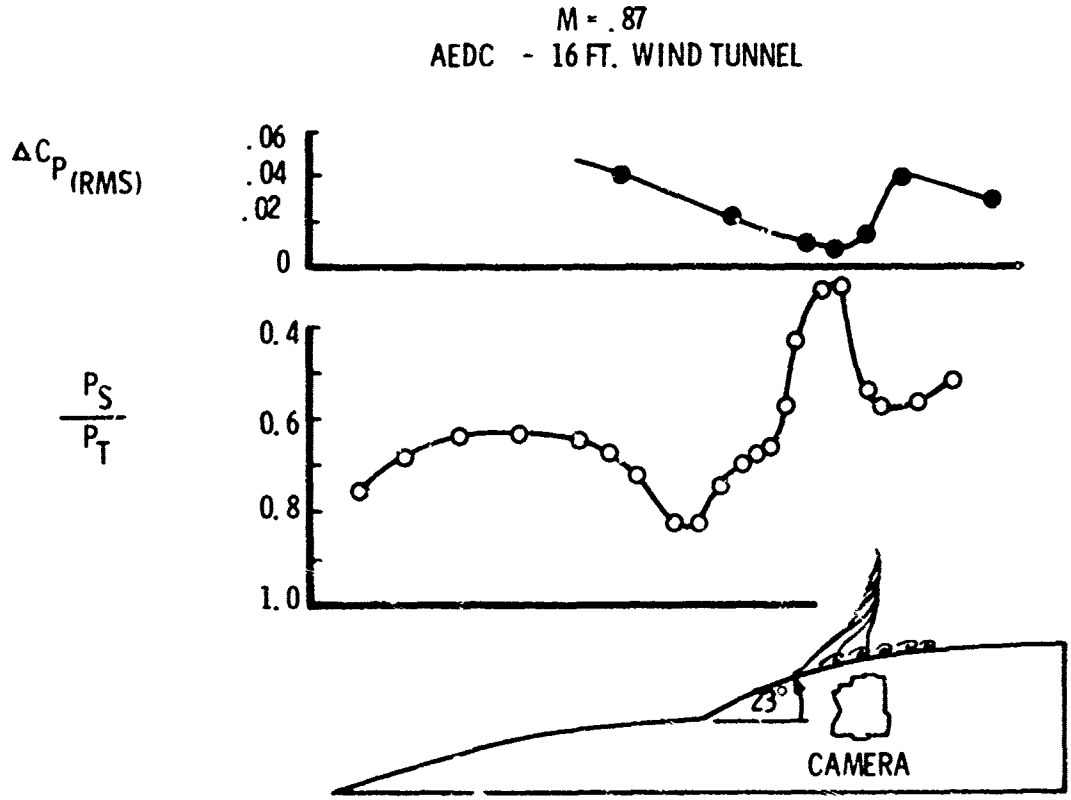


FIGURE 7 - RF-4C REPAIRED NOSE

AWACS RADOME PITCHING MOMENTS

Flight tests on the prototype aircraft for the Airborne Warning and Control System (AWACS) disclosed that air load distribution on the large 30-foot diameter radome was considerably different from that measured in the Boeing High-speed Wind Tunnel.

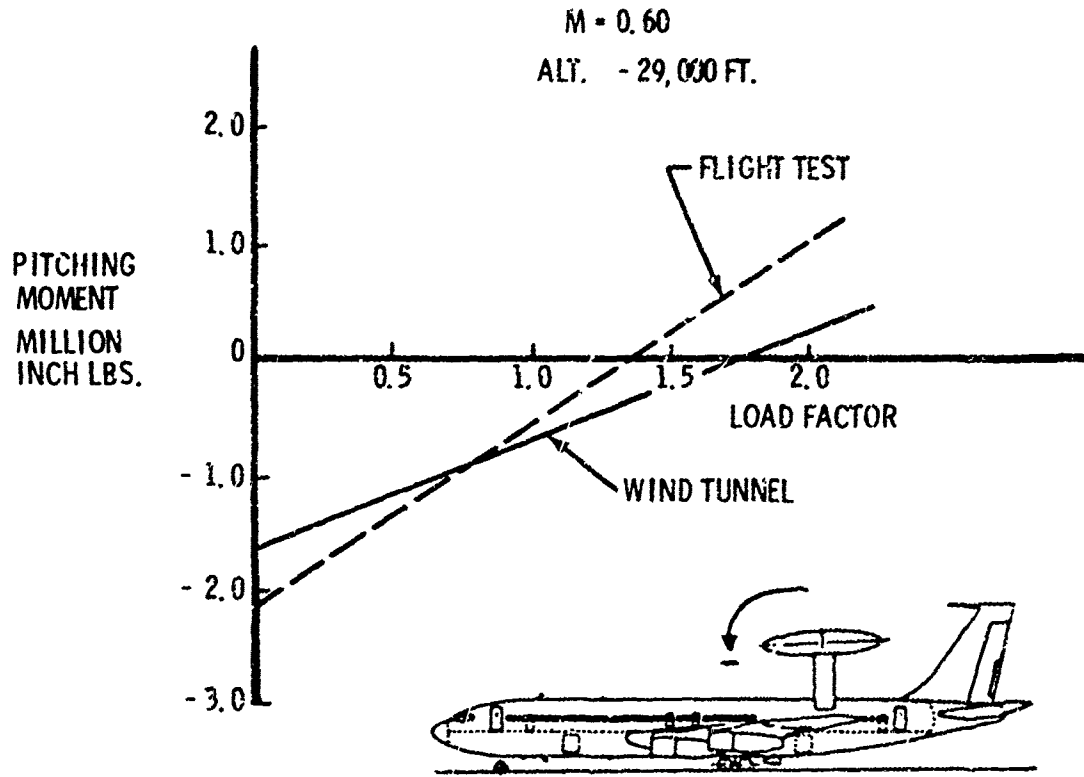


FIGURE 8 - RADOME PITCHING MOMENT

Figure 8 shows that the slope of the radome pitching moment curve versus aircraft load factor was considerably greater in flight than had been measured in the wind tunnel. This difference persisted at all Mach numbers and would have presented a serious structural load problem at the highest airspeed conditions for the AWACS airplane. The radome was instrumented with a single pressure belt along the center line and a complete survey of the pressure field was made by rotating the radome through a complete revolution for each flight condition.

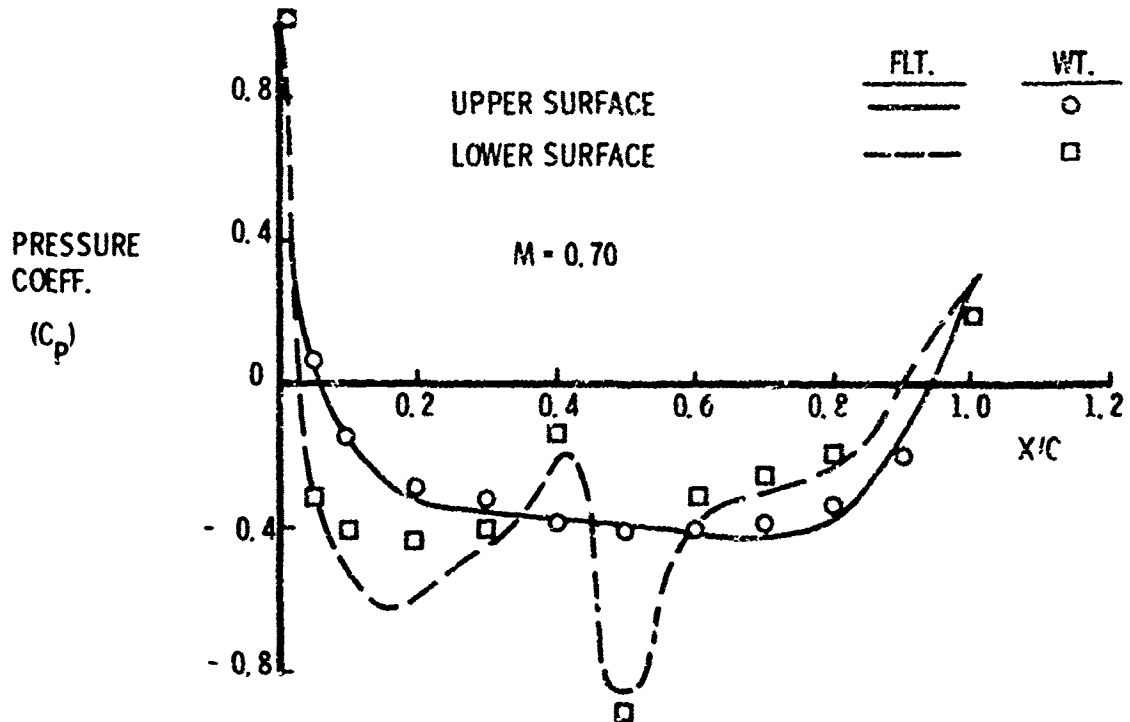


FIGURE 9 - RADOME PRESSURE DISTRIBUTION

Figure 9 shows the results of the flight tests compared to the wind tunnel results for both the upper and lower surface. Reynolds Number based on radome diameter was approximately 60 million in the flight test and 3.5 million in the Boeing wind tunnel test. The results indicated that the pressures are more negative on the lower surface near the leading edge in flight than measured in the wind tunnel. Additional wind tunnel tests were made with and without boundary layer transition strips, but, these were unsuccessful in duplicating the full scale radome loads. Pressures measured in flight over the upper and lower surface behind the leading edge region were in good agreement with the wind tunnel data.

Fortunately, the problem was easily solved by changing the radome incidence 1.5 degrees in a nose-up direction when it was found that the antenna performance was not seriously degraded in the new position. It does indicate that either wind tunnel test techniques or model scale effects should be studied in greater depth to improve the wind tunnel simulation on certain types of configurations. It also indicates that aerodynamic analysis techniques which can accurately predict the local pressure distribution over complex configurations should be developed to supplement the wind tunnel. We cannot always count on solving air load problems by simple modifications as in the case of the AWACS aircraft.

PROMISE OF NEW PREDICTION TECHNIQUES

Many design problems are not discovered early in the development because the wind tunnel models may not closely simulate the final configuration in all respects. It is often assumed that corrections can be made to account for local changes in fuselage shape, tail surface intersections and filleting around the wing and tail surfaces. Also, details of the propulsion system such as secondary air flow passages and bleed outlet may not be simulated because of the scale problem. These areas are examined and analytical corrections applied to account for drag and stability increments. If a serious aerodynamic problem is suspected, additional models may be constructed to research the problem, but, the pressures of program schedules and budget limitations restrict this type of research to a bare minimum. The analysis methods used are largely empirical and are usually based on experimental data from other programs. If the configurations are quite similar, as was the case for the C-5A and C-141, very good results may be obtained. Problem areas will be apparent and solutions can be rapidly investigated in the next series of tunnel tests. Unfortunately the trend toward fewer new systems is working against the designer. His background data on similar configurations will be less as time goes on since fewer new aircraft are being developed.

The development of improved aerodynamic analysis techniques which could be used for the detailed design of new configurations is needed to supplement wind tunnel testing. Many of the current computerized analysis techniques were developed to rapidly examine a variety of aerodynamic configurations to determine the optimum arrangement of the wing/fuselage and control surfaces. Most of these methods are satisfactory for preliminary design purposes and are quite useful in optimizing a new concept before detailed design is initiated. At this stage, various trade-offs are being studied and these methods provide a rapid means of studying the effect of many configuration variables on the mission performance, gross weight, cost and many other parameters. Usually viscous and non-steady effects are neglected. These techniques use linearized potential flow computational routines and usually contain some approximation for compressibility effects. The latest programs can treat arbitrary configurations with lift. These recent developments were preceded by less general methods based on constant-strength source panel building blocks which could compute three dimensional flows over complex configurations, but, lacked the ability to handle problems involving circulation and lift (6). The first general routine for analyzing the flow about arbitrary configurations including both lift and thickness was assembled by combining the source panel and vortex lattice building blocks (7, 8). These complex programs have been enlarged and improved in recent years to the point where they now can handle complete configurations including nacelles, external stores with pylons and large protuberances. The majority of these methods make use of the feature of linearity: if a source, vortex or other such singularity can be used to represent a simple flow, a number of such singularities, of adequate strength and strategically located can represent the flow about more complex bodies. The configuration is subdivided into a number of panels and a system of equations is set-up, expressing the influence of every panel at a number of control points, where suitable boundary conditions must be satisfied. Solution of the resulting matrix yields the strengths of the singularities which in turn can be used to calculate the velocities and pressures.

Figure 10 shows typical representative configurations which can be analyzed by a program developed by The Boeing Company (9). This program has the capability of analyzing non-symmetrical configurations, as shown by the sketch of the yawed wing-body combination, bodies in ground effect, fan-in-wing configurations, external stores, pylon-nacelle junctions, large protuberances and wind tunnel wall effects. Internal flows in nacelles can also be analyzed. This program utilizes constant strength source panels distributed over the exterior surface to represent non-planar surfaces. Wing and tail surfaces are represented by source panels on the surface to account for thickness effects. Horseshoe vortices are distributed on the mean camber plane to account for lifting effects and also to neutralize excessive variations in source strengths on the surface when these occur.

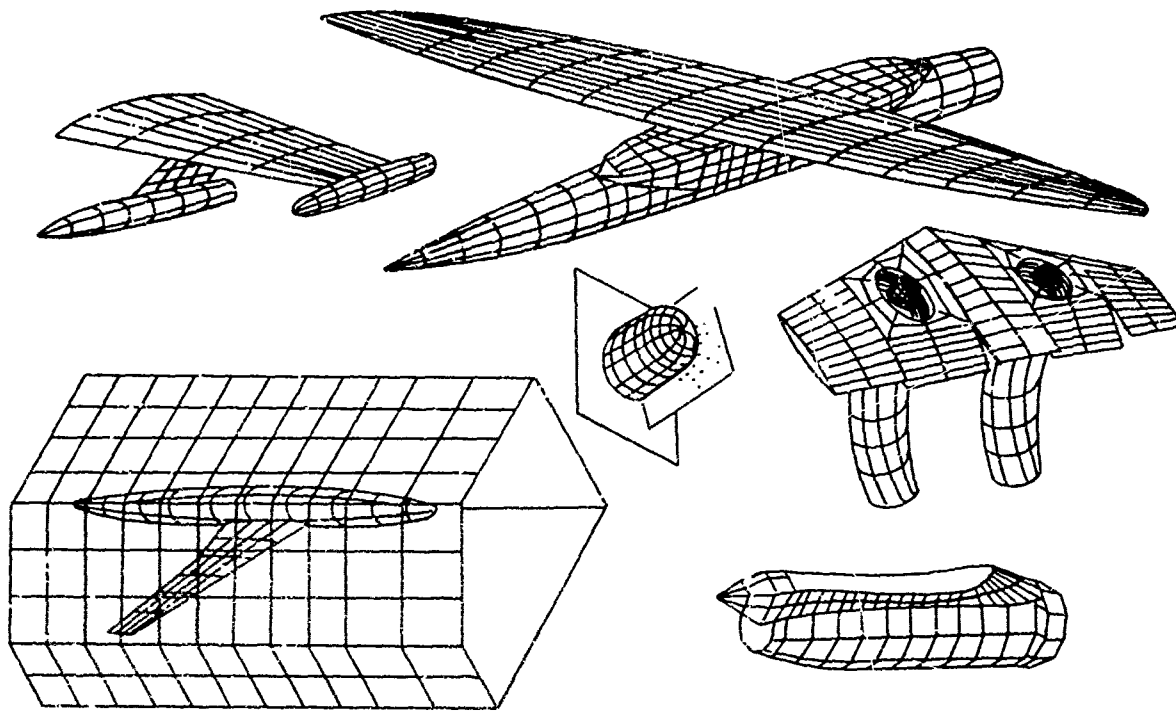


FIGURE 10 - TYPICAL CONFIGURATIONS

Most of current programs are satisfactory for computing wing pressure distributions along the span. Figure 11 shows a comparison of three different programs; the Boeing Program (TEA 230), a program developed by North American Rockwell and a NASA-Ames wing-body program. In order to reduce computation costs and simplify the analysis, the programs may contain assumptions which limit configuration details such as wing fillets and non-symmetric fuselage shapes. Figure 12 shows the detailed body panelling used for the NAR, Boeing and NASA-Ames Programs (19). It is obvious that an analysis of the wing-body junction would require detailed panelling to be used to simulate the complex flow patterns in this area.

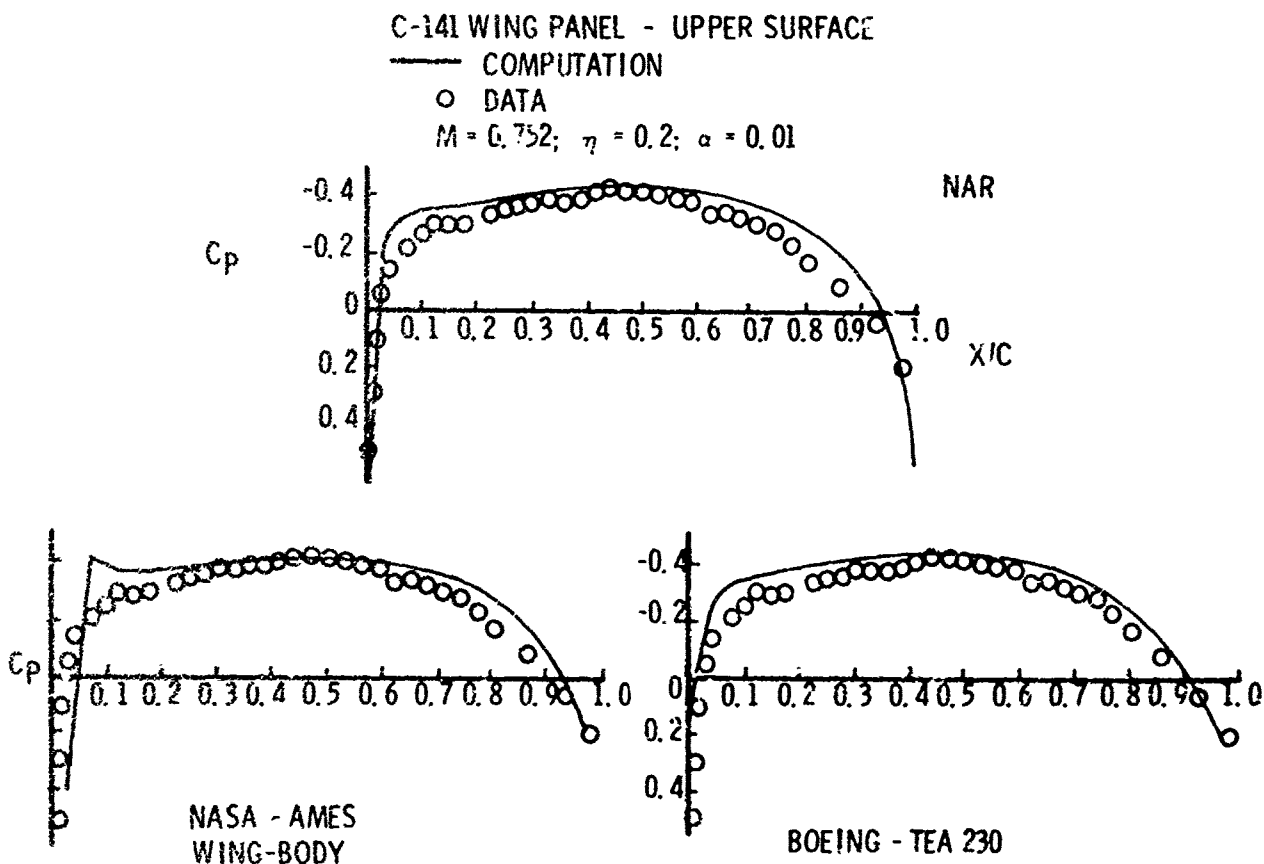


FIGURE 11 - COMPARISON OF THREE METHODS

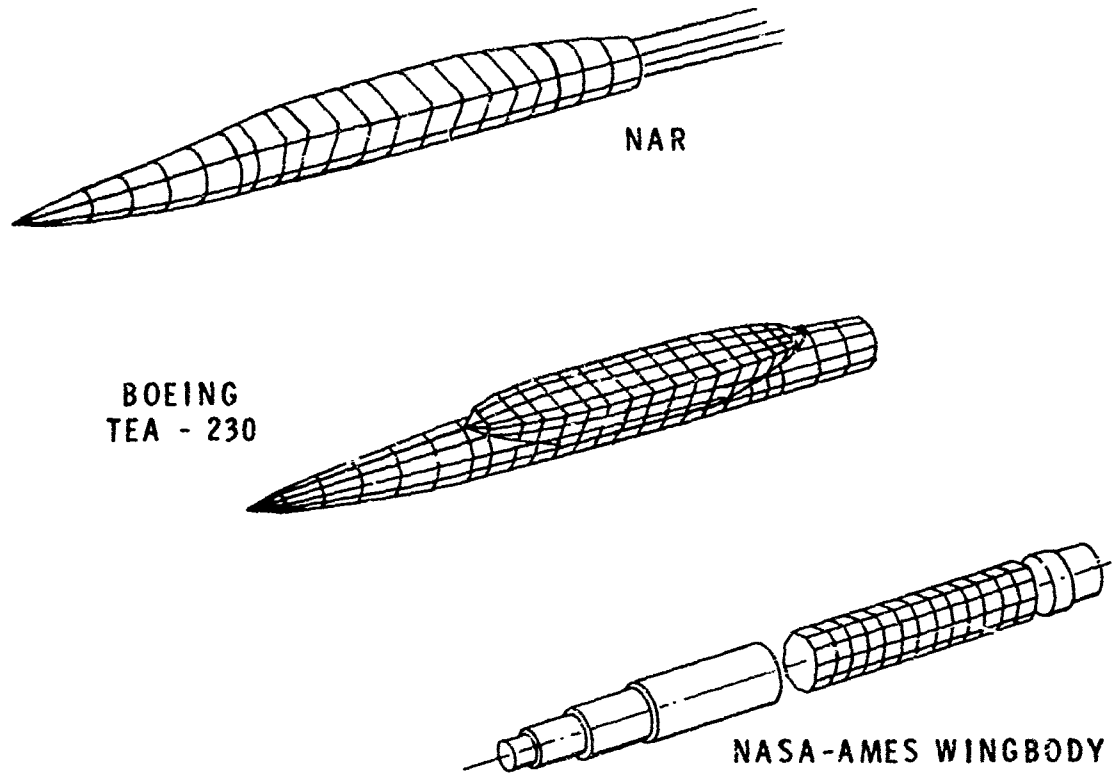


FIGURE 12 - BODY PANELLING

The real promise of these advanced programs lies in their ability to analyze detailed flow interactions associated with a fixed configuration. This is the type of problem often encountered in the detailed design of a new aircraft. The integration and design of some part of a configuration in the presence of other parts, whose shape may be relatively fixed, is a difficult task which may require a large number of wind tunnel models and a long series of tests. Figure 13 shows how the Boeing program was used for the detailed design of a wing-body leading edge fairing (9). Oil flow patterns indicated that the fuselage boundary layer was separating ahead of the wing leading edge causing a vortex which wrapped around the wing-body intersection.

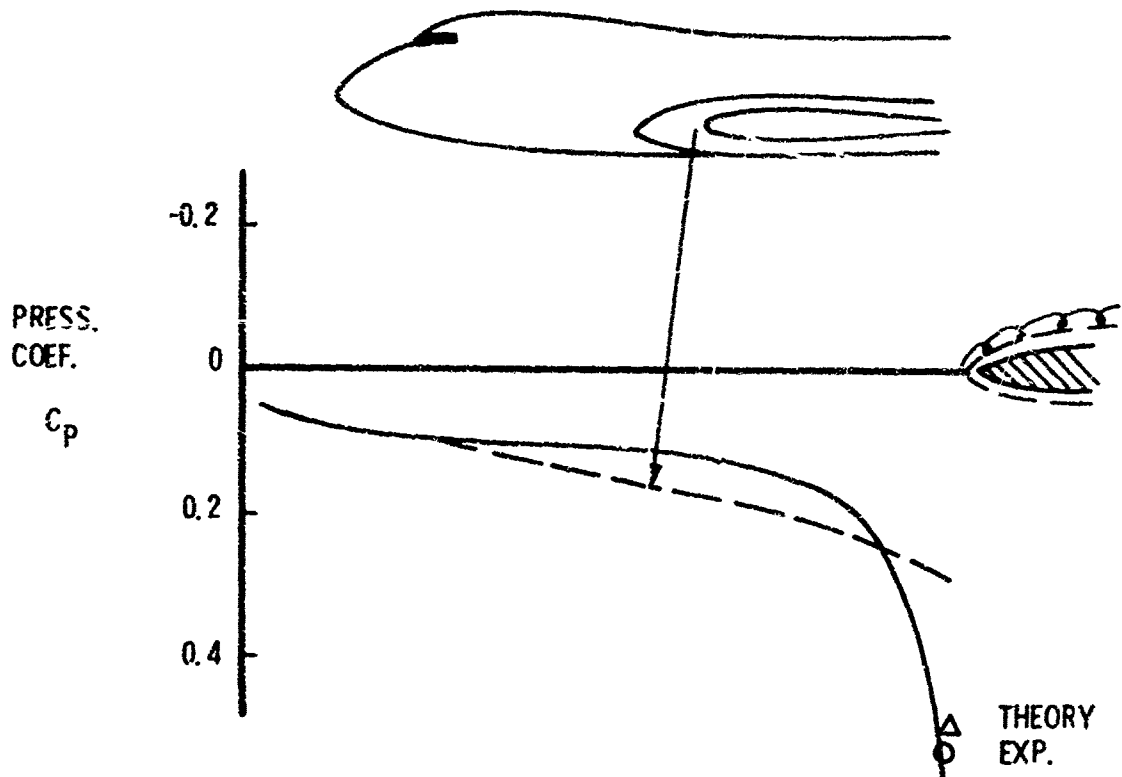


FIGURE 13 - WING-FUSELAGE FAIRING

In order to find a cure for this separation, an analysis was conducted using densely spaced fuselage panning to evaluate the fuselage pressure distribution without fairing shown by the solid line of this figure. A theoretical boundary layer analysis along the fuselage, using the calculated pressures, identified the problem to be the strong adverse pressure gradient approaching the wing leading edge intersection which resulted in flow separation. To cure this problem, flow field streamlines approaching the intersection were computed to determine the general path of the oncoming flow. A small fairing was then designed to be aligned with the oncoming flow and shaped to eliminate the local stagnation region. The theoretical pressure distribution along the fuselage and leading edge of the fairing is shown by the dotted line of this figure. The strong gradient was no longer present and oil flow patterns confirmed that flow separation had been eliminated. The good agreement between theory and the experimental data is shown by the two points on this figure. This example shows the success of this design approach and indicates that the technique of utilizing three dimensional lifting potential flow analyses to design and integrate the components of a new configuration may hold the key to better, more efficient aircraft. These techniques should be introduced into the design process at an early stage to guide the evolving design. The full potential of these new techniques will be lost if they are used only for preliminary design.

LOSS OF CONTROL AT HIGH ANGLES OF ATTACK

A problem area that often exists at the completion of the preliminary design process is the lack of high angle-of-attack aerodynamic data and related analyses of flying qualities at near-stall angles-of-attack. In the past, consideration and analyses of stall/post-stall/spin characteristics have been almost non-existent during the preliminary design process because of great emphasis on the important performance requirements that must be met within the operational flight envelope. Consequently, the degradation in lateral-directional stability at near-stall angles-of-attack that many of our older tactical aircraft exhibits became a serious problem in that maximum performance maneuvering potential was compromised and maneuvering boundaries were reduced. Departures from controlled flight often occurred close to the angle-of-attack for minimum turning radius because of poor lateral-directional static stability. The subsequent post-stall gyration, and sometimes spin, was responsible for a large loss of aircraft and crew.

In the USAF alone, loss of control accidents due to exceeding the flight envelope between 1 January 1965 and 30 September 1971 destroyed 229 aircraft, caused 233 crew fatalities which added up to a total loss of 315 million dollars.

Although departures from controlled flight and spins are usually associated with fighter-type tactical aircraft, other types of aircraft have not escaped this problem. Medium weight airplanes such as assault transports, heavy attack aircraft, and light weight airplanes like primary trainers and observation aircraft have all contributed to losses attributed to stall/spin problems. Indeed, there have even been cases of bomber-type aircraft entering a spin after a departure from controlled flight.

Preliminary design approaches and procedures should include initial investigations of aircraft stability and control characteristics at high angles-of-attack so that an early determination of possible flying qualities problems and the trade-offs that might be necessary relative to performance and the flight control system can be made. This will improve the probability of producing a design that will have a high degree of departure and spin resistance.

The kind of stability and control analysis that should be conducted during preliminary design, in order to obtain a reasonable estimate of flying qualities at near-stall angles-of-attack, involves the determination of high angle-of-attack basic aerodynamic data via the wind tunnel. These data might be supplemented, if necessary, with aerodynamic characteristics obtained by the many estimation methods available to the designer. Specifically, longitudinal and lateral-directional static stability characteristics should be determined up to at least the maximum trim angle-of-attack. Also, a stall/departure prevention device, as an integral part of the flight control system, should be studied very early in the preliminary design since this may be eventually needed or desired.

Analysis of developed spin characteristics should also be conducted as early as possible. The spin tunnel (vertical wind tunnel) can be used to determine developed spin modes and recovery characteristics. Models can be fabricated in such a way as to allow for design changes which might affect developed spin and recovery characteristics. An early entry into a spin tunnel is highly recommended.

Static aerodynamic characteristics at near-stall angles-of-attack should be obtained early during the initial design stages. The designer then has available basic data from which low speed, high angle-of-attack stability and control analysis can be conducted. At this point, a criterion or criteria for predicting departure boundaries and spin susceptibility would be desirable.

DEPARTURE AND SPIN SUSCEPTIBILITY CRITERIA

In recent years there have been several studies conducted to determine the applicability of existing stability criteria to predicting departure boundaries and spin susceptibility. These criteria are known as the " $C_{n\beta, \text{dynamic}}$ " and "lateral control divergence" parameters and are based on lateral-directional static stability. Many aircraft, particularly fighter-type aircraft, suffer serious degradations in lateral-directional stability at near-stall angles-of-attack and these undesirable characteristics are primarily responsible for losses attributed to stall/spin problems.

The parameter $C_{n\beta, \text{dynamic}}$ is generally a primary factor in determining the undamped natural frequency of the Dutch Roll mode and has been shown by Moul and Paulson to correlate with directional divergence of inertially slender configurations (12). Further, it was shown by Chambers that the expression $C_{n\beta, \text{dynamic}} < 0$ is an approximate criterion for divergence in the form of lateral-directional oscillatory instability (11).

$C_{n_{\beta, \text{dynamic}}}$ is defined as

$$C_{n_{\beta, \text{dynamic}}} = C_{n_{\beta}} \cos \alpha - (I_z/I_x) C_{l_{\beta}} \sin \alpha$$

The lateral control divergence parameter (LCDP) relates to divergence characteristics when lateral control is applied and is defined as

$$\text{LCDP} = C_{n_{\beta}} - C_{l_{\beta}} (C_{n_{\delta a}}/C_{l_{\delta a}})$$

LCDP must be positive to avoid lateral divergence and is an approximation to the undamped roll natural frequency squared (12).

One method of applying the criteria is shown in Figure 14. The parameters are calculated for a range of near-stall angles-of-attack, as high as the maximum trim angle-of-attack and preferably to five or ten degrees above this value. The aerodynamic characteristics used to calculate $C_{n_{\beta, \text{dynamic}}}$ and LCDP should be applicable over a relatively wide range of sideslip angles (at least over the entire linear range). Depending upon where values of the criteria plot a judgment can be made regarding departure and spin susceptibility.

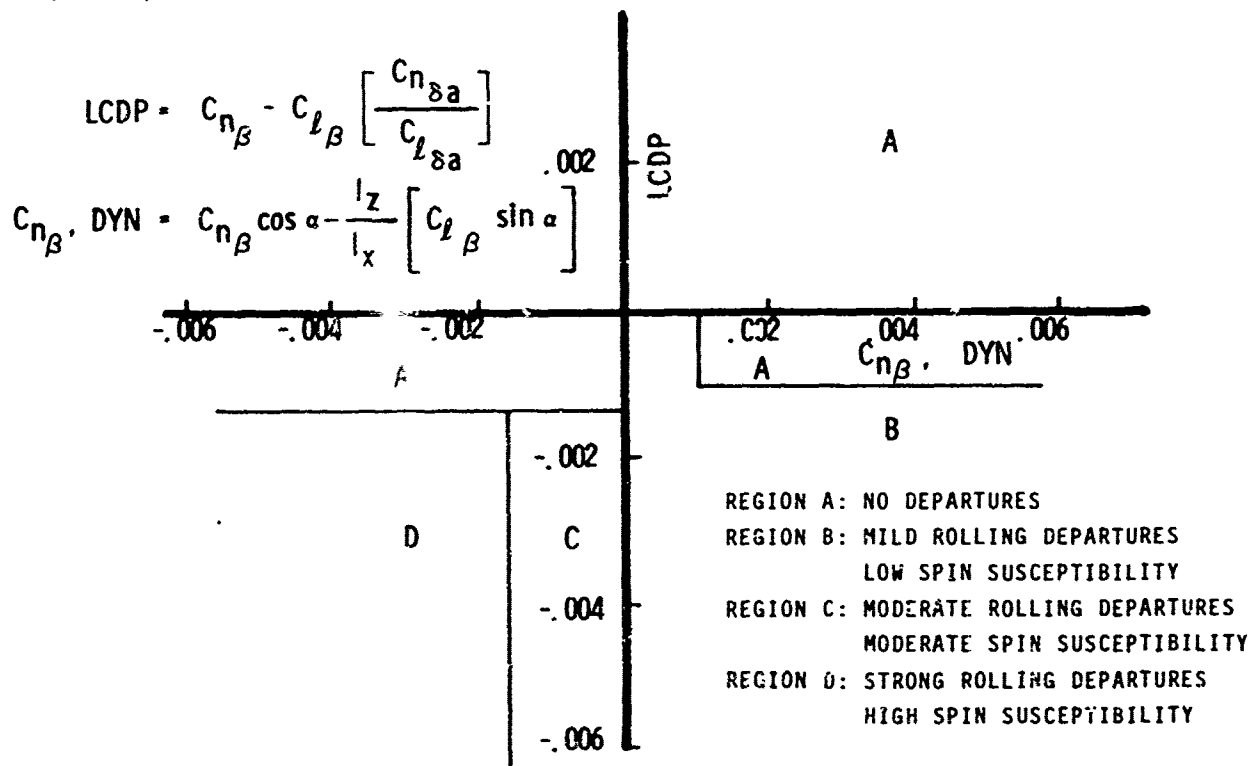


FIGURE 14 - SPIN SUSCEPTIBILITY CRITERIA

The regions shown in Figure 14 were determined from studies by Weissman which involved six degree of freedom motion analysis of airplanes where a full set of aerodynamic characteristics were available as well as stall/spin flight test results (13, 14). Briefly summarizing some of the results relative to Figure 14, the motion exhibited by a fighter-type aircraft laterally disturbed from a trim condition at near-stall angles-of-attack was analyzed to determine if a departure from controlled flight resulted as a consequence of the lateral disturbance. A departure is considered to be a rolling motion in a direction opposite to that commanded by lateral control input (roll reversal), a divergence in yaw (nose slice, directional divergence), or a combination of these two types of motion. At each trim angle-of-attack considered, the criteria were calculated and correlated with the computed motion. It was found that for values of $C_{n_{\beta, \text{dynamic}}}$ and LCDP in region A, there was no tendency to depart from controlled flight. With values of the criteria plotted in region B, the airplane motion exhibited a departure in the form of an initial divergence in yaw followed by a roll reversal. After completing two or three rolls, spins were usually obtained if no recovery control was applied following departure. The spin modes were steep and oscillatory with low rates of rotation or so oscillatory that a consistent buildup in angle-of-attack and yaw rate did not occur. Consequently, spin susceptibility is considered low in region B.

The motion of an airplane whose lateral-directional static characteristics were such that $C_{n_{\beta, \text{dynamic}}}$ and LCDP plotted in region C also exhibited a departure in the form of an initial directional divergence followed by a roll reversal in the same direction as the divergence. However, as indicated in Figure 14 the departure was more severe than that experienced in region B and there was a faster buildup as well as higher values of yaw rate and angle-of-attack in region C. The spin modes obtained in this region had higher rates of rotation, were not as oscillatory, and had higher average angles of attack. Spin susceptibility is

considered moderate in region C. Generally, region D represents values of the criteria where departure consists of a strong directional divergence together with some rolling motion. The airplane completed one to two roll in the same direction as the yawing motion and entered a spin while still rolling. Consequently, region D is considered a region of high spin susceptibility. An example of this motion is shown in Figure 15.

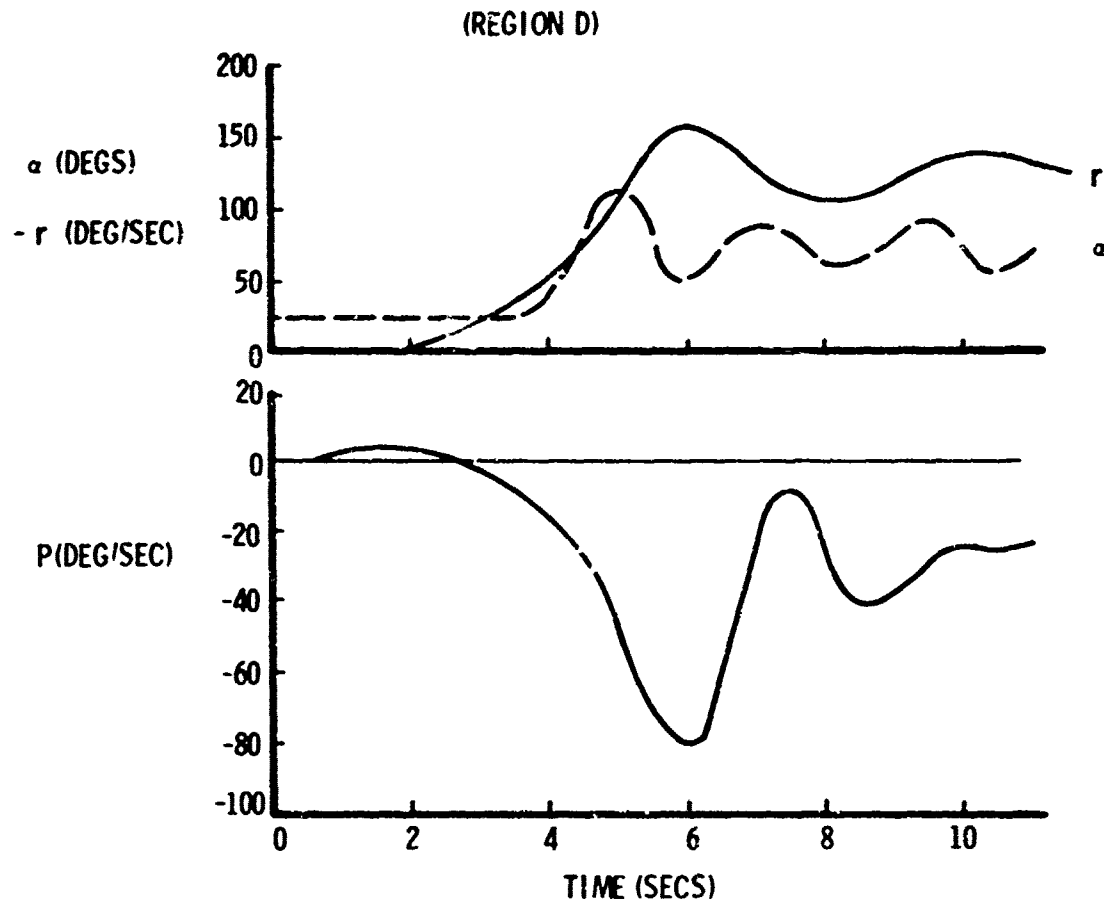


FIGURE 15 - SPIN DEPARTURE

A recently completed study by Larson deals with application of the linearized, uncoupled, small-perturbation lateral-directional equations of motion to an unsymmetric high angle-of-attack flight condition (15). The purpose of the study was to use the simplified equations of motion to define an airplane's angle-of-attack, sideslip angle envelope and from this be able to predict stability characteristics at near-stall angles-of-attack. It was found that the $C_{n\beta, \text{dynamic}}$ and lateral control departure parameters correlate very well with instabilities indicated by the roots of the characteristic equation for the particular airplane considered in this study (the A-7 aircraft).

Figure 16 is an example of the results obtained from this study. The line labeled " $C_{n\beta, \text{dynamic}}$ " represents the locus of the smallest sideslip angle at which the parameter first indicates instability (a negative value). Likewise, the lines labeled "LCDP" and "roots" represent the points where the lateral control departure parameter (LCDP) first becomes negative and where the real part of the Dutch Roll pair of complex roots first become positive. The dashed line is a departure boundary based on A-7 flight test data and considering that actual departures from controlled flight represent dynamic flight conditions, the correlation is quite good. It should be noted that this study represents an application to a particular aircraft and this method of analysis may not always result in as good a correlation for another aircraft. However, the method appears to satisfactorily predict the general conditions under which high angle-of-attack lateral-directional stability might be a problem and has promise for use in preliminary design.

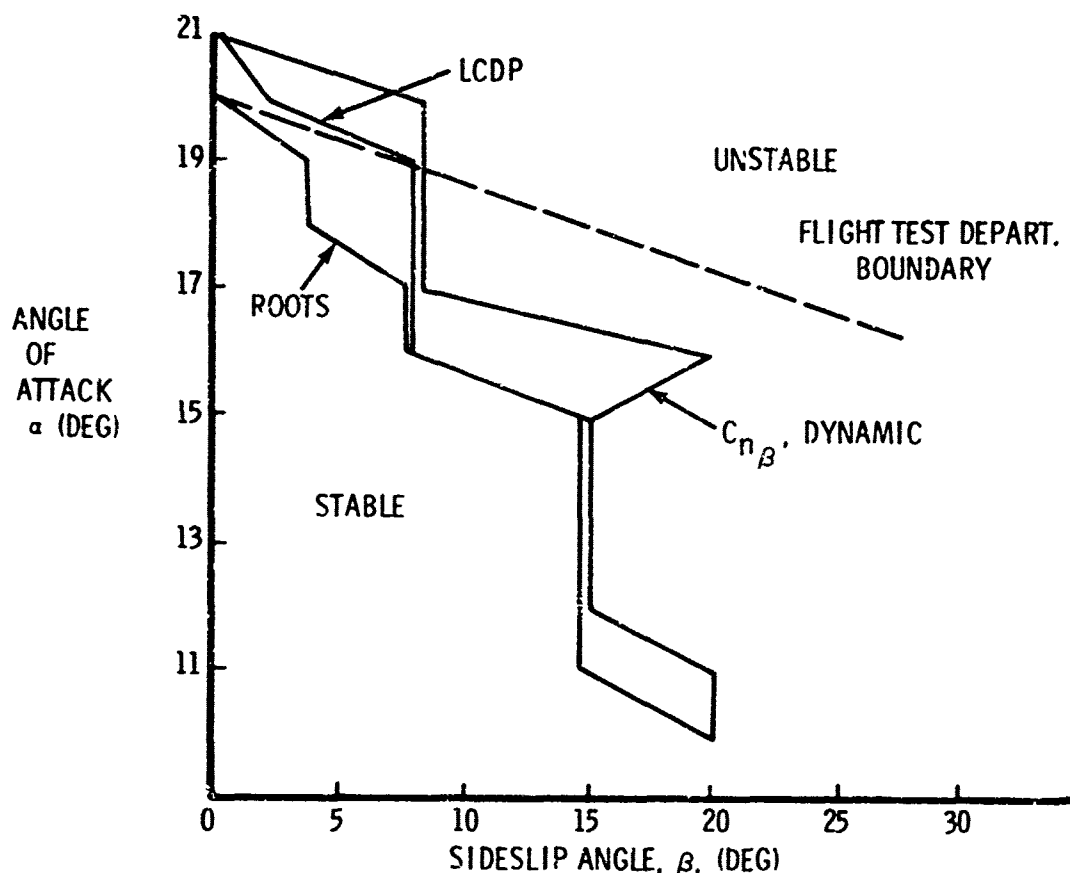


FIGURE 16 - CORRELATION OF PREDICTED DEPARTURE BOUNDARY WITH FLIGHT TEST RESULTS

CORRELATION OF CRITERIA WITH EXPERIMENTAL RESULTS

In terms of criteria based on static lateral-directional stability, results of a limited correlation with experimental data indicate that the $C_{n\beta, \text{dynamic}}$ and lateral control departure parameters can be used as preliminary design criteria for predicting departure characteristics and spin susceptibility. Figure 17 shows model and full-scale flight test results for several aircraft. The angle-of-attack noted next to each point is the angle-of-attack at which the criteria has been calculated.

As was noted above, for values of the criteria (at a given angle-of-attack) which plot in region A, there is no tendency to depart from controlled flight. Criteria values for the F-5 airplane will always be positive at near-stall angles-of-attack; consequently, the one point shown in figure 17 in the upper right hand quadrant is representative for this airplane. The maximum flight test excursions in angle-of-attack encountered during simulated air-to-air combat maneuvers were about 30 degrees and presented no problems regarding departure resistance. Some recent F-5 tethered free-flight model test results have also shown that the aircraft is virtually departure resistant to angles-of-attack greater than 30 degrees. Regarding spin susceptibility, an abnormally abrupt application of aft longitudinal control at maximum rate is needed to enter a spin.

Correlation of the criteria with F-4E departure characteristics is shown for the airplane with and without the effects of wing leading edge slats. Without leading edge slats (lower left hand quadrant) tethered free-flight model test results indicate directional divergence at an angle-of-attack of about 26 or 27 degrees. Full-scale flight test results show departures generally occurring between the 27 and 40 degree angle-of-attack range. F-4E tethered free-flight model test results show that the addition of slats delays the occurrence of directional divergence to an angle-of-attack of about 32 degrees (lower right hand quadrant) and limited full-scale flight tests indicated a mild yaw divergence at about 29 degrees angle-of-attack although this was not always experienced. The F-4E without leading slats is considered highly susceptible to spin following a departure if recovery control input is delayed or if the wrong recovery control technique is used. With leading edge slats, spin susceptibility is not as high because of the improved departure characteristics at the higher angles-of-attack.

Free-flight model tests have established a departure boundary as a function of angle-of-attack and wing sweep angle for the F-111 airplane. Full-scale flight test results are limited and for the most part unpublished. The data shown in Figure 17 for the F-111 at a wing sweep angle of 26 degrees indicates departure from controlled flight over an angle-of-attack range of 22 to 35 degrees. Model testing established the departure angle-of-attack for this wing sweep angle at about 25 degrees and for a sweep angle of 50 degrees, departure occurs at about 30 degrees angle-of-attack.

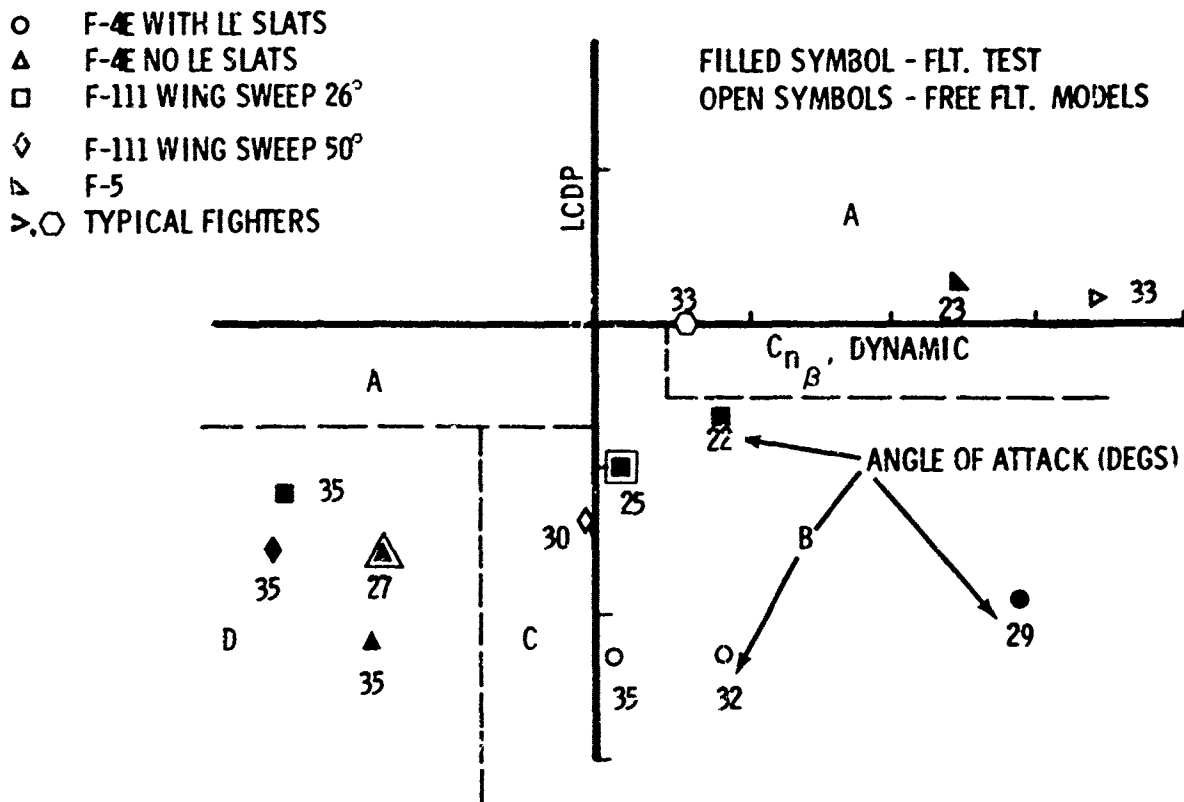


FIGURE 17 - CORRELATION OF MODEL TESTS WITH FLIGHT TESTS

Departure characteristics are basically rolling departures, the severity of the departure increasing as angle-of-attack increases. The airplane exhibits a strong directional divergence followed by a rapid rolling motion in the direction of yaw. The values of the criteria shown for the F-111 are in the regions where this kind of motion was obtained in the analytical studies described above. Full-scale flight test results are not now available for a 50 degree wing sweep angle; however, at a 45 degree sweep angle the airplane departs at about 30 degrees angle-of-attack. Spin susceptibility is considered high at any wing sweep angle, particularly at the higher angles-of-attack shown in Figure 17. Criteria value for a sweep angle of 50 degrees at an angle-of-attack (35 degrees) above the model test-determined departure angle-of-attack (30 degrees) is also shown.

CONCLUSIONS

The development of a new aircraft requires many trade-offs and design compromises which may introduce severe aerodynamic problems caused by complex flow interactions when certain components are integrated into the complete configuration. Early detection and correction of aerodynamic problems is essential to avoid costly design changes and possible delays in program schedules.

There is a very definite need for investigating aircraft stability and control characteristics at high angles-of-attack during preliminary design. Design criteria based on lateral-directional static stability have been developed to reveal possible problems relative to departure characteristics and spin susceptibility. In particular, results of a limited correlation indicate that directional and lateral control divergence parameters can be used as preliminary design criteria for analyzing departure characteristics and spin susceptibility of aircraft. It was found that these parameters correlate very well with instabilities indicated by the roots of the characteristic equation for one aircraft. Reasonably good correlation was obtained with the criteria using model and full-scale flight test results.

The application of three dimensional aerodynamic analysis methods and spin prevention criteria early in the development of a new aircraft may be the key to better design optimization and improved integration of the components. Further research to improve the accuracy of these prediction methods would provide a large payoff in terms of aerodynamic efficiency, mission capability, and life cycle costs.

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INITIAL-DESIGN OPTIMISATION ON CIVIL AND MILITARY AIRCRAFT

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SUMMARY

Aircraft design is an extremely complex process involving many interacting technical factors drawn from different engineering disciplines. Recent advances in computer technology and in mathematical optimisation theory have made possible the development of computer programs, combining aircraft initial-design equations with advanced numerical optimisation techniques, which are capable of defining the optimum aircraft design for a chosen standard of technology and a specified performance requirement.

This paper describes the development at the Royal Aircraft Establishment, Farnborough of a computer program which can optimise the preliminary design of a subsonic swept-wing jet transport aircraft. This program can be used to assess rapidly the effects on the optimum design of changes in the specified performance or of advances in aerodynamic, structural or engine technology. Compound optimisation functions including several of the aircraft characteristics, e.g. operating cost, noise, ride comfort, etc., with different weighting factors can be used to produce designs in which large improvements in some characteristics have been obtained at the cost of small penalties in others.

The development of a similar program for the initial-design optimisation of military aircraft presents some special problems; in particular, the definition of a compound optimisation function which accurately reflects the effectiveness of a military aircraft. These problems are discussed, and the utility and benefit of such a program for military aircraft are assessed.

1. INTRODUCTION

The task of finding the best aircraft design to fill a given role is extremely complicated. Even if a satisfactory criterion of merit can be defined, the designer is faced with a problem composed of a large number of interacting technical factors drawn from several different engineering and scientific disciplines, and by the need to satisfy numerous mission requirements and constraints. His difficulties are compounded by the rapid rate of technical progress in the aeronautical sciences, which forces the designer either to work with uncertain data from the frontiers of knowledge or to use proven technology and risk producing an obsolescent aircraft.

The designer generally places much reliance on the large body of experience, built up over a number of years in his design organisation, to guide him towards the best solution to any particular requirement. In the past such guidance has often been supplemented by parametric studies, in which one or two of the principal design characteristics, (e.g. wing loading, thrust/weight ratio) were varied to discover how the variation affected the aircraft design and performance. Although such studies did help designers with imagination and sound technical insight to improve aircraft designs, they were not entirely satisfactory because varying only a few of the design characteristics did not reveal the entire range of potential improvements, and because the results obtained by varying a single design characteristic while others remain fixed can be misleading, as the fixed values are necessarily most appropriate to one value of the varying characteristic and this value may not be the best.

However the advance of digital computer technology during the last two decades has led to dramatic changes in the type and complexity of the problems which computers can help to solve. Large computer programs are already being used to design¹ aerofoils with supercritical flow, analyse the design stress levels in aircraft structures², and calculate³ aircraft performance for complex mission profiles. During the same period improvements in numerical optimisation techniques have made possible the development of computer programs to optimise, for example, the structural design of an aircraft to have minimum mass yet sustain a prescribed load⁴, the shape of a supersonic transport to create the minimum sonic boom⁵, and the climb profile of a fighter aircraft to give the minimum time to intercept⁶. In addition to such programs used in the different aeronautical disciplines of structures aerodynamics and performance, several interdisciplinary programs^{7,8,9} to optimise the design of the whole aircraft at the initial-project stage have also been developed.

This paper describes in section 2 a computer program for aircraft design synthesis and optimisation which has been developed^{10,11,12} during the last few years in the Royal Aircraft Establishment at Farnborough. Although the detailed design and development of new aircraft in the United Kingdom is the concern of the airframe-manufacturing industry, the RAE is responsible for helping Government Departments to assess new projects and to guide the aeronautical research program towards those areas where improved knowledge will be important for the development of future aircraft. It is therefore necessary to have within the RAE a sound appreciation of the aircraft design process, and the capability of studying the effects on aircraft design of technological, commercial and military developments. To help provide this capability, the RAE computer program for transport aircraft design synthesis and optimisation was created by combining aircraft design equations for aerodynamics, structures and propulsion with a numerical optimisation technique so that the program could define the 'best' aircraft design for a specified mission. The program can choose the values of up to 20 of the principal design characteristics of a subsonic swept-wing jet transport aircraft which give the best value of a selected criterion of merit and satisfy the specified requirements and constraints, using a chosen standard of aeronautical technology. The criterion of merit, and any of the requirements and constraints, may if required be a complex function dependent on the aircraft design characteristics. The input data to the program must contain sufficient information on

the aerodynamic, engine and structural design standards for the program to calculate the economics and performance of each of the aircraft designs (typically about 10000) considered during the optimisation process. The output from the program contains the estimated values of the size, mass and cost of the various aircraft components, as well as the optimum values of the design characteristics and the performance of the optimum aircraft. The program is written in FORTRAN and, on an ICL 1907 computer, one optimisation requires about 30 minutes, including some 15 minutes of central processor time.

Section 3 discusses some of the problems which were encountered in the development of the program, and the methods which were used to overcome them. Some examples of the applications of the program are presented in section 4 each example illustrating a different facet of the program's capability. The examples demonstrate how the program may be used to investigate the effects of changing the specified field length¹³, of using an advanced standard of aerofoil design¹⁴ and of changing the criterion of merit^{15,16}.

Section 5 describes another program which is being developed at the RAE for the design synthesis and optimisation of military aircraft (other than transports which could be handled by the existing program). The primary purpose of the program for military aircraft, as of the program for transport aircraft, is to provide the means of rapidly identifying the best aircraft to perform a specified mission (thus providing the basis for more detailed design studies) and of assessing the effects of advances in aeronautical technology. A further, and perhaps more ambitious, purpose of the military aircraft design program is to assist the Air Staffs in framing their requirements by demonstrating how the chosen value of the aircraft performance affect its design and economy. In the current exploratory study a program for optimising the wing geometry of an aircraft required to satisfy given performance and mission requirements is being written, but it is expected that this program will be developed to include optimisation of the variables governing the mission, payload, range and performance. The special problems associated with the optimisation of military aircraft include

- (1) the difficulty of defining a satisfactory assessment criterion
- (2) the tendency of military aircraft to fly at the limits of the currently-attainable flight envelope where the data required to formulate aerodynamic design relationships is very scarce and very nonlinear
- (3) the integrated layouts characteristic of military aircraft with strong interactions between, for example, the fuselage size, engine dimensions, fuel capacity, etc.

These difficulties, though complex and considerable, should not be overestimated as they are not different in kind but only in degree, from those which have already been successfully overcome in developing the existing RAE program for design synthesis and optimisation of transport aircraft. Indeed it may be argued that the complexity of military aircraft design strengthens the case for computer-based optimisation studies because only this type of investigation can take account of all the relevant interactions and constraints.

The use of computer programs for optimising aircraft design should bring several important advantages. In the initial-design phase, the time spent in defining the best aircraft design to meet a complete set of requirements and constraints should be considerably reduced, thus shortening the dangerous gap between the military or commercial forecasts underlying the requirements and the aircraft's entry into service. Furthermore such programs can be used to indicate quickly the best way of exploiting an actual or anticipated advance in aeronautical technology, and the most cost-effective way of dealing, at any stage in development, with changes in the mission requirements or in the economic and technical design data. There is of course a danger that overreliance on such programs could lead to diminished understanding of the physical realities on which they depend, but this danger is lessened by analytical optimisation studies^{17,18} where these physical realities are revealed by the design equations rather than concealed behind the computer's blinking lights.

2 THE RAE PROGRAM FOR AIRCRAFT DESIGN SYNTHESIS AND OPTIMISATION

2 Description of the program

During the last few years a computer program which can optimise the design of a subsonic swept-wing jet transport aircraft has been developed^{10,12} in the Royal Aircraft Establishment at Farnborough. This program uses a numerical optimisation technique¹¹ to find a set of values for the *design variables*, defining the principal characteristics of the airframe and engine, which satisfies the specified *mission requirements*, *mission constraints* and *design constraints* and which gives the best value of the chosen *optimisation function*. The optimum values of the design variables are then used in conjunction with the *design parameters*, which govern the assumed standards of aerodynamic, structural and engine technology, to calculate the performance, mass breakdown and direct operating costs of the optimum aircraft.

The *design variables* to be optimised are

- wing area
- wing sweepback
- wing thickness/chord ratio
- wing aspect ratio
- wing taper ratio
- engine size
- chord and span of the flap
- chord of the full-span slat
- flap deflections on take-off and landing
- slat deflections on take-off and landing

cruising speed
cruising altitude.

During the optimisation process, each of these design variables may vary within a range bounded by two limits, fixed generally by the lack of sufficient reliable mass and performance data outside that range. Because the program is at present directed principally towards optimising the geometry of the wing and high-life devices, this geometry is specified in more detail than other parts of the aircraft; for example only the size of the engine is varied in the optimisation while its bypass ratio, fan pressure ratio, etc., are fixed.

The *mission requirements* are defined by the

number of passengers
mass of baggage and freight
furnishing and galley standards
range with full payload
diversion distance and holding time.

The number of passengers and the furnishing standard chosen are used to calculate the size and mass of the fuselage, which therefore remains fixed and plays no part in the optimisation process. The diversion distance and holding period are used to calculate the mass of fuel reserves required.

Each of the mission requirements is specified by a fixed value and the performance of the optimum aircraft must attain these values. But this performance must also satisfy several *mission constraints* which are

cruise speed must exceed a specified value
cruise altitude must exceed a specified value
take-off distance at max. take-off weight must be less than a specified value
engine-failed climb gradient at take-off speed must exceed the value specified
by airworthiness requirements
approach speed at normal landing weight must be less than a specified value

The lower limits on cruising speed and altitude are generally fixed by inter-airline competition or competition from other transport modes and by air traffic control requirements respectively. The limits on take-off distance and approach speed may be associated with specified values of runway roughness, altitude and temperature. The specified values of the mission constraints form upper (or lower) limits which must not be transgressed but do not define the performance of the optimum aircraft which is free to have a shorter take-off distance or a higher cruise speed than those specified, provided that the value of the optimisation function is improved thereby.

This design of the optimum aircraft must satisfy some *design constraints* which are

satisfactory longitudinal and lateral stability
acceptable fuselage angles on take-off and landing
adequate fuel tank capacity

It is assumed that satisfactory stability characteristics can be obtained by specifying that the fin and tailplane shall be large enough to give conventional values of the corresponding volume coefficients, and by ensuring that the wing sweep and aspect ratio do not combine to give unacceptable stalling stability characteristics. More detailed methods¹⁹ can be used to estimate the stability and controllability of the optimum aircraft, but these methods have not yet been incorporated into the optimisation program. To allow the aircraft to operate with less-than-maximum payload but over a greater range, the fuel tank capacity inside the wing-box of the optimum aircraft must exceed that required to accommodate the fuel required for a flight at maximum take-off mass but only a fraction (generally around 0.4) of the maximum payload.

At present the minimum direct operating cost is generally used as the *optimisation function*, i.e. the aircraft design variables are optimised to achieve the minimum possible value of direct operating cost, but it is interesting sometimes to use other optimisation functions to study the resulting changes in the design and performance of the optimum aircraft. For commercial transport aircraft with specified mission requirements, the alternative optimisation functions include

minimum first cost
minimum (fare + value of journey time)
minimum noise footprint area
minimum fuel consumed
maximum passenger comfort
maximum airline profit.

It is also possible to use in the program compound optimisation functions, combining two or more of those listed, and an example of this approach is given in section 4.4.

The *design parameters* are used in the design relationships to calculate the aerodynamic performance of the wing and high-lift devices at cruise, take-off and landing, the thrust and fuel consumption characteristics in cruise, diversion and hold conditions, the noise footprint area, and the masses, first costs and maintenance costs of each aircraft component. The aerodynamic parameters are those associated with a particular standard of aerofoil section design and with a particular type of high-lift system (e.g. a double-slotted Fowler flap combined with a leading-edge slat). The airframe mass parameters are determined by the chosen structural material and method of structural design and by the chosen arrangement of the wing and engines; rear-mounted engines must be associated with, for example, a mass penalty on the rear fuselage. The engine parameters, defining its thrust, fuel consumption and noise characteristics, are those associated with a particular type of engine design (e.g. bypass ratio = 5, single-stage fan),

with a particular level of powerplant technology (e.g. component efficiencies; turbine entry temperature, installation losses, etc.) and with a particular silencing arrangement (e.g. cowling length, wall liners, splitters etc.).

The characteristics associated with a certain engine type are chosen as fixed design parameters because variation of the engine design induces discontinuous changes in the characteristics (for example, the engine mass/unit thrust changes sharply as the design changes from a two-stage fan to a single-stage fan or from a single-stage fan to a geared fan) and such changes disrupt the optimisation procedure. Different engine types, or different types of high-lift systems, can of course be compared by repeating the optimisation with a different set of design parameters to find the optimum aircraft design associated with each type*. The design parameters defining the first costs and maintenance costs of the airframe and engine are based on the results of design studies by the airframe and aero-engine manufacturers; there is inevitably some uncertainty about the chosen values of the cost parameters but comparison of several optimised designs using the same parameters are unlikely to be misleading.

2.2 Optimisation procedure

The program is started by supplying to the computer some input data including values defining the

mission requirements,
mission constraints,
design constraints, and
design parameters

and values giving first guesses and acceptable limits of the design variables. These guessed values of the design variables are used in conjunction with the mission requirements and design parameters to design an aircraft which is capable of carrying the required payload over the required distance with appropriate fuel reserves, but which may not, and probably does not, satisfy the constraints. The program then alters the design variables to minimise the sum of the squares of the differences between the aircraft performance and the constraints which are not currently satisfied, and thus to obtain an aircraft design which can perform the specified mission and does satisfy the mission constraints and design constraints. This design is termed the 'first feasible solution'. For this design the program calculates the partial derivative with respect to each design variable of a penalised optimisation function which is the sum of the optimisation function and several penalty functions, each of which is proportional to the reciprocal of the distance of the design from one of the constraints. The use of these penalty functions prevents the constraints being crossed during the optimisation procedure and allows the use of a method for unconstrained minimisation of the penalised function; this program uses a modified version¹¹ of Davidon's gradient method. The partial derivatives are used to calculate the changes to the design variables which most rapidly improve the value of the penalised function. These changes define the direction of a vector in multi-dimensional space and the design variables are changed to move the aircraft design along this vector (the *search direction*) until the penalised function ceases to improve. At this point fresh partial derivatives are calculated and the cycle is repeated. When the value of the penalised function cannot be further improved the first estimated optimum design has been reached. The penalty functions are then altered to make them steeper thus changing the best value of the penalised function and the aircraft design associated with it, and the search procedure is repeated to find the second estimated optimum design. The completion of each search procedure marks the end of the 'stage' and successive stages are performed until the rate of change of the estimated optimum design becomes insignificant; experience of using the program to optimise the design of transport aircraft suggests that six stages is generally enough for such aircraft.

The optimisation procedure reviewed in this section is presented and discussed in more detail in Ref.11; the following five sections contain brief descriptions of the methods used to determine the aerodynamic, engine and performance characteristics, the mass breakdown and the noise footprint.

2.3 Aerodynamics

The aerodynamics section of the program computes lift and drag characteristics for the take-off, cruise, diversion, holding and landing phases of the flight. For the phases of the flight with the high-lift devices retracted, the lift and drag coefficients are derived directly from the mission requirements and from the independent design variables which together define the aircraft mass, wing geometry and cruise conditions. For take-off and landing, the lift and drag coefficients associated with the geometry of the particular wing and high-lift devices being considered are calculated from the known aerodynamic characteristics of a reference aircraft, using empirical factors to allow for the differences in the wing geometry (aspect ratio sweepback, thickness/chord ratio, taper ratio) and the size and deflection of the high-lift devices.

Thus, for example, the drag coefficient in the cruise condition is expressed as the sum of

the profile drag coefficient of the wing
the extra-to-wing profile drag coefficient
the additional profile drag coefficient due to compressibility, and
the lift-dependent drag coefficient.

The profile drag coefficient of the wing depends on the exposed wing area, the thickness/chord ratio, the aerodynamic sweep and the Reynolds number based on mean chord. The extra-to-wing drag depends on the wetted areas of the fuselage, empennage and nacelles. The drag increment due to compressibility is expressed as an empirical function, based on drag data from recent transport aircraft, of the difference between the cruise Mach number and the design Mach number of the wing; the design Mach number and the

* In an alternative version²⁰ of the program the variation of the engine characteristics with specific thrust and with the level of acoustic treatment are approximated by a series of smooth curves; aircraft designs can then be optimised with engine specific thrust and engine silencing as additional design variables.

sectional lift coefficient of the wing are related to the thickness/chord ratio and the aerodynamic sweep by an expression based on the drag-rise characteristics of rooftop aerofoil sections and including a coefficient defining the chosen standard of aerofoil section design. The lift-dependent drag depends on the square of the cruise lift coefficient, the aspect ratio and a typical value of the lift-dependent drag factor.

2.4 Engine

The performance of the engine during the take-off is estimated from the engine mass using two design parameters which define the engine mass/unit thrust and the variation of the engine thrust with forward speed. The engine performance in cruise is calculated from a design parameter giving the engine mass/unit cruise thrust at a chosen datum condition (e.g. $M = 0.8$ at 30000 ft) and a factor interpolated from data giving the variation of engine thrust with Mach number and altitude. Similarly the values of the specific fuel consumption at cruise, diversion and hold conditions are obtained by interpolation from data giving the variation of fuel consumption with Mach number, altitude and throttle setting. The wetting area of the engine nacelle needed for drag estimation is calculated from a fixed design parameter defining the nacelle area/unit thrust typical of the chosen engine type. The design parameters for thrust and specific fuel consumption include allowances for the effect of installation losses and of air and power offtakes.

2.5 Performance

The take-off distance is obtained from an expression involving the take-off mass of the aircraft, its static thrust, its wing area and the take-off lift coefficient calculated as described in section 2.3 above; these variables are related to the take-off distance by an empirical coefficient derived from an analysis²¹ of the take-off performance of current aircraft. The take-off lift coefficient is related to the value of the lift/drag ratio at take-off, and this value in turn is fixed by the take-off thrust/weight ratio and by the minimum acceptable engine-failed climb gradient at the take-off safety speed. Similarly the landing distance is obtained from an expression involving the approach speed and the standard of braking assumed.

An equivalent cruise range is calculated as the sum of the stage range specified in the mission requirement, and the lost range associated with the extra fuel used in the climb and descent; this lost range is calculated²² from the aircraft's design characteristics and its cruise speed and altitude. The fuel required for this equivalent cruise range, flown at constant speed and altitude, is then calculated using a formula which takes account of the change in the lift/drag ratio during the cruise. The diversion is treated in a similar way and is assumed to be flown at the best-range speed. The holding phase is assumed to be flown at the minimum drag speed.

2.6 Mass breakdown

The aircraft components may be divided (see Fig.3) into the payload-dependent items, such as fuselage, furnishings and crew, which do not vary during the optimisation and the items, such as the wing and tail structure, the engine, the systems and the undercarriage, which are dependent on the design variables.

In the first category the number of passengers, together with the chosen seating, galley and toilet standards, is used to define the fuselage dimensions and mass. The mass of furnishing and operators items is calculated from the number of passengers and a fixed design parameter, typically 55 kg per passenger.

The items dependent on the design variables need to be considered in much greater detail. The mass estimation must be as accurate as possible, but it is also essential when an optimisation procedure is being used that the partial derivatives of the various component masses with respect to each design variable should reflect the true situation as accurately as possible. The equations used for mass estimation in the design synthesis and optimisation program are therefore considerably more complex than the simple regression-analysis formulae commonly used in initial-design studies and are based on detailed analysis of the factors which determine the mass of each component. The wing-box mass for example, is given by the sum of the masses of the wing box covers, spar webs, ribs, joints, tip and undercarriage support structure and the masses of these six components are calculated independently as functions of the wing box chord, the wing area, aspect ratio, taper ratio, thickness, sweepback, design load factor and eleven empirical coefficients.

2.7 Noise footprint

Nowadays the noise of a transport aircraft is considered to be one of its most important design characteristics so, to help the comparison of the alternative aircraft designs produced by the design synthesis and optimisation, the program calculates for each design the shape of the footprint area inside the noise contour associated with a specified noise level. In this calculation it is assumed that the aircraft flight path consists of a straight landing approach of constant gradient, take-off ground run and a straight climb at the take-off safety speed and take-off thrust setting. It is also assumed that the engine noise may be represented by a point source with spherical symmetry so that as the aircraft moves along its approach, take-off and climb paths the instantaneously-spherical noise contours sweep out cylindrical noise contours with their axes along the flight paths. The intersections with the ground of the cylindrical noise contours around the approach and climb paths are elliptical so the noise footprint area within a specified noise contour consists of two semi-ellipses joined by a rectangle astride the take-off ground run, as illustrated in Fig.4.

The radius of the noise contours at take-off and approach is calculated from data on the variation of the noise level of the chosen engine type with distance and throttle setting, the data being adjusted to allow for the number of engines and the engine thrust on the aircraft design under consideration.

Although this method takes no account of the actual asymmetry of the engine noise source or of ground attenuation, it does provide a useful guide to the relative noisiness of alternative aircraft designs, whether this noisiness is expressed as the noise levels at selected measuring points or as the length or area of the noise footprint within a chosen noise contour.

3. DEVELOPMENT OF THE PROGRAM

In this section some of the experience gained during program development is presented and discussed. In general it was found that the initial version¹⁰ of the program needed more appropriate equations for estimating the aircraft characteristics as functions of the design variables and more detailed and accurate calculation methods. It was also found that the program could in the initial stages suffer from divergent design iterations, and could in the latter stages be satisfied with a local rather than a global optimum. At the end of this section the results of the aircraft design optimization program are compared with the characteristics of a transport aircraft.

During the development of the program, it was discovered that many of the design relationships used in the initial version¹⁰ were unsuitable for an aircraft design optimization program. These design relationships were the simple expressions used in project studies for estimating purposes in situations where the principal design characteristics of the aircraft considered are fixed, and they could not be used to provide realistic values of the rates of change of masses, performance, etc., with the design variables. As an example of this problem, it is interesting to consider the expression used in the initial version¹⁰ for the mass of the wing-box

$$m_{WB} = \frac{J_{WB} N^{\frac{3}{2}} A^{\frac{1}{2}} S (1 + \lambda) g m_{ZFW}}{t \cos^2 A}$$

where N = aircraft load factor
 A = wing aspect ratio
 S = wing area
 λ = wing taper ratio
 t/c = wing thickness/chord ratio
 A = wing sweepback
 m_{ZFW} = critical design mass
 J_{WB} = empirical constant
 g = acceleration due to gravity.

The use of this equation in the optimization program gave optimum values of the thickness/chord ratio which were unrealistically high, and it was realized that the presence of the thickness/chord ratio in the denominator of the wing-box mass equation was misleading. Study¹² of more accurate equations for the masses of the components on the wing box showed that the thickness/chord ratio appeared in the denominator of the expression for cover mass but appeared in the numerator of the expressions for the masses of the spars and ribs. The use of these more accurate equations in the optimization program gave more reliable values of the derivative of wing-box mass with respect to thickness/chord ratio and more realistic values of the optimum wing thickness. Another example is the expression used in the initial version¹⁰ to estimate the mass of systems and equipment as a fixed fraction of the take-off mass. Although adequate for project estimation, this approximation gives an unduly-high value of the derivative of systems mass with respect to take-off mass, and it was replaced by a more detailed equation¹² which relates different parts of the systems mass to the size of the payload, high-lift devices, empennage, wing span and take-off mass.

In other areas of the program it was found that more detailed calculation methods were required. For example, the stage fuel required was initially estimated using the Braguet equation and the fuel reserves and allowances were neglected. However, when the program was used to study short-range aircraft, it was necessary to include a method²² of estimating the range lost during climb and descent and a method of estimating the fuel reserves and allowances, not only because they constitute a significant proportion of the total fuel load of such aircraft but also because they affect the landing weight and hence the design of the wing and high-lift devices. The introduction of these methods allowed a more accurate assessment of different aircraft designs for short ranges. Another example concerns the tail arm, which was assumed in the initial version¹⁰ to be a fixed proportion of the fuselage length. It was realized that this assumption understated the penalty associated with increasing the size of rear-mounted engines, as this moved the centre of gravity aft and required larger fin and tailplane areas to maintain the same stability, and a balance equation was included¹² in the program to calculate the tail arm as a function of the centre of gravity position, which depends on the masses of the various components of the aircraft.

Some other imperfections in the program, however, could not so easily be corrected. It was found that first guesses for the values of the design variables, which must be included in the computer input data, must not be too far from the correct values otherwise the program is incapable, because of divergent design iterations, of designing an aircraft to fulfil the specified mission requirements. This breakdown of the program can generally be avoided by some simple project design calculations to guide the selection of the first-guess design variables.

Another problem, inherent in optimization programs, is the danger of the program selecting a local rather than a global optimum. This error is comparable to that of a mountaineer who sets out to climb a mountain by following the steepest upward path, reaches the top of one of the foothills, and believes, because the ground slopes downwards in every direction from the hilltop, that he has reached the mountain peak. The optimization program similarly plods 'upward' in multidimensional space, continuously improving the value of the optimization function until no further improvement seems possible but unable to lift its eyes to the higher peaks in the distance. In some cases²³ it is possible to establish the existence of a unique global optimum, or to divide the constrained region into a finite number of sub-regions each having a single local optimum which can be located and compared with the other local optima to find which of them is the global optimum. In most practical cases, however, the only procedure available is to repeat the optimization several times starting from several different first guesses; this procedure is similar to making the foolish mountaineer start from different points around the foot of the mountains. Another

technique, which may be used when the optimum value chosen for one of the design variables is particularly unexpected, is to fix the value of the suspicious variable at several points throughout its acceptable range, use the optimisation program to find the best design associated with each of these points and compare the different values of the optimisation function attained. This comparison should indicate whether the unexpected value of the design variable represents a local or a global optimum.

Local optima are more likely to occur as

- (1) the design relationships are made more detailed and nonlinear, because this makes the mountain contours more irregular
- (2) the number and complexity of the constraints is increased; constraints may be visualised as fences crossing the mountain, one or more of which may prevent the mountaineer from reaching the peak, and, if there are too many fences, the mountaineer may be guided into a cul-de-sac on the lower slopes
- (3) the number of design variables is increased, because this aggravates the problems associated with the design relationships and constraints.

They are less likely to occur if the program uses a small number of design variables, simple design relationships and a few simple constraints, or if one of the constraints is severe enough to confine the program to the steeper lower slopes of the mountain, where even complex design relationships only vary the steepness of the slope rather than producing hillocks. Contrary to the more gloomy forecasts, experience with the RAE program for optimising the design of transport aircraft has suggested that local optima are comparatively rare in this type of design problem, but the possibility of their occurrence must be kept firmly in mind when considering the results.

When these improvements had been made, as reported in Refs. 12 and 24, the program was tested by comparing the results of its calculation with the design characteristics of actual transport aircraft. In the first stage the values of the design variables

wing area
 wing sweepback
 wing thickness/chord ratio
 wing aspect ratio
 wing taper ratio

engine mass

chord and span of the flap
 chord of full-span slat
 flap deflections on take-off and landing
 slat deflections on take-off and landing

cruising speed
 cruising altitude

and of the mission specification and constraints

number of passengers
 mass of baggage and freight
 furnishing and galley standards
 range with full payload
 diversion distance and holding time

maximum take-off and landing distances
 minimum cruise speed and altitude

were obtained from published data on several transport aircraft. The mass breakdown and performance of each aircraft were calculated from the appropriate set of design variables using the design relationships in the program and the results compared with the actual mass breakdown and performance of the aircraft. The optimum values of the design variables, giving the aircraft with the minimum direct operating cost for the specified mission, were then determined using the design synthesis and optimisation program. Table 1 presents, as an example, the actual characteristics of a short-range twin-jet passenger transport aircraft, the mass breakdown and performance of this design calculated using the program's design relationships, and the characteristics of the optimum aircraft for the specified mission.

Comparison of columns one and two shows the accuracy of the design relationships in calculating the performance of engines, high-lift devices, etc., and the mass breakdown of the aircraft. Comparison of columns one and three shows that the design characteristics of the optimised aircraft are very close to those selected for an aircraft designed for the same mission. It is possible that the choice, in column three, of a comparatively-low wing sweepback represents a flaw in the design relationship in the program, but it is equally possible that the aircraft designer consciously (and prudently) chose a wing which was slightly larger, thicker and more highly swept than the optimum in order to obtain, at a negligible economic penalty, greater assurance of buffet-free flow and more fuel capacity than specified. Such comparisons with real aircraft have inspired confidence that the design relationships and design parameters in the transport aircraft design synthesis and optimisation program are reasonably accurate. The program has therefore been used for several research investigations, such as those discussed in the following section 4.

4. APPLICATIONS OF THE PROGRAM

4.1 Introduction

To demonstrate the value of the RAE computer program for aircraft design synthesis and optimisation, the following four sections describe some examples of the use of the program to study the effect on the optimum design and performance of a short-range swept-wing jet transport aircraft of

- (1) changing a design constraint (*viz.* field length)
- (2) using advanced technology (*viz.* advanced aerofoil section with supercritical flow)
- (3) using a compound optimisation function (*viz.* DOC + noise)
- (4) using an alternative optimisation function (*viz.* fuel consumed).

These examples have been drawn from the results of recent studies at RAE. Within each of the following four sections, the aircraft compared have been designed to satisfy exactly a chosen set of mission requirements and constraints; these sets are however different in different sections.

4.2 Effect of changing field length

As part of an assessment of future CTOL, RTOL and STOL projects in early 1972, the RAE program was used¹³ to study the effect on the optimum design of a future short-range 180-seat aircraft of changes in one of the design constraints, *viz.* the field length for take-off and landing. The principal design characteristics of four aircraft optimised to have the minimum direct operating cost with different field length constraints are presented in Table 2.

In studying Table 2, it should be noted that the wing sweep angle was limited in this study to values above 20° because of insufficient data below this value to substantiate the design equations; similarly the aspect ratio was limited to values below 9.5.

This table shows that reducing the allowable landing field length produces both an increase in the optimum approach lift coefficient and a decrease in the wing loading. The lower wing loading gives a larger wing mass fraction and a smaller lift/drag ratio in cruise, which in turn gives a larger cruise fuel fraction. At field lengths below 1500 m, the reduction of the cruise C_L , at constant cruise Mach number and wing sweepback, is associated with an increase in the allowable wing thickness/chord ratio. Despite the reduction in the allowable take-off field length, the static thrust/weight ratio remains almost constant as a result of a complex balance involving the take-off speed and lift/drag ratio and the thrust lapse rate with forward speed. The reduction in field length is also associated with an increase in the gust load factor, implying a noticeable worsening in the ride comfort for passengers.

In about 10 hours of computing time (on a ICL 1907 computer), the RAE initial-design and optimisation program produced, using consistent technological standards and estimation methods, more than 20 aircraft designs, each fully-optimised to satisfy *exactly* the specified mission requirements and constraints. The output data from each design included the optimum values of the design variables, the dimensions, area and masses of the principal aircraft components, aerodynamic performance data for different phases of the flight and an estimate of the direct operating cost. This example illustrates the utility of the program in providing *rapidly* a large number of *consistent* designs to answer questions on the effect of capacity, cruise performance, field length, etc. on the optimum design and performance of a transport aircraft.

4.3 Effect of advanced aerofoil section design

The RAE program was recently used¹⁴ to assess the effects on the design of a short-range swept-wing jet transport aircraft of

- advances in the design of aerofoil sections for high subsonic speeds
- improvements in powerplant performance
- the use of new materials in the wing-box construction
- weight savings in the fuselage, furnishings, empennage, systems and equipment.

The levels of improvement assumed in each of these areas were chosen to represent what might be achieved in a next-generation design, provided that research and development in these areas continues and means with the degree of success currently being predicted.

As an example of the use of the RAE program to assess the effects of advanced technology, Table 3 presents the leading characteristics of two aircraft, both of which were optimised to have the minimum possible value of direct operating cost. The datum aircraft was designed using current technology and the advanced aircraft was designed after introducing into the input data the assumed parameters for an advanced aerofoil section design. The aircraft were required to cruise at a speed of 940 km/h at an altitude of 9140 m; both the designs chose optimum values of speed and height equal to these lower limits.

The advance in aerofoil section design could have been used to increase the cruise Mach number, or to reduce the wing sweepback, or to increase the thickness/chord ratio, or to increase the cruise lift coefficient. However, the optimisation program decided that it would be best to use it to combine a moderate reduction in sweepback with a moderate increase in thickness/chord ratio. These changes produce a higher wing loading and improved values of the lift/drag ratio in cruise and at take-off, thereby reducing the fuel required for cruise and the required static thrust/weight ratio respectively. These improvements,

together with the decrease in the wing mass arising from the higher wing loading combine to give significant reductions in the take-off mass, the noise footprint area and the direct operating cost.

This example illustrates how the RAE program can be used to assess the *full* benefits which can be obtained from an advance in technology. The benefits obtained by a complete reoptimisation of the aircraft design are *much greater* than those obtained by using the advance to alter just one or two of the aircraft design characteristics.

4.4 Airframe design to reduce noise footprint area

The responsibility for making aircraft quieter does not fall on the aeroengine designer alone. The airframe designer can also make a significant contribution by biasing his design towards lower thrust requirements, so that less noise is generated, and/or steeper climb and approach paths, so that the source of noise is farther from populated areas. Various ways of achieving these aims have been identified, but it is important to discover how they may be achieved at minimum economic penalty. The RAE program provides a quick method of finding which of the design variables of an aircraft designed for minimum direct operating cost should be altered to reduce its noise at the smallest increase in cost.

In a recent study¹⁵, the design of a short-range swept-wing jet transport aircraft was optimised to give the minimum value of

$$\text{direct operating cost} + (\text{coefficient} \times \text{noise footprint area})$$

and the optimisation was repeated several times for different values of the coefficient, which can be regarded as a noise cost with units of £/km². Table 4 compares two aircraft with different values of this coefficient.

This table shows that the aircraft designed for a smaller noise footprint has lower wing loading, higher aspect ratio and smaller flap deflections on take-off and landing. These changes improve the take-off lift/drag ratio so that the climb gradient is increased, despite a virtually-constant static thrust/weight ratio; they also improve the lift/drag ratio in the approach and this combines with the steeper approach angle chosen to reduce the thrust required during the approach. In the calculation the approach throttle setting was constrained to be greater than 13%, as it was believed that at lower values the engine response in the case of a baulked approach would be unsatisfactory. The results suggest that a significant noise reduction (20% smaller footprint area is equivalent to about 2 PNdB quieter engine) can be obtained for a modest economic penalty by reoptimising the airframe design to reduce the noise footprint. The scope for noise reduction by this means is limited, but a blend of airframe design and engine silencing may be a more cost-effective way of achieving a given noise reduction than relying on silencing alone.

This study provides a simple example of the use of *compound optimisation functions*, i.e. functions which combine two or more of the aircraft characteristics, to improve the aircraft performance in one respect without greatly degrading it in another. For commercial transport aircraft this technique may be used to improve, for example, passenger comfort, stretch potential or operational flexibility at minimum economic penalty by a set of interrelated small changes to the design variables. For military aircraft, compound optimisation functions including several aspects of mission performance and base and flight-line maintenance might be appropriate, provided that the benefits and penalties of each of these aspects can be quantified (see section 5.3).

4.5 Aircraft designed for minimum fuel consumption

As an example of the use of alternative optimisation function, Table 5 presents the principal design characteristics of two aircraft, one of which was optimised to give the minimum direct operating cost while the other was optimised to consume the minimum fuel. These results were taken from a study¹⁶ of the effect of rising fuel prices on the design and performance of subsonic and supersonic aircraft.

The table shows that the aircraft designed for minimum fuel consumption has thinner wing with more sweepback to reduce the compressibility drag, and a higher aspect ratio and lower wing loading to give an improved lift/drag ratio at a higher cruise altitude. The large increase in wing weight is partially offset by a decrease in the stage fuel and allowances. As a result of the reoptimisation, the fuel consumed is reduced by 8½% at the cost of a 6½% rise in direct operating cost. It should be noted however that the use¹⁶ of a compound optimisation function in the form

$$\text{direct operating cost} + (\text{coefficient} \times \text{fuel consumed})$$

indicated how smaller fuel savings could be achieved much more economically than is implied by the quoted figures.

5. PROGRESS TOWARDS A PROGRAM FOR MILITARY AIRCRAFT DESIGN SYNTHESIS AND OPTIMISATION

5.1 General considerations

The success achieved with the design optimisation program for transport aircraft has provided sufficient incentive to embark on the development of an optimisation program for military aircraft. Certain sections of the transport aircraft program, for example the optimisation segment, can be applied directly to a military version. However, the wider flight envelope of military aircraft poses difficulties in defining the aerodynamic performance throughout this envelope, as a function of the aircraft geometry, with sufficient confidence and accuracy. The lack of an agreed optimisation function for military aircraft presents a further problem which is discussed in section 5.3.

Initial development of the program will be concentrated on the problem of optimising design variables, related directly to the geometric and engineering features of the aircraft, for given performance and

mission requirements. Subsequent expansion of the program will allow the inclusion of further design variables related to the mission, payload and performance and will therefore permit the aircraft cost/effectiveness to be minimised.

5.2 The design synthesis method

The RAE design synthesis method for military aircraft, which is currently used in its own right as an independent computer program for completing parametric studies, will be integrated with the existing optimisation segment¹ to form a new program.

The synthesis process consists of a method of estimating, as a function of the principal design variables, the mass and size of an aircraft capable of completing a prescribed mission. Initial work with the program will be concerned with defining optimum wing geometries for a variety of aircraft roles, consequently the design variables describe the wing geometry in greater detail compared with other parts of the aircraft. The current list of design variables is:

wing loading
thrust/weight ratio
wing sweep
wing aspect ratio
wing taper ratio
wing thickness/chord ratio.

This list of design variables, directly related to the aircraft, will be expanded at a later stage to include those associated with the engine and high-lift devices.

In the initial optimisation studies the aircraft will be designed to achieve specified minimum levels of the following performances:

take-off distance
landing distance
normal acceleration in sustained turn
normal acceleration in instantaneous turn
specific excess power
dash Mach number
low-level gust response.

The synthesis scheme provides the choice of basic aircraft layout, together with a corresponding fuselage packaging scheme. The forms of the design equations vary with the chosen layout and, because the fuselage is closely integrated, they also vary with the packaging scheme which defines the type and location of the engines, the type of cockpit, the undercarriage configuration, the distribution of fuel, the avionics fit etc. The fuselage is divided into a number of functional bays with corresponding lengths and gross surface areas. The lengths and gross surface areas of the nose and cockpit are assumed invariant with aircraft size whereas the lift engine bay, fuel bay and propulsion engine bay are functions of the design variables. For example, the size of the fuel bay is dependent on the quantity of fuel that can be carried in the wing and becomes a function of the wing geometry as well as the thrust/weight ratio which defines the fuselage cross-section.

An equation for the mass of the fuselage structure as a function of the gross surface area, taking into account the engine intakes, can then be obtained in terms of the design variables and the fuel available. The take-off mass of the aircraft is written as the sum of the component masses, in a similar fashion to the transport program, and the resulting equation solved to obtain the fuel available for the given set of design variables at a trial take-off mass. The geometry of the aircraft can then be specified from which the drag is calculated, together with other aerodynamic and engine properties required for the mission. The drag of a variety of external stores is available within the program together with interference factors depending on the proposed installation.

A simplified form of the RAE mission analysis program³ is included as a subroutine to determine the total fuel required for the specified mission, which can consist of up to 10 legs including search and combat phases together with store-dropping sequences. The take-off mass is changed by an iteration procedure and the complete design process and mission calculations repeated until a 'solution' aircraft is found such that the fuel capacity available is just sufficient for the fuel required for the mission.

Experience with the transport aircraft optimisation program has shown that a number of the procedures currently used in the military design synthesis process will be unsuitable when used in conjunction with an optimisation procedure. This arises particularly in the equations used to determine the component masses. It is essential that each equation should be derived from a realistic physical model describing the variation of a component mass with the design variables, rather than derived from a statistical correlation of existing data in terms of a convenient but unrealistic variable.

The estimation of the aerodynamic characteristics required for calculation of the drag within the synthesis process and for later calculation of the performance of the 'solution' aircraft presents some special problems. At present, many of the dependent variables such as lift-dependent drag factors, lift curve slope, are calculated with separate computer programs outside the synthesis scheme and inserted as data together with the design variables. Many of these computer programs are not entirely suitable for inclusion within an optimisation program because of either the large storage requirements, the large computing time or the fact that the methods are not based on sound physical principles; therefore, more realistic and efficient methods have to be formulated.

Although a number of these calculations are similar to the methods employed in the transport program, others are beyond the scope of recognized project estimation procedures because the military aircraft has to operate over a large range of speed and incidence compared with a transport aircraft. For example, the

manoeuvre performance of an air-superiority fighter is of paramount importance and would represent a flight condition towards which the wing design would be biased in an attempt to achieve the maximum possible lift and good buffet penetration without adverse handling qualities. This part of the flight envelope is characterised by the appearance of strong shock waves and large flow separations such that the classical-flow methods which can be used to estimate the lift, the drag or the lift-curve slope of high-aspect-ratio swept wings near the design condition are no longer applicable. The design of wings for operation under these conditions is complicated by the presence of external stores, the large fuselage interactions, the low aspect ratios usually employed and the presence of high-lift devices if these are considered for high speed manoeuvring.

Calculation of the aircraft performances is relatively straightforward provided the aerodynamic characteristics can be defined with sufficient accuracy. However, the methods available for calculating the take-off and landing distances of military aircraft are generally less accurate than for transport aircraft. This is principally due to the longer airborne distances involved, the absence of regulations defining pilot techniques and the special problems associated with V/STOL.

Fig. 5 shows an example of some typical results from the design synthesis scheme used to illustrate the increase in aircraft mass as a result of increased performance and mission requirements. The design synthesis method represents a first essential part of an optimisation program and its satisfactory operation, by comparison with results from more detailed design studies, has provided confidence in the techniques adopted. However, as indicated in the earlier discussion, these techniques will be refined after initial experience with the optimisation procedures.

5.3 Optimisation functions

Initial studies with the military optimisation program will use the take-off mass as the optimisation function principally because it is simple and, for aircraft with similar levels of technology, provides an indication of the relative manufacturing costs of different aircraft designs. At a later stage a better approximation to the manufacturing cost will be included in order to take account of the varying cost/unit weight of the component parts, in a similar fashion to the transport program. During the initial stages of a project the manufacturing and development costs would be of major concern but the total life-cycle costs are also important and can be expressed as the sum of:

research and development	- independent of number of aircraft
manufacturing	} - dependent on number of aircraft
fuel and weapons	
spares and servicing	
pilots and ground crew, including training ground support equipment, new bases etc.	

The cost-dependent optimisation functions that could be used are therefore:

take-off mass
 manufacturing cost
 manufacturing, research and development costs
 total life-cycle costs.

These functions should prove satisfactory for investigations in which the performance and mission requirements have already been specified so that the *capability*, which depends on the aircraft performance, and hence the *effectiveness*, which depends on the operational environment of the aircraft as well as its performance, remain essentially constant for each of the different aircraft produced during the optimisation process. If the aircraft capability varies significantly, particularly when the mission and performance requirements are not precisely defined, then the optimisation functions based on cost are inadequate. A simplified measure of the relative capabilities of different aircraft can be obtained by applying a 'figure of merit' to selected mission and performance parameters, e.g. range, payload, sustained turn rate, specific excess power, in relation to a predetermined scale. These figures of merit can then be summed, after applying weighting factors to distinguish those parameters of greater importance for the aircraft role considered, and the final figure used as an indication of the capability of the aircraft. The optimisation function would then become the cost/capability. This method is still inadequate in that it cannot represent the performance and mission parameters in the correct relation to provide a measure of the effectiveness. Therefore, a more detailed investigation of the operational aspects involved in defining an aircraft effectiveness is justified.

As an example of the parameters involved and the way in which they can be related to the characteristics of a particular aircraft, consider the war-time effectiveness of a battlefield support aircraft. The effectiveness can be judged by the total number of targets, N , hit during a given period of time. A simplified expression for the target hits during a period of d days with a fleet of n aircraft and a percentage attrition of A per sortie, can be obtained as

$$N = a \times s \times p \times n \left(1 - \frac{A}{100}\right)^{\sum_{r=0}^{(s \times d) - 1} \left(1 - \frac{A}{100}\right)^r}$$

on the assumption that the attrition for an aircraft on the out-bound and return journeys is the same,

where a is the number of attacks per aircraft per sortie
 s is the number of sorties per day per aircraft and
 p is the probability of a hit on a target.

These parameters must now be related to the characteristics of a particular aircraft and ultimately to the design variables.

If the payload is considered as a design variable and if only targets of opportunity are considered, the number of attacks per aircraft per sortie varies linearly with the payload, provided the range and field lengths have already been specified. However, it can only take integral values due to the assumption of a fixed number of weapons being required for each attack. If the range and field lengths are also design variables then the program must find the optimum combination of range, payload and field lengths.

The number of sorties per day per aircraft is a function of the duration of the mission, the time for refuelling and re-arming, the time associated with rectification of faults, and the probability of being unable to make a sortie due to major damage. The avionics fit affects whether night or all-weather operations are possible and together with the proposed operating environment, determine the time available per day for sorties. The reliability of the aircraft affects the time required to rectify faults whereas the probability of an aircraft requiring repair is related to the vulnerability and can be expressed in a simplified form as a proportion of the attrition.

The probability of a hit on a target is a function of the number of weapons used per attack, the accuracy of the nav/attack systems and the manoeuvre capability of the aircraft during target tracking. For the majority of studies the first two factors would be fixed and a suitable method for estimating the manoeuvre capability must be formulated in terms of the aircraft performances.

The attrition rate is a major parameter influencing the effectiveness and can be divided into attrition from ground defences and air-to-air attrition. For a low-level aircraft the ground attrition is related to its 'detectability' and its speed and low altitude capability in order to avoid ground defences. The latter capability is enhanced if terrain following or terrain avoidance equipment is fitted and the detectability depends on the aircraft size, shape, infra-red emission, etc. Similar features also affect the chances of detection by enemy air-superiority fighters but the air-to-air attrition is further influenced by the combat capability of the aircraft and the option of increasing this capability on an outbound journey by dropping the stores. In a similar manner to the manoeuvre capability, the combat capability must be formulated in terms of the relevant aircraft performances, e.g. sustained and instantaneous turn rates and specific excess power. The way in which these performances should be formulated, together with measures of the aircraft flying qualities has already received some attention²⁵.

Further aircraft losses occur, in addition to those caused by direct enemy action, if the base is subject to attack from the enemy, in which case it is necessary to estimate the probability of arriving at an alternative airfield. This problem has been investigated²⁶ as part of a model for evaluating the differences in the cost/effectiveness between V/STOL and CTOL combat aircraft.

The preceding discussion has given a brief outline of some of the problems associated with the formulation of an aircraft effectiveness. It is clear that a great deal of further work will be required, principally on operational aspects and their dependence on the basic characteristics of an aircraft, before a reliable measure of the effectiveness can be used with confidence in an optimisation program. The formulation of an effectiveness varies considerably for different aircraft roles, nevertheless many of the constituent parts, such as combat capability, sortie rates and attrition from ground defences, are common to different aircraft roles and therefore provide the components from which alternative measures of effectiveness can be constructed.

6. CONCLUDING REMARKS

This paper describes the development at the Royal Aircraft Establishment, Farnborough of a computer program for the initial-design synthesis and optimisation of subsonic swept-wing jet transport aircraft. This program provides a rapid method of finding the best design for a transport aircraft to satisfy specified mission requirements using given standards of technology. The aircraft designs obtained using the program correspond closely with those of real aircraft designed, using traditional methods, for the same performance and mission specification. The program has already been used to study the effects on the optimum aircraft design characteristics of changes in field length requirements, of a likely increase in the price of fuel, and of actual or anticipated advances in aerodynamic, structural or engine technology.

The program is being continuously improved by the introduction of more accurate design relationships; it is planned that in the near future an engine design method will be incorporated so that the program will be able to optimise simultaneously both the engine and airframe characteristics.

The success of the transport aircraft program has provided the incentive to embark on the development of a design optimisation program for military aircraft, using the same numerical optimisation technique. The satisfactory operation of a military aircraft design synthesis method provides the basis for the military optimisation program although some of the techniques used at present will need further refinement. Also, new methods for calculating aerodynamic characteristics over the large range of flight conditions encountered by a military aircraft will be required before real confidence can be placed on the results from an optimisation program.

The absence of a defined optimisation function for military aircraft presents a problem for later studies in which it is anticipated that the mission and performance parameters will be optimised to achieve a minimum cost/effectiveness. Some progress has already been made towards the definition of an aircraft effectiveness but a great deal of further work is required principally, on operational aspects.

Table 1 - Comparison with real aircraft

Specification		Aircraft A	Calculated mass breakdown	Optimized aircraft design
Number/class of passengers		100/tourist	100/tourist	100/tourist
Freight	kg	2280	2280	2280
Range with full payload	km	1800	1800	1800
Design variables				
Powerplant mass	kg	3440	3440	3450
Wing area (trapezoidal)	m ²	92.4	92.4	89.4
Wing sweepback	deg	20.0	20.0	14.9
Wing aspect ratio		8.79	8.79	8.89
Wing taper ratio		0.30	0.30	0.32
Wing thickness/chord	%	11.0	11.0	9.9
Flap chord	%	30.0	30.0	28.3
Flap deflection on take-off landing	deg	13/45	13/45	11/45
Mass breakdown				
Fuselage	kg	5200	5450	5450
Tail unit	kg	1100	1010	970
Wings	kg	4580	4350	4510
Powerplant	kg	3440	3440	3450
Undercarriage	kg	1400	1790	1790
Systems and avionics	kg	4400	4620	4550
Furnishings and supplies	kg	4160	3900	3900
Crew	kg	390	450	450
Fuel	kg	9110	9500	9270
Payload	kg	11250	11350	11350
TAKE-OFF MASS	kg	45200	45900	45700
Performance				
Take-off distance	m	2300	2310	2300
Engine failed climb gradient		0.024	0.019	0.024
Landing distance	m	1500	1490	1500
Cruise speed	km/h	850	850	850
Cruise altitude	m	9140	9140	9140

Table 2 - Effect of field length

Take-off and landing field	m	900	1200	1500	1800
Trapezium wing area	m ²	256	161	124	108
Aspect ratio		9.42	9.49	9.50	9.50
Sweepback	deg	20	20	20	24
Thickness/chord ratio	%	14.1	12.8	11.5	12.3
Fin + tailplane area	m ²	116	52	34	28
Static thrust	kN	2 x 164	2 x 136	2 x 130	2 x 125
Static thrust/weight ratio		0.348	0.354	0.362	0.359
Profile drag coefficient		0.0162	0.0185	0.0208	0.0224
Cruise lift coefficient		0.183	0.228	0.291	0.324
Cruise lift/drag ratio		9.7	10.7	11.4	11.6
Take-off speed	m/s	54	62	70	76
Take-off lift coefficient		2.13	2.08	2.02	1.91
Take-off lift/drag ratio		8.8	8.9	9.0	9.3
Approach lift coefficient		2.34	2.19	2.13	2.05
Wing loading	kg/m ²	376	487	593	660
Wing	kg	16620	10020	7830	6660
Fuselage	kg	11800	11800	11800	11800
Tail	kg	6010	2710	1390	1070
Undercarriage	kg	3460	2820	2640	2550
Engine	kg	11170	9250	8840	8490
Systems and avionics	kg	8940	7140	6620	6410
Furnishings and supplies	kg	7740	7740	7740	7740
Crew	kg	630	630	630	630
APS mass	kg	66370	51720	47500	45350
Payload	kg	16330	16330	16330	16330
Stage fuel and allowances	kg	9330	6960	6220	5930
Fuel reserves	kg	4110	3400	3320	3290
Take-off mass	kg	26110	78460	73360	70900
Gust load factor		5.76	4.96	4.38	4.05
Relative direct operating cost		1.341	1.102	1.073	1

Table 3 - Effect of advanced aerofoil design-

Aircraft		Datum	Advanced
Trapezium wing area	m ²	198	159
Aspect ratio		8.0	9.2
Sweepback	deg	26.0	20.3
Thickness/chord ratio	%	9.1	10.6
Rooftop position	%	57.0	64.4
Fin + tailplane area	m ²	78.7	60.3
Static thrust/weight ratio		0.337	0.330
Profile drag coefficient		0.0172	0.0194
Cruise lift coefficient		0.309	0.357
Cruise lift/drag ratio		12.9	13.3
Take-off lift coefficient		1.54	1.81
Take-off lift/drag ratio		10.0	10.3
Approach lift coefficient		1.79	2.08
Wing loading	kg/m ²	531	612
Wing	kg	13950	11060
Fuselage	kg	12980	12980
Empennage	kg	4020	2920
Undercarriage	kg	3600	3340
Engine	kg	12160	11030
Systems and avionics	kg	8560	8000
Furnishings and supplies	kg	9900	9900
Crew	kg	630	630
APS mass	kg	65800	58960
Payload	kg	19960	19960
Stage fuel and allowances	kg	14750	13310
Fuel reserves	kg	4640	4240
Take-off mass	kg	105140	97360
Relative direct operating cost		1.0	0.53
Relative noise footprint area		1.0	0.92

Table 4 - Design for reduced noise

Noise coefficient	f/km ²	0	3
Take-off mass	kg	75430	78990
Wing loading	kg/m ²	593	562
Trapezium wing area	m ²	127	141
Aspect ratio		9.4	10.5
Sweepback	deg	20.0	21.1
Thickness/chord ratio	%	11.5	12.4
Flap deflection at take-off	deg	26.6	20.8
Flap deflection at landing	deg	45.0	31.6
Static thrust/weight ratio		0.362	0.357
Take-off lift/drag ratio		8.95	10.51
Approach lift/drag ratio		6.44	8.31
Climb angle	deg	9.7	10.4
Approach angle	deg	3.0	4.5
Approach throttle setting	%	32	13
Relative footprint area		1	0.799
Relative direct operating cost		1	1.045

Table 5 - Design for minimum fuel

Optimisation function		DOC	Fuel used
Trapezium wing area	m ²	172	208
Aspect ratio		8.9	10.5
Sweepback	deg	23.7	27.5
Thickness/chord ratio	%	10.8	9.1
Fin + tailplane area	m ²	64.9	72.6
Static thrust/weight ratio		0.318	0.294
Profile drag coefficient		0.0186	0.0167
Cruise lift coefficient		0.365	0.379
Cruise lift/drag ratio		14.1	16.8
Cruise altitude	m	9140	10490
Take-off lift coefficient		1.64	1.55
Take-off lift/drag ratio		10.8	12.0
Approach lift coefficient		1.89	1.72
Wing loading	kg/m ²	586	515
Wing	kg	1060	17950
Fuselage	kg	12330	12330
Empennage	kg	2230	2620
Undercarriage	kg	3950	4200
Engine	kg	8040	8560
Furnishings and supplies	kg	9000	9000
Crew	kg	630	630
APS mass	kg	57440	65410
Payload	kg	19960	19960
Stage fuel and allowances	kg	19060	17440
Fuel reserves	kg	4130	4100
Take-off mass	kg	100 600	106 902
Relative direct operating cost		1	1.065
Relative fuel used		1	0.915

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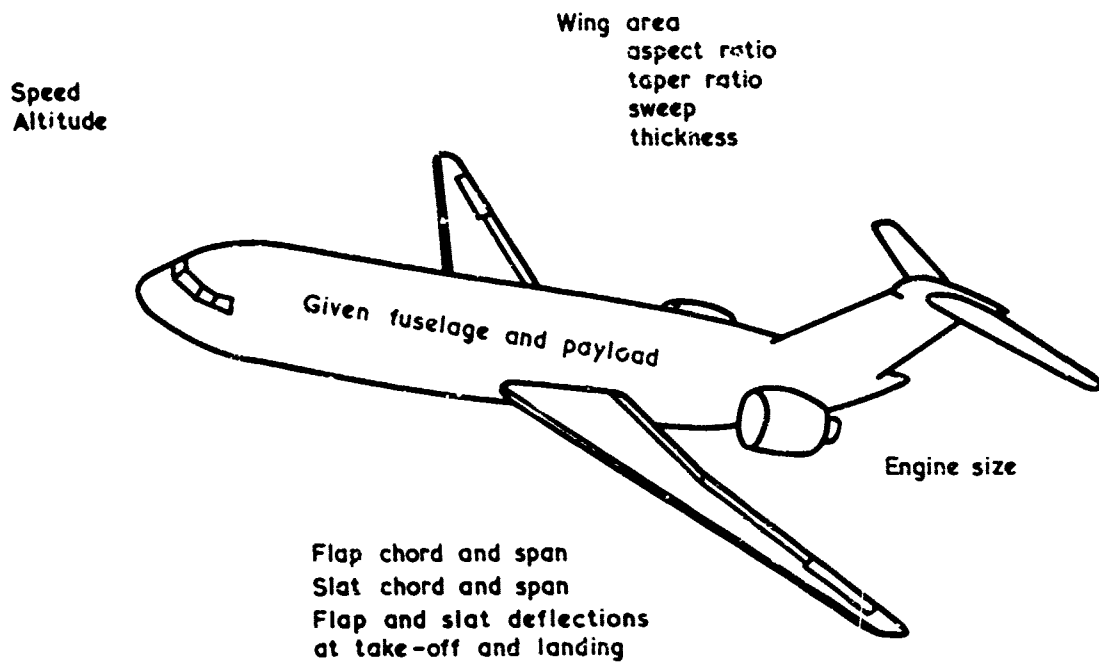


FIG.1 VARIABLES TO BE OPTIMISED

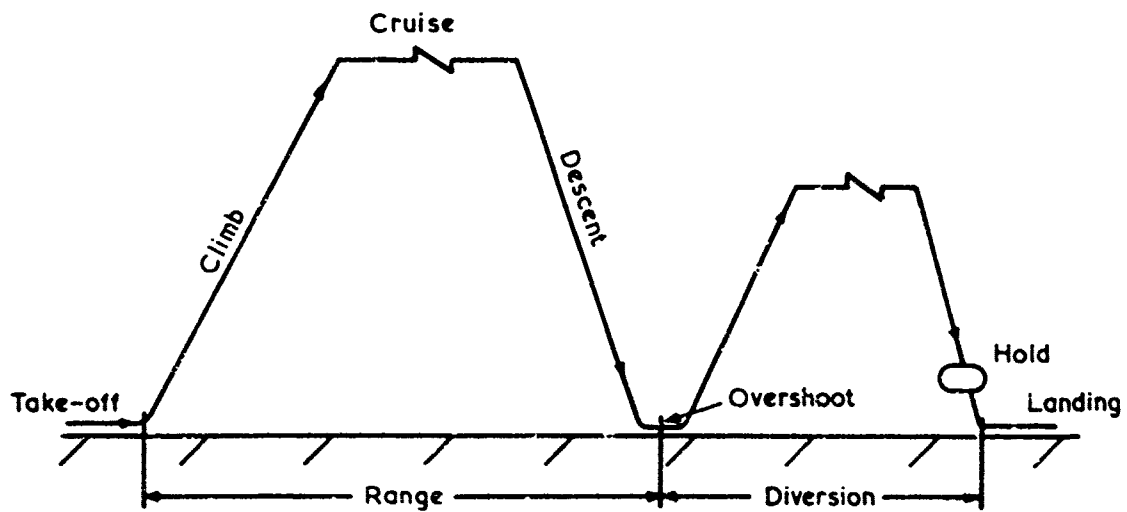


FIG.2 TRANSPORT AIRCRAFT MISSION PROFILE

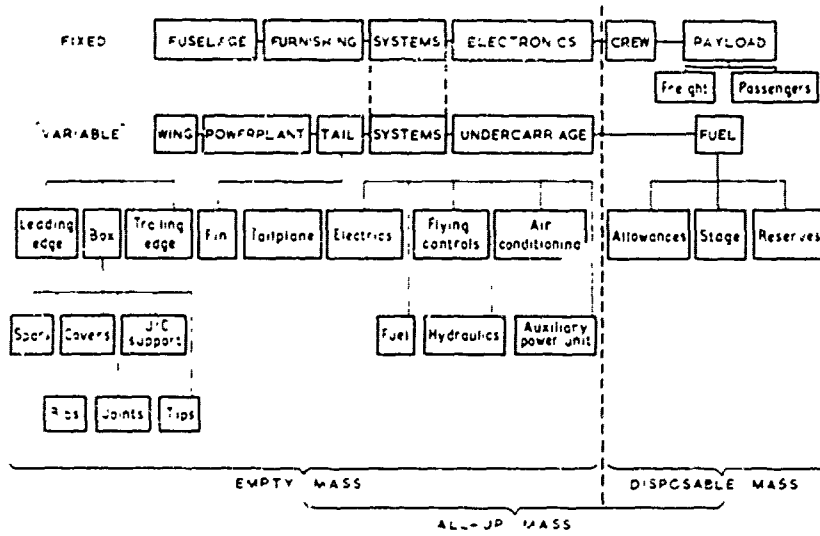


FIG 3 TRANSPORT AIRCRAFT MASS BREAKDOWN

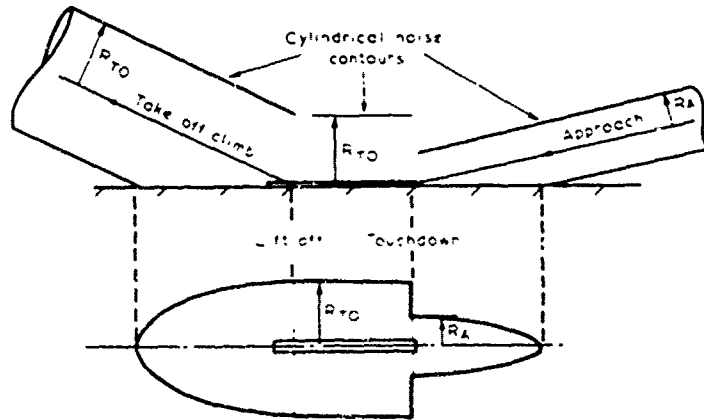


FIG 4 ASSUMED FLIGHT PATH AND SHAPE OF NOISE CONTOUR (NOT TO SCALE)

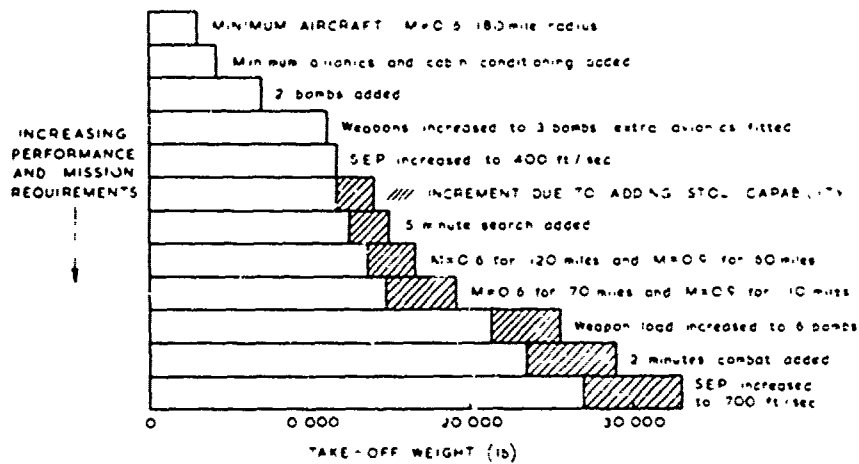


FIG 5 THE EFFECT OF INCREASING MISSION AND PERFORMANCE REQUIREMENTS ON THE SIZE OF A MILITARY AIRCRAFT

TECHNOLOGIES NOUVELLES ET RENTABILITE DES HELICOPTERES

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RESUME

Cette étude est divisée en trois parties. Dans la première partie nous proposons deux critères fondamentaux de rentabilité : d'une part, le "Prix de Revient Spécifique" qui constitue le repère du prix d'achat de l'hélicoptère ; d'autre part, le "Prix Spécifique Opérationnel" qui constitue le repère du coût en utilisation de l'hélicoptère. Dans la seconde partie nous abordons le problème du prix de la performance, de la sécurité et du confort pour les formules actuelles d'hélicoptères. Dans la troisième partie nous évaluons l'apport des technologies nouvelles dans le compromis coût-efficacité des hélicoptères.

INTRODUCTION

L'hélicoptère en tant que moyen de transport, en tant qu'outil ou en tant qu'arme répond à des besoins ; il assure donc certaines fonctions en remplissant certaines missions et en respectant certains critères de rentabilité. La détermination d'une mission sous tous ses aspects de performances, de sécurité, de confort, c'est-à-dire en fait la détermination de l'hélicoptère qui remplira cette mission, résulte donc de l'optimisation d'un enchevêtrement de compromis entre le coût des moyens à mettre en oeuvre pour fabriquer l'hélicoptère et la valeur des services rendus par cet hélicoptère. L'évolution des techniques permet de déplacer progressivement le point d'optimisation en reculant les limites physiques et économiques des performances, de la sécurité et du confort au niveau du développement, de l'industrialisation, de la fabrication et de l'utilisation. Tout choix technique doit donc découler directement d'une analyse de rentabilité aussi poussée que possible en tenant compte de toutes les possibilités offertes par la technique au moment du choix.

CRITERES DE RENTABILITE

REMARQUE PRELIMINAIRE

L'hélicoptère, de par ses caractéristiques particulières, est un véhicule capable de remplir un grand nombre de missions qui lui sont propres et qu'aucun autre moyen de transport civil ou militaire n'est capable de réaliser, du moins pour la plupart d'entre elles. Parmi ces missions, on peut citer le transport de charges en milieu accidenté, le secours en montagne, le ravitaillement de plates-formes pétrolières en mer, la mise à pied d'oeuvre rapide de commandos, etc ...

Mais l'hélicoptère ne se contente pas d'être un "outil" hautement spécialisé ; il peut aussi rivaliser avec certains transports traditionnels sur courtes ou moyennes distances grâce à ses performances intéressantes et à ses capacités d'export.

Or, cependant, malgré ses qualités, ce type d'aéronef a été longtemps handicapé par sa complexité et ses coûts : l'exploitation à l'heure de vol assez élevés. Cela est dû essentiellement aux caractères de dissymétrie et de variations cycliques de fonctionnement du rotor. Le fait que l'hélicoptère soit demeuré longtemps un appareil à vocation essentiellement militaire à une époque où la notion d'efficacité primait sur la notion de rentabilité a encore aggravé cette tendance. Depuis quelques années, des efforts importants ont été consentis par les constructeurs pour améliorer progressivement les appareils, non seulement sur le plan des performances mais aussi et surtout, sur celui de la rentabilité.

Dans notre civilisation moderne, cette notion de rentabilité tend à devenir capitale quel que soit le domaine que l'on aborde. L'utilisateur d'un produit cherche à savoir, avant d'investir, quel sera le meilleur usage qu'il pourra en tirer et surtout avec quel profit. Et ce n'est plus, dans cet esprit moderne, uniquement le prix d'achat qui compte, mais aussi ce que lui coûtera, à l'usage, le matériel et ce qu'il pourra lui rapporter finalement.

La rentabilité d'un matériel utilisé en transport de passagers par exemple est aisément calculable, par contre la notion de "service rendu" s'avère plus difficile à chiffrer sur le plan commercial. En effet certains services rendus revêtent apparemment un intérêt purement humanitaire, mais il est peu aisé de déterminer à première vue l'intérêt qu'on peut en retirer financièrement. C'est le cas, entre autre, du sauvetage en montagne qui met en oeuvre des moyens souvent considérables. Pourtant des études ont montré que la vie humaine, représentait un capital très important et qu'il était "rentable" de mettre en oeuvre des moyens appropriés pour la sauver. On pourrait citer aussi le cas des noeuds routiers dont la configuration dangereuse entraîne de nombreux accidents. Ce même genre d'étude montre qu'en chiffrant la vie humaine, il s'avère rentable d'investir des sommes importantes à l'amélioration du dit noeud routier.

Donc, la rentabilité, quelle que soit la façon dont on la calcule, est un élément déterminant dans le choix d'un matériel. L'hélicoptère en tant que moyen de transport de passagers ou de travail aérien peut être considéré comme un "outil" ; un outil qui coûte de l'argent à l'achat et en utilisation, mais qui doit rapporter des bénéfices à l'utilisateur. La rentabilité doit par conséquent être, au même titre que la performance, la sécurité et le confort, un souci majeur du constructeur.

NOTIONS DE PRIX SPECIFIQUE OU PRIX AU KILOGRAMME

Des critères simples sont nécessaires à l'hélicoptériste pour orienter et guider ses options techniques dans le respect des critères de rentabilité. Les notions de "prix spécifique" ou "prix au kilogramme d'hélicoptère" sont des points de repère fondamentaux, en particulier le "prix de revient spécifique" ou prix de revient du kilogramme fabriqué et le "prix spécifique opérationnel" ou prix du kilogramme de l'hélicoptère en utilisation.

Prix de revient spécifique :

C'est le prix de revient de l'hélicoptère pour l'hélicoptériste, c'est-à-dire le coût de fabrication plus l'amortissement des outillages et des frais d'étude et de mise au point divisés par la masse à vide de la machine. Il tient compte du prix des moteurs mais ne comprend pas le prix des équipements spéciaux. Ce prix moyen se situe aux environs de 800 F/Kg avec une fourchette de 500 F/Kg à 2000 F/Kg des appareils les plus rustiques aux appareils les plus sophistiqués. Il ne rend pas compte de la rentabilité de la machine mais il représente pour l'utilisateur la notion traditionnellement fondamentale de prix d'achat.

Prix spécifique opérationnel :

Par définition, c'est, pour l'utilisateur, le coût total des dépenses imposées par l'achat de l'hélicoptère et son utilisation durant toute son existence divisé par la charge payante moyenne. Le prix spécifique opérationnel est empreint de beaucoup moins de précision et de certitude que le prix de revient spécifique car on est obligé pour le calculer de faire de nombreuses hypothèses sur le profil et la rentabilité des missions effectuées. En supposant que les profits tirés de l'emploi de l'hélicoptère durant toute son existence, en moyenne dix ans, compensent, donc hors bénéfices, exactement l'ensemble des coûts d'achat et d'utilisation, on déduit que le prix spécifique opérationnel représente le manque à gagner imposé par le kilogramme d'hélicoptère supplémentaire. La valeur du prix spécifique opérationnel moyen varie beaucoup suivant le type et la durée de vie de la machine, suivant la nature et le nombre des missions etc... , il se situe aux environs de 10.000 F/Kg avec une fourchette de 5.000 F/Kg à 2.000 F/Kg, du travail à l'élingue sur hélicoptère léger de manoeuvre par exemple au transport de passagers sur hélicoptère moyen par exemple. Du point de vue de la rentabilité ce prix spécifique opérationnel est fondamental pour l'hélicoptériste.

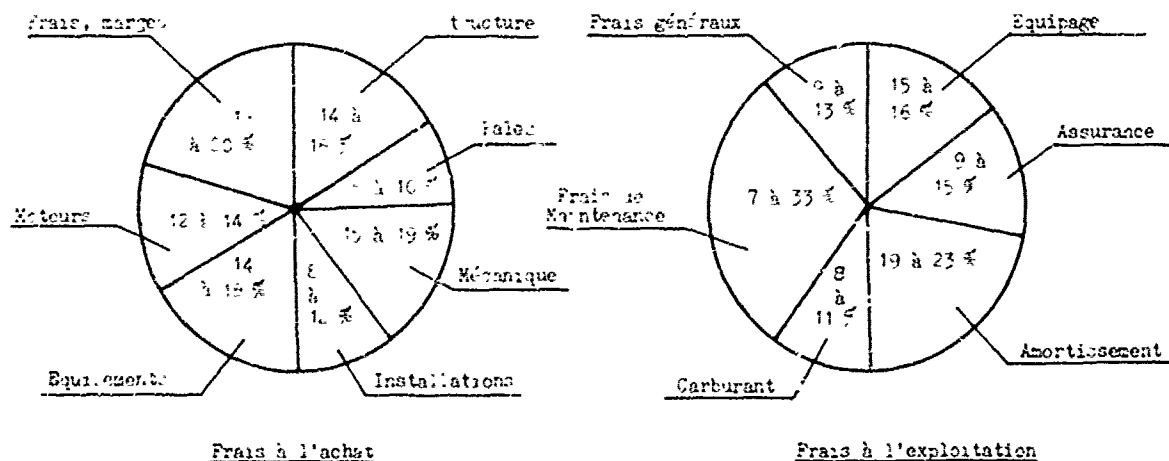


Figure 1 : Répartition moyenne des frais

CARACTERISTIQUES FONDAMENTALES DE L'HELICOPTERE

Les trois caractéristiques fondamentales d'un hélicoptère, celles qui conditionnent ses missions et par conséquent sa rentabilité sont : l'ensemble de ses performances, sa sécurité et son confort. Nous tenterons, dans les lignes qui suivent, de dégager à l'aide d'exemples les principaux critères d'optimisation de chacune de ses caractéristiques fondamentales, c'est-à-dire en d'autres termes de chiffrer l'augmentation de coût imposée par un accroissement de performances, de sécurité ou de confort, avec les moyens techniques industriels actuels pour les formules d'hélicoptères existants.

L'AUGMENTATION DES PERFORMANCES

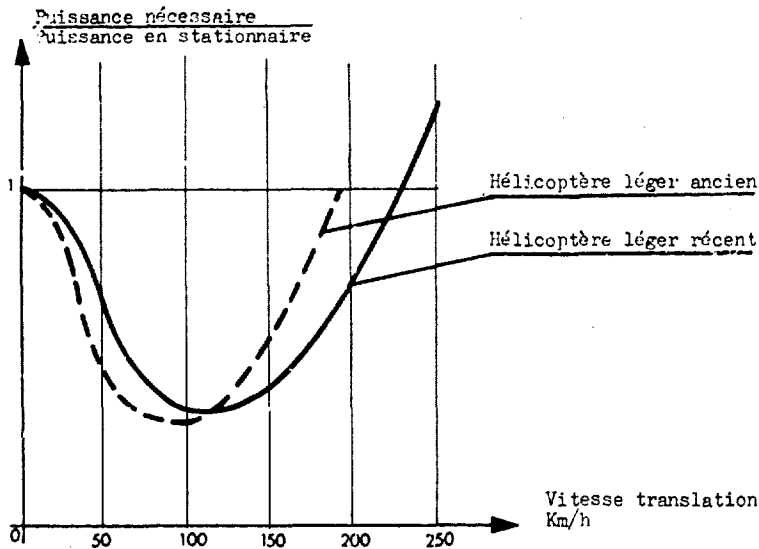
Aspects techniques de la vitesse

Les progrès réalisés dans le domaine des performances, particulièrement en ce qui concerne la vitesse, depuis les premiers hélicoptères opérationnels sont remarquables. L'accroissement constant des records illustre parfaitement ces progrès (bien que les possibilités courantes des appareils de série soient légèrement en retrait) :

178 Km/h	en 1944	sur Sikorsky R 5 A
208 Km/h	en 1949	sur Sikorsky S 52
236 Km/h	en 1953	sur Piasecki YH 21
350 Km/h	en 1963	sur SA. 3210 Super-Frelon
355 Km/h	en 1970	sur Sikorsky S 67.

Les gains sur les vitesses maximales s'obtiennent à partir de l'augmentation de la puissance, de l'affinement des fuselages et de l'amélioration des paramètres de fonctionnement aérodynamique et dynamique du rotor. (La vraie solution, au-delà d'une certaine vitesse, réside davantage dans l'augmentation de la finesse que dans la surmotorisation). Des travaux importants sont actuellement en cours, chez la plupart des constructeurs d'hélicoptères dans ce domaine. Une anticipation prudente sur les progrès à obtenir dans les années à venir permet d'affirmer que l'hélicoptère va suivre une évolution parallèle à celle qu'a suivi l'avion et réduire ainsi son écart de finesse ; il est plausible d'envisager les hélicoptères futurs volant couramment dans la gamme de 350 Km/h avec des vitesses de pointe de l'ordre de 400 Km/h.

Mais l'augmentation de la vitesse est coûteuse. D'abord parce qu'elle exige de la puissance supplémentaire : au-delà de 250 Km/h la puissance nécessaire varie sensiblement comme la puissance 2,5 de la vitesse toutes choses égales par ailleurs, ce qui implique un accroissement notable de la consommation des moteurs, et même dans



certain cas d'hélicoptère monomoteur non surmotorisé, la nécessité de monter un moteur plus puissant. Ensuite parce que conjointement à cette augmentation de puissance elle suppose un affinement du fuselage ce qui implique certaines contingences sur le volume interne, la nécessité d'un train d'atterrissage rentrant, la nécessité de carénages appropriés, etc ... Enfin parce qu'elle entraîne une augmentation des efforts aérodynamiques vibratoires appliqués sur les pales qui devraient se traduire par une élévation du taux de travail en fatigue de l'ensemble des éléments mécaniques travaillant et une détérioration du niveau vibratoire du fuselage et qui implique en conséquence une optimisation beaucoup plus poussée des caractéristiques aérodynamiques et dynamiques des pales et du comportement vibratoire de la structure.

Figure 2 : Puissance nécessaire en fonction de la vitesse de translation

Répercussion sur le prix de revient spécifique

Il est difficile de chiffrer avec précision le coût de la vitesse au niveau de l'hélicoptériste, mais à titre d'exemple d'ordre de grandeur l'on pourrait dire que dans le domaine de l'hélicoptère léger actuel, avec les moyens technologiques existants, le passage de la vitesse de croisière économique de 200 à 250 Km/h augmente de 30 à 40 % les frais de développement et de 15 à 25 % le prix de vente de la machine, par les répercussions entraînées par l'affinement aérodynamique et dynamique du rotor, l'affinement aérodynamique du fuselage, l'optimisation vibratoire de l'ensemble plus poussée et par l'exigence d'un moteur plus puissant.

Répercussion sur le prix spécifique opérationnel

Prenons l'exemple sur hélicoptère moyen 7 tonnes du passage de 250 à 275 Km/h dans la mission transport de fret interne sur 150 Km. Sans rien modifier à la machine, l'augmentation de consommation en carburant neutralise le gain obtenu par la réduction du temps de mission, le prix spécifique opérationnel n'est pas réduit, cette opération n'est pas rentable. Par contre si l'on compense la perte due à l'augmentation de carburant par l'adoption de moteurs de nouvelle génération (actuellement en développement ; de puissance massique double et de consommation spécifique réduite du tiers), le prix spécifique opérationnel est réduit de plus de 10 %, ce qui en d'autres termes signifie que l'hélicoptère "nouveaux moteurs" à 275 Km/h, sera 10 % plus rentabilisable que l'hélicoptère de base à 250 Km/h. Si l'on admet que les bénéfices de l'utilisateur représentent 10 % de la somme totale investie, cela signifie que cette opération double les bénéfices.

L'AUGMENTATION DE LA SECURITE

Très schématiquement et du strict point de vue de la sécurité, il est de règle de classer les principaux éléments vitaux d'un hélicoptère en trois grandes catégories : ceux dont la détérioration en vol met en péril la machine ; ceux dont la détérioration risque d'interrompre la mission sans mettre en péril la machine ; ceux dont la détérioration nécessite une intervention après vol sans compromettre la mission. Du point de vue du mode de travail, il y a les éléments à durée de vie qui travaillent en fatigue, et les éléments à potentiel, essentiellement les pignons et les roulements.

Eléments à durée de vie

La sécurité a été assurée jusqu'à ces dernières années, pour les éléments travaillant en fatigue, par la notion de durée de vie. Cette notion repose essentiellement sur le fait que la résistance à la fatigue d'une famille d'éléments et les efforts qu'ils supportent en vol sont l'objet de dispersions qui obéissent à des lois de répartition "gaussienne", donc elle ne peut prendre en compte les défauts de fabrication exceptionnellement non détectés par le contrôle, les configurations de vol anormales et les erreurs de maintenance. La durée de vie de l'élément est calculée à partir des enregistrements en vol effectués à l'intérieur du domaine de vol et à partir des limitations déduites des essais de laboratoire sur pièces conformes. Ces calculs sont tirés d'un nombre forcément limité d'essais de laboratoire et de mesures en vol ; les marges de sécurité dynamiques choisies sont telles qu'elles garantissent un risque de 10^{-6} avec un coefficient de confiance de 90 %. De plus si la solution de la durée de vie limitée avec un certain coefficient d'incertitude est correcte sur le plan de la sécurité, elle est peu satisfaisante sur le plan de la rentabilité car elle implique de jeter au terme de leur durée de vie "sûre" des pièces apparemment en excellent état. C'est pourquoi la tendance actuelle est d'obtenir une durée de vie infinie pour tous les éléments vitaux. Cette condition, contraignante au niveau de la conception et de la mise en série car elle suppose un juste dimensionnement des pièces et une connaissance exacte des efforts supportés en vol, demeure peu contraignante du point de vue de la répercussion en masse : les parties critiques d'un élément vital représentent une très faible part du volume total, le reste étant dicté essentiellement par des conditions de raideur du mouvement à transmettre, de volume à contenir, de surface à offrir, de continuité de forme à assurer etc ...

Des recherches effectuées sur les causes des accidents de nos hélicoptères en vol démontrent que 12 % des défaillances sont dues à des erreurs de maintenance et 8 % à des défauts de fabrication, les 80 % restants étant imputables à des fautes humaines. A cette notion de "durée de vie sûre", est donc venue se superposer depuis quelques années la notion de "Fail-Safe" dont les grands principes peuvent se définir comme suit : réduire au maximum, au niveau de la conception, les risques d'erreurs en maintenance et les conséquences d'une erreur de maintenance ; assurer la redondance des éléments ou circuits vitaux ce qui signifie qu'une fonction vitale doit toujours être au moins doublée, chaque élément simple étant capable d'assurer seul la fonction ; assurer une détection de crique qui élimine tout risque de perte de fonction. Appliqués brutalement ces principes seraient très lourds de conséquences et risqueraient même d'aller à l'encontre du but recherché, par la complexité générale qu'ils pourraient entraîner. L'adoption de techniques nouvelles comme celle des stratifiés ou des lamifiés et la recherche d'effets secondaires comme l'élévation du niveau vibratoire appareil provoquée par l'introduction de souplesse de commande consécutive à des criques d'éléments simples ou l'utilisation des liquides colorés de ressuage, permettent de satisfaire aux conditions de Fail-Safe sans répercussions prohibitives sur les masses. Dans ces conditions, il est possible d'afficher des répercussions inférieures à 20 % de la masse des éléments à redondance sans augmentation notable des encombrements correspondants. Le respect total des principes Fail-Safe sur l'ensemble moyen rotor principal amènerait dans ces conditions une augmentation de masse de 70 Kg pour un hélicoptère moyen de 7 tonnes d'où l'on déduit une augmentation de 3,5 % du prix spécifique opérationnel en mission transport de fret interne sur 160 Km donc 3,5 % moins rentabilisable. En mission transport de passagers sur 400 Km ce chiffre devient 6 %.

Eléments à potentiel

La philosophie sur la sécurité des éléments à potentiel, c'est-à-dire essentiellement les pignons et les roulements a suivi le même cheminement que celle sur la sécurité des éléments à durée de vie. Initialement, il y a la notion de potentiel qui impose le démontage pour révision à intervalles de temps strictement spécifiés d'éléments mécaniques apparemment en excellent état dont la plupart pourraient continuer à fonctionner encore longtemps ; de plus cette notion ne peut tenir compte qu'imparfaitement des défauts de fabrication échappant au contrôle et des erreurs de maintenance. C'est pourquoi aujourd'hui, la notion de potentiel tend à être remplacée par la notion de "dépose selon état" qui permet, moyennant la possibilité de détecter et diagnostiquer les détériorations, de connaître ou prévoir à tout instant l'état du matériel, l'importance des éventuelles détériorations, le moment optimal d'intervention. Ce changement de philosophie suppose que le matériel ait été étudié et justifié avec de nouveaux concepts tenant compte en particulier de la nature des détériorations présumées et de l'intégration des dispositifs d'examen et de contrôle.

En parallèle, la notion de Fail-Safe tend à s'introduire aujourd'hui sur les moyens de lubrification soit par introduction de lubrification de secours soit par redondance des circuits de lubrification, demain peut-être, sur les carters par détection des criques, sur les roulements par redondance. L'application des principes de redondance sur les pignons et les arbres amènerait une pénalisation de 30 Kg environ pour un hélicoptère moyen c'est-à-dire une chute de 1,5 % sur le prix spécifique opérationnel en mission transport de fret interne sur 150 Km.

LE PRIX DU CONFORT

La réduction du bruit

Le problème du bruit est relativement récent sur les hélicoptères et coïncide avec l'ouverture et le développement du marché civil. En fait le problème est double : d'une part, nuisance du bruit externe en liaison avec les conditions de pénétration urbaine, les sources sonores en cause se situant au niveau des moteurs et des rotors ; d'autre part, ambiance sonore interne influant sur le confort des passagers et de l'équipage, la source sonore se situant essentiellement au niveau de la boîte de transmission principale.

En ce qui concerne le bruit externe, un des objectifs à atteindre serait un niveau sonore de 90 PNDB (PNDB signifie : Perceived Noise décibel ; intéresse toutes fréquences occasionnant une gêne acoustique physiologique) dans un rayon de 150 mètres autour de l'appareil en vol stationnaire au niveau du sol. Cela signifie qu'il faudrait obtenir, pour les hélicoptères légers actuels une réduction de 10 à 15 PNDB du bruit moteur et une diminution du bruit rotorique de 6 à 7 PNDB. A l'heure actuelle les réalisations pratiques de lutte contre le bruit externe concernent essentiellement les moteurs. La vitesse périphérique du compresseur tronsonique génère un bruit intense à spectre très riche en harmoniques avec une fondamentale au moins égale à 6000 hertz. Les remèdes peuvent associer l'effet d'absorption par des matériaux à résonateurs et l'effet de masque par une géométrie d'entrée d'air adéquate. Les gains obtenus ainsi sont de l'ordre de 10 PNDB au prix d'une perte d'environ 2 à 3 % sur la puissance du moteur ce qui limite le domaine de performances de la machine notamment en altitude. Quant au bruit rotorique, l'action doit se porter en général d'abord sur le rotor arrière. Une diminution de 10 % de la vitesse périphérique des pales arrière permet des gains de 3 PNDB. S'il faut agir sur le rotor principal, un affinement du profil et une optimisation de la forme en plan de l'extrémité de la pale, permettront, en augmentant le Mach de divergence de traînée et en diluant le tourbillon marginal, d'obtenir des gains de 3 à 5 PNDB ; un gain supérieur, 10 à 15 PNDB nécessiterait une réduction de la vitesse périphérique des pales principales, c'est-à-dire en fait une conception nouvelle du rotor principal, et de toute manière un sacrifice important sur les performances.

En ce qui concerne le bruit interne, rappelons que 80 db SIL (db SIL signifie : Décibel Speech Interference Level ; intéresse les fréquences de la parole de 500 à 4.000 hertz) est une ambiance sonore qui nécessite d'élever la voix pour se parler à plus de 20 cm. Si l'on compare aux 70 db SIL des avions commerciaux et aux 85 db SIL des avions légers monomoteurs l'on déduit que l'objectif à atteindre est une ambiance sonore au plus égale à 20 db SIL. Cela signifie qu'il faut obtenir pour les hélicoptères légers une réduction de 10 à 15 db SIL, de 20 à 25 db SIL pour les hélicoptères moyens. Par les procédés actuels d'écran acoustique la perte de masse correspondante est de 1 à 1,5 % de la masse totale de la machine. Au niveau de la conception de la boîte de transmission, à partir d'une technologie appropriée des engrenages et d'un choix judicieux des fréquences propres des arbres et des qualités acoustiques des carters, cette perte de masse ne serait plus de 0,25 à 0,3 % de la masse totale, mais cette technique n'est pas maîtrisée aujourd'hui.

L'amélioration du niveau vibratoire

Le fuselage d'un hélicoptère vibre essentiellement sur la fréquence $b\Omega$ = nombre de pales du rotor principal x vitesse de rotation du rotor. Pour un hélicoptère donné cette fréquence est en général constante (aux variations exceptionnelles de régimes d'autorotation près) et suivant l'hélicoptère se situe entre 16 et 24 Hz. Pour faciliter les analyses et normaliser les appréciations de niveau vibratoire, il est nécessaire d'effectuer des mesures d'accélération vibratoires : une accélération dynamique verticale de 0,1 fois la pesanteur ou 0,1 g à 18 Hz est considérée comme "très bonne" ; à titre de référence, et bien que les exigences des nouveaux programmes américains soient nettement plus sévères, les normes admettent communément 0,15 g jusqu'à la vitesse de croisière, 0,2 g jusqu'à la vitesse maximale et 0,3 g dans la zone des vitesses de transition. En plus des termes d'accélération verticales le "niveau" vibratoire se compose des termes d'accélération transversales et longitudinales généralement d'amplitudes plus faibles qu'en vertical. Il comporte également l'ensemble des vibrations de parois, d'instruments de vol, de commandes qui constitue l'"ambiance" vibratoire.

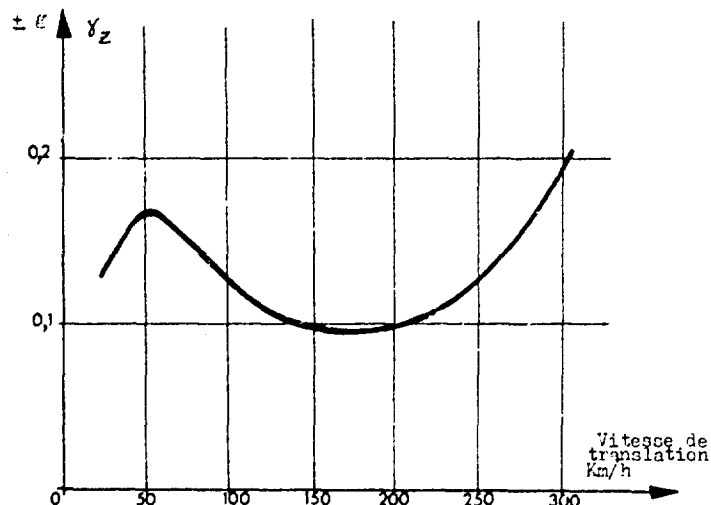


Figure 3 : Evolution du niveau vibratoire en fonction de la vitesse

La préoccupation d'assurer un excellent niveau vibratoire n'est pas nouvelle chez les hélicoptéristes. C'est durant la phase de mise au point que l'amélioration du niveau vibratoire est chère ; la part des crédits utilisés pour ce problème peut varier entre 10 et 50 % du total suivant l'ampleur du domaine de vol visé et la difficulté des compromis à satisfaire. Les moyens d'action se situent au niveau de la pale en minimisant les efforts aérodynamiques appliqués et leur transfert de la pale au moyeu, au niveau de la boîte par une suspension adéquate, c'est-à-dire en utilisant au mieux les efforts d'inertie de la boîte soit par des moyens passifs soit par des moyens actifs, au niveau du fuselage en optimisant les caractéristiques vibratoires de la structure. Ces travaux interviennent dès le début de la conception de la machine, d'autant plus efficacement que l'expérience est grande et les méthodes de calcul précises et fidèles, puis en laboratoire dès que les maquettes ou éléments échelle grandeur le permettent, enfin sur les bancs et le prototype en vol.

Jusqu'à des valeurs de niveau vibratoire d'environ 0,15 g, il est possible de dire que la qualité du niveau vibratoire est sans répercussion appréciable sur le prix de fabrication de la machine si les travaux de mise au point ont été menés dans cet esprit : il n'est finalement pas plus cher de fabriquer une pale accordée correctement entre deux fréquences excitatrices qu'une pale accordée sur une fréquence excitatrice. Au-dessous de ces valeurs et compte-tenu de l'état actuel de nos connaissances et de nos moyens il devient nécessaire de faire appel à des raffinements plus ou moins coûteux.

Conclusion sur le prix du confort

La conquête du marché civil pour hélicoptère, implique d'offrir, pour le passager des liaisons inter-aéroport, pour le Président Directeur Général en liaison "affaire" etc ... une présentation interne et externe, un confort, une ambiance qui soient comparables à ceux que le client est habitué à trouver sur les avions de ligne, dans le moyen courrier d'affaire, dans l'automobile de classe.

Pour donner à la cabine d'un hélicoptère léger, par exemple un "cinq places", un espace vital identique à celui d'une automobile type Rolls ou Mercedes en hauteur, en largeur et en profondeur, les modifications de structure coûtent, par rapport à l'hélicoptère actuel, une dizaine de kilogrammes, l'accroissement correspondant du maître couple de la cabine entraîne une augmentation de 8 à 10 % de la traînée de l'appareil.

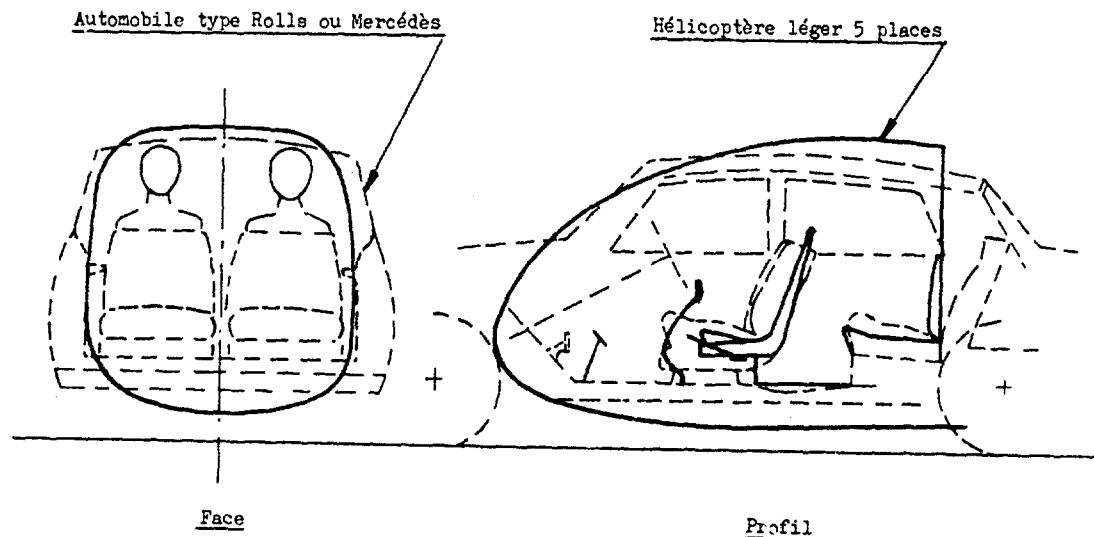


Figure 4 : Comparaison de l'aménagement interne d'un hélicoptère léger avec celui d'une automobile

Par ailleurs, le conditionnement nécessaire à l'obtention d'une bonne ambiance sonore et vibratoire, en accord avec les règles de l'esthétique et des commodités classiques de l'automobile ou de l'aviation VIP coûte une trentaine de kilogramme par les moyens traditionnels. Ces impératifs imposent un accroissement du prix spécifique opérationnel de l'ordre de 10 %. De même pour donner au cargo d'un hélicoptère moyen le niveau de confort exigé par le transport de passagers classique, il est nécessaire d'accepter un accroissement du prix spécifique opérationnel de l'ordre de 10 %.

APPORT DES TECHNOLOGIES NOUVELLES

Le chapitre précédent raisonnait sur des hélicoptères existants, c'est-à-dire conçus il y a quelques années et résultant d'un compromis entre l'état de la technologie, les notions de rentabilité et une extrapolation des besoins opérationnels de l'époque. Les nouveaux besoins en performances, confort et sécurité apparus depuis entraînent un certain "déséquilibre" entre l'appareil existant et l'appareil souhaité du fait que technologiquement, aucune amélioration ne peut être apportée sans répercussion néfaste de masse et de prix. En essayant de chiffrer ces améliorations, nous avons vu que chaque tentative à technologie constante se soldait par une augmentation du prix de revient spécifique et du prix spécifique opérationnel. La nécessité d'introduire de nouvelles technologies pour reculer les points d'optimisation et améliorer les rapports coûts/efficacité, est donc apparue clairement. (À noter, que l'introduction de technologies nouvelles se produit aussi souvent sous la pression des techniciens eux-mêmes). Dans le chapitre qui suit, nous nous efforcerons à l'aide de quelques exemples, de définir la politique de l'Aérospatiale en matière de nouvelles méthodes de calcul, de nouveaux concepts et de nouveaux matériaux.

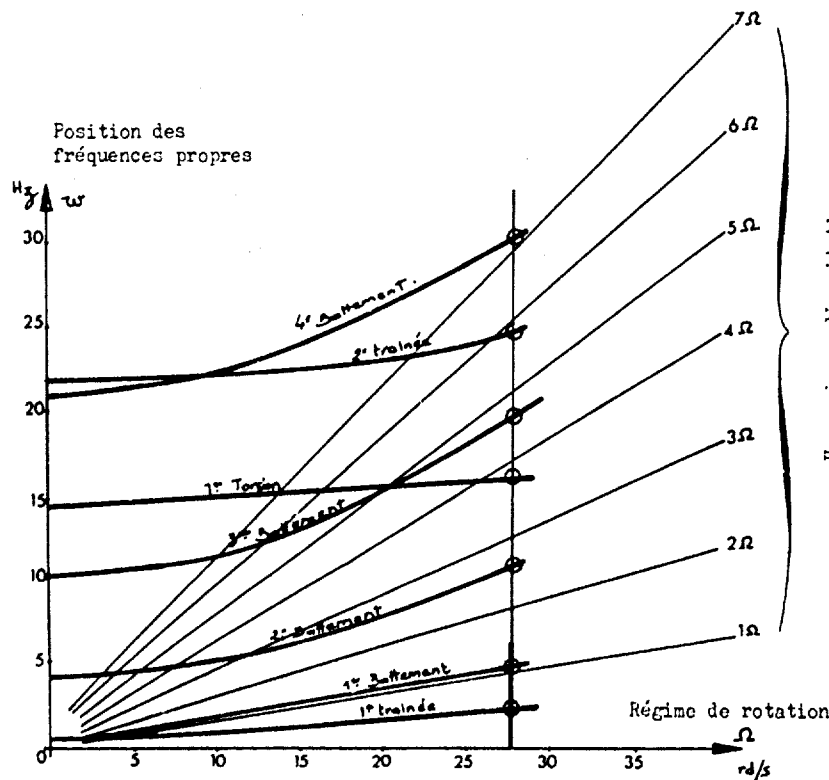
NOUVELLES METHODES DE CALCUL

Remarques préliminaires

Particulièrement dans les domaines de l'aérodynamique des rotors, des vibrations et de la tenue des éléments travaillant en fatigue, domaines fondamentaux dans la technique de l'hélicoptère, de nouvelles méthodes de calcul, apparues récemment, permettent une participation totale de l'ordinateur à chaque stade de la mise au point d'un hélicoptère : conception, études, essais de laboratoire et essais en vol. On obtient ainsi une meilleure analyse des conditions techniques à satisfaire, un choix plus sûr et plus précis des solutions techniques correspondantes, une compréhension plus rapide et plus complète des phénomènes rencontrés. Dans certains cas, même, c'est une dimension nouvelle qui est proposée, c'est-à-dire la possibilité d'effectuer des explorations par calcul qui sans ces nouvelles méthodes eussent été impossibles. Nous donnons ci-dessous trois exemples d'application de nouvelles méthodes de calcul.

Optimisation dynamique des pales

Du point de vue aérodynamique, la pale de l'hélicoptère peut être considérée comme une aile animée, par rapport à l'air, du mouvement de rotation autour de l'axe rotor et du mouvement de translation d'avancement de



l'hélicoptère. Les efforts aérodynamiques appliqués sur les pales d'un rotor d'hélicoptère comportent des termes périodiques importants parce que résultant de la combinaison de la vitesse de translation de l'appareil et de la vitesse de rotation du rotor. Ces termes sinusoïdaux des efforts aérodynamiques appliqués ont donc des fréquences multiples de la vitesse de rotation du rotor. Du point de vue dynamique, la pale est une poutre très longue et très flexible, qui en fonctionnement est tendue par la force centrifuge. Elle possède donc de très nombreux modes propres de battement, de trainée et de torsion, dont la position des fréquences propres par rapport aux harmoniques d'excitation détermine ou l'amplification ou l'atténuation des efforts appliqués sur la pale. Chaque pale transmet au moyeu rotor les efforts aérodynamiques qui lui sont appliqués au travers de sa "fonction de transfert".

Figure 5 : Diagramme de fréquences propres de pale

Le principe de l'Optimisation Dynamique est de trouver la loi des masses et des raideurs suivant l'envergure de la pale qui permettra de transmettre au moyeu le minimum d'effort pour l'ensemble des principaux harmoniques d'excitation de battement de trainés et de torsion et ce, à l'intérieur du corset imposé d'une part par l'aérodynamique en ce qui concerne le rayon du rotor, la corde, l'épaisseur du profil, le vrillage, etc ..., d'autre part par les conditions de résistance statique et en fatigue. Toutes ces conditions laissent peu de marge de manoeuvre dans les technologies traditionnelles de pale. L'apparition des roving verre-résine et carbone-résine ouvre des perspectives d'action très grandes de par les qualités mécaniques du matériau et de par leur souplesse d'emploi et permet de concilier un nombre plus importants d'impératifs. Dans la méthode d'Optimisation Dynamique des pales, l'ingénieur responsable recherche les valeurs optimales en dialogue direct avec l'ordinateur. Le calcul démarre à partir d'une configuration de base de la pale en masse et en raideur. Les conditions d'optimisation peuvent être la position des fréquences propres par rapport aux harmoniques d'excitation, ou mieux, peuvent être les transferts des efforts aérodynamiques appliqués. Par calcul linéarisé, on cherche, par exemple, les variations de fréquences propres obtenues pour des petites variations des fonctions de répartition de la pale. Il est prévu, dans le programme actuel, de modifier simultanément dix sections au maximum, sur un découpage total de trente sections dans un ordre arbitraire, cette action étant répétitive et cumulative. Lorsque les modifications ont un effet non linéaire, l'ordinateur le signale. Il est alors possible de stocker les fonctions de répartition et les fréquences propres de base et de calculer directement le nouveau diagramme de fréquences remplaçant le diagramme de départ.

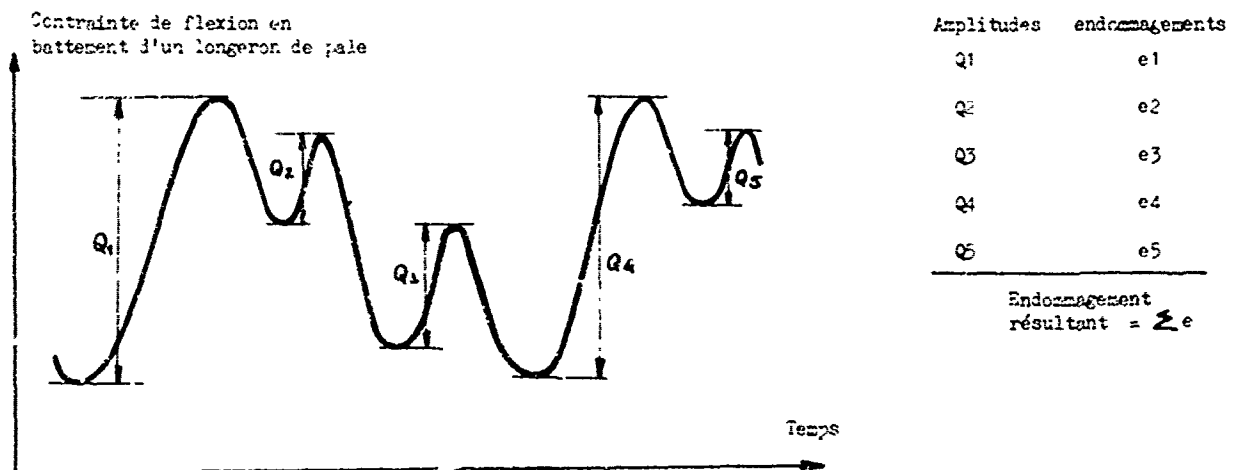
Cette méthode de calcul, à partir de la connaissance des efforts aérodynamiques appliqués et grâce à la souplesse d'action procurée par la technologie des roving verre et carbone-résine doit permettre d'optimiser les conditions de transfert des pales, c'est-à-dire obtenir un niveau vibratoire et un niveau de contraintes faibles dès le début de la mise au point des prototypes, c'est-à-dire réduire considérablement les délais et les coûts de mise au point.

Dépouillement automatique des contraintes

De par le fonctionnement dissymétrique et cyclique du rotor, les divers composants d'un hélicoptère sont soumis à des efforts sinusoïdaux et travaillent en fatigue. Il convient donc de calculer pour chaque élément vital la durée de vie en utilisation.

Les méthodes classiques de calcul d'endommagement en fatigue sont basées sur l'établissement d'un canevas de vol à partir des principales configurations de vol stabilisées ou évolutives imposées par la mission de la machine, puis la détermination pour chaque configuration de vol et pour chaque élément vital, de l'amplitude et de la fréquence de la contrainte dynamique supportée. Ces méthodes de calcul impliquent un dépouillement manuel des bandes d'enregistrement des efforts de vol donc une procédure très lourde et soumise aux erreurs humaines d'interprétation des phénomènes.

L'Aérospatiale a mis au point une méthode de dépouillement automatique des contraintes basée sur l'acquisition des amplitudes "crête à crête" successives et leur comparaison à des niveaux d'endommagement de référence pré-affichés. Les endommagements élémentaires, calculés à partir des courbes de Wohler déduites par essais de laboratoire, sont comptabilisés, suivant la loi des dommages cumulatifs de Miner, soit directement en vol par calculateur embarqué ou par transmission télémètre, soit après vol par restitution des enregistrements de contraintes sur bande magnétique.



Cette nouvelle méthode de dépouillement apporte au stade de la mise au point prototype la possibilité de connaître à chaque instant d'un vol la croissance de l'endommagement des principaux éléments travaillant, au stade du lancement en série, la possibilité d'effectuer des statistiques très complètes sur les efforts supportés dans les configurations de vol transitoire et d'une façon générale une évaluation beaucoup plus correcte des endommagements réellement supportés en vol c'est-à-dire un dimensionnement plus juste des éléments travaillant et une durée de vie plus précise.

On obtient donc dans tout le domaine des opérations de surveillance des endommagements en vol ou de calcul des durées de vie un gain de temps considérable (temps global divisé par 5) ainsi qu'une précision et une fiabilité accrues qui autorisent en supprimant les marges inutiles fruit de l'ignorance soit une réduction de la masse sur les éléments vitaux, soit un accroissement de leur durée de vie.

Analyse Structurale

A la base des calculs de résistance et de vibrations se situe toujours une schématisation de l'ensemble à calculer. Cette schématisation est d'autant plus délicate ou plus imprécise que l'ensemble est de forme plus complexe. Pour le fuselage d'un hélicoptère actuel, moyen ou lourd, l'idéalisation par les moyens classiques devient illusoire.

Fondée sur les méthodes de calcul dites par "éléments finis", l'Analyse Structurale est employée dans deux domaines : l'analyse des contraintes supportées par les éléments structuraux ou les pièces de forme complexe

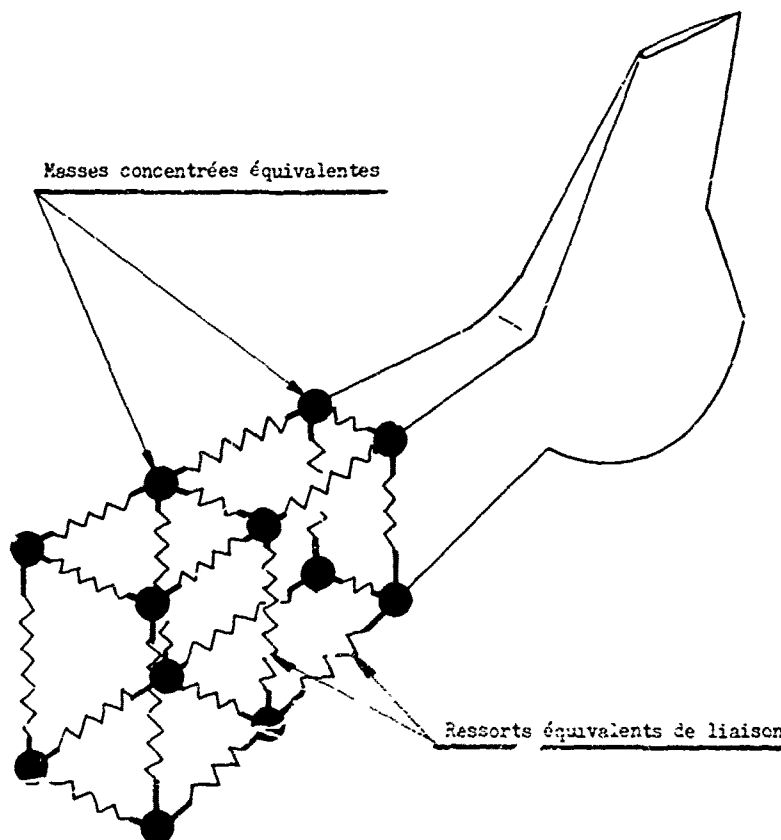


Figure 6 : Schématisation d'une structure par un ensemble de masses concentrées - ressorts équivalents.

et l'étude des vibrations des structures. Elle consiste essentiellement à substituer à la structure réelle un assemblage équivalent constitué, soit d'éléments travaillant simples du type barre de traction, poutre de flexion, plaque triangulaire etc ... pour l'analyse de résistance, soit d'éléments simples purement massiques reliés entre eux par des éléments simples purement élastiques pour l'analyse de vibrations. La finesse du découpage est fonction de la complexité de la structure, de la précision recherchée et de la capacité des ordinateurs utilisés.

Appliquée aux pièces mécaniques de forme complexe la méthode de l'Analyse Structurale permet d'obtenir une approximation excellente de la distribution réelle des contraintes, d'où une détermination précise au stade de la conception, un volume d'essais réduit au stade de la mesure des contraintes et des justifications. Appliquée aux structures elle permet, par la résolution des équations de dynamique, une détermination dès le stade du projet des modes et des fréquences propres des fuselages et de leur participation dans les couplages avec les rotors, ce qui réduit considérablement le volume des essais de vibrations sur prototype, facilite l'interprétation des résultats des essais de laboratoire et en vol, oriente plus efficacement les modifications de mise au point. Le gain obtenu sur le temps de mise au point vibratoire des fuselages est de l'ordre de 50 %.

NOUVEAUX CONCEPTS

Remarques préliminaires

Pour tenter de réduire à la fois les prix de revient industriels et les coûts d'exploitation, l'AEROSPATIALE a depuis quelques années porté ses efforts dans la recherche de concepts originaux, mieux adaptés à l'utilisation rationnelle des appareils.

Moyeu rotor arrière type fenestron

Parmi les solutions recherchées pour réduire les coûts tout en augmentant la fiabilité ou la sécurité, une des plus intéressantes est l'adoption d'un rotor de queue de type fenestron en remplacement du rotor arrière classique. Ce rotor, fonctionnant à l'intérieur d'une veine carénée, est d'une conception particulièrement simple, puisqu'il utilise des pales matricées en alliage léger à durée de vie infinie, interchangeables sans réglage et articulées en incidence sur des manchons en plastique ne nécessitant aucun entretien. Sa remarquable fiabilité, alliée à une maintenance pratiquement nulle, en font un ensemble très économique. Ainsi, la répercussion sur les coûts d'entretien représente une diminution de 50 à 75 % par rapport aux coûts d'un rotor arrière classique.

De par sa conception entièrement carénée dans le prolongement de la poutre de queue, ce rotor présente une grande sécurité d'emploi du fait qu'il est entièrement protégé lors de heurts éventuels avec des arbres, des objets au sol ou des appareils en stationnement et que le personnel d'entretien au sol se trouve lui aussi protégé. Cet avantage se traduit : par une économie au niveau des accidents du personnel et des pièces de rechange appareil. Par ailleurs, la conception originale de ce dispositif permet de supprimer les équipements de dégivrage, qui sont habituellement lourds et onéreux sur les autres types de rotor arrière. D'autres avantages sont obtenus sur le plan des performances : consommation pratiquement nulle de puissance en vol de translation ; meilleure finesse générale de l'appareil ; suppression des phénomènes d'instabilité de pales ; poursuite possible du vol en translation en cas d'avarie du rotor arrière grâce à la surface de la dérive qui agit en anti-couple.

Conception modulaire

Une autre source d'économie pour l'utilisateur réside dans la conception modulaire des ensembles mécaniques. Cette conception consiste à scinder un ensemble considéré en un certain nombre de modules différents, aisément démontables, chacun d'eux pouvant avoir un potentiel particulier. Elle permet au mécanicien d'intervenir rapidement au niveau du module et de le remplacer par un autre module en état de marche, sans immobilisation notable de l'appareil. Depuis quelques temps, les principaux motoristes de l'aviation commerciale réalisent leurs moteurs suivant ces concepts et permettent aux compagnies aériennes d'économiser d'importantes sommes d'argent, d'une part en évitant à leurs appareils de longues immobilisations au sol, d'autre part en réduisant l'importance du stock de rechange qui se limite ainsi à quelques modules et non plus à des moteurs entiers. A noter que le temps d'échange d'un module ne doit pas dépasser 2 heures.

Dans le domaine des mécaniques hélicoptères, on compte encore peu d'application de ce concept. Pour sa part, l'AÉROSPATIALE entend l'appliquer à certaines réalisations, en particulier aux boîtes de transmission principales et aux moyeux rotor qui sont des ensembles complexes et onéreux. Ainsi, en ayant la possibilité d'intervenir lui-même sur certains modules, l'utilisateur accroîtra la disponibilité de ses appareils et évitera des retours en usine pour réparation qui sont toujours longs et chers.

La conception modulaire est une méthode rationnelle d'utilisation du matériel. Elle tend à supplanter de plus en plus les traditionnelles déposes d'ensembles mécaniques programmées ou non qui constituaient jusqu'à présent un lourd handicap pour les utilisateurs civils et militaires dans le domaine de la rentabilité et de la disponibilité.

Notion de "dépose selon état"

La notion de "dépose selon état" tend à remplacer la notion de "potentiel". Un matériel, utilisé normalement suivant les besoins pour lesquels il a été créé, peut se détériorer pour différentes raisons : usure, corrosion, fatigue, vieillissement, milieu défavorable, mauvais entretien, etc ... Tout particulièrement dans le domaine des hélicoptères, et surtout pour les ensembles mécaniques principaux, il importe de savoir jusqu'à quel moment on peut continuer à faire voler un élément qui commence à se détériorer sans que la sécurité soit compromise. Dans la méthode actuelle, faute de pouvoir connaître avec exactitude ce seuil, on définit des périodes de fonctionnement, appelées "potentiels", à l'issue desquelles on dépose systématiquement les mécaniques pour les envoyer en révision. Ces potentiels sont définis par le calcul, par des essais de fatigue et d'endurance, par expérience obtenue sur des matériels similaires. En général le potentiel de départ est de l'ordre de 300 h et on le fait évoluer ensuite par paliers successifs jusqu'à des valeurs de 1500 à 2000 h en fonction de l'état des pièces examinées en révision. Cette méthode est longue et pénalisante financièrement pour l'utilisateur car elle impose de déposer à périodicité fixe ses ensembles mécaniques sans qu'il puisse connaître avec exactitude leur état réel et l'oblige à constituer des stocks importants de rechange, surtout lorsque les potentiels sont bas.

Par ailleurs, cette méthode entraîne souvent des retours en usine injustifiés du fait que le matériel aurait pu encore fonctionner de façon satisfaisante de nombreuses heures.

Du point de vue financier, tant en ce qui concerne la disponibilité des appareils que les coûts directs d'exploitation, l'idéal pour l'utilisateur est donc de pouvoir connaître à tout instant l'état dans lequel se trouve le matériel, l'importance des éventuelles détériorations et le moment qu'il convient de choisir pour procéder à sa dépose tout en sauvegardant la sécurité. On arrive ainsi à la notion de "dépose selon état" qui doit prendre dans les prochaines années une grande importance. L'application de cette nouvelle méthode implique au préalable que le matériel soit étudié, calculé, justifié et utilisé sous certaines conditions, en particulier :

- tous les composants doivent être conçus pour des durées de vie infinies
- le potentiel probable calculé doit être d'au moins 2000 h
- le matériel doit être étudié pour que les détériorations puissent être détectées, diagnostiquées et suivies sans ambiguïté.

La détection et le diagnostic des détériorations ne peuvent être obtenus qu'avec des moyens adaptés à la nature de l'ensemble. Pour les boîtes de transmission, on peut envisager par exemple : l'analyse des particules du bouchon magnétique et du filtre à huile, l'analyse spectrale des huiles, la détection de la température au niveau de certains composants, l'analyse des bruits, les détecteurs de vibrations, la mesure de certains jeux, l'endoscopie, etc ... Ces moyens risquent d'être onéreux ; il convient donc de les employer avec prudence et de faire à chaque fois un bilan coût/efficacité. Les compagnies aériennes utilisent couramment cette méthode sur les moteurs de nouvelles générations et il est certain que les investissements qui lui sont consacrés sont largement compensés par les gains apportés sur l'utilisation du matériel.

Pour sa part, l'AEROSPATIALE pense qu'il est possible d'appliquer efficacement cette notion de "dépose selon état", tout au moins en ce qui concerne les ensembles mécaniques principaux. Elle s'y emploie actuellement dans l'étude des nouveaux matériels.

Durée de vie infinie des éléments vitaux

Les durées de vie des éléments vitaux d'un aéronef sont calculées à partir d'un spectre de vol, c'est-à-dire à partir des efforts réels supportés par la structure ou par les ensembles mécaniques au cours des différentes phases de vol. Jusqu'à présent, au cours de l'étude d'un nouvel appareil il était peu aisé de connaître à priori et avec précision les différents spectres de vol qu'il était susceptible de rencontrer au cours de son existence. Cette connaissance imprécise avait pour conséquence de limiter certaines durées de vie à des valeurs relativement basses et d'entraîner des répercussions non négligeables sur les coûts d'exploitation. Il est intéressant de noter que dans le calcul des coûts directs d'exploitation le poste "pièces à durée de vie" entre pour environ 10 %.

La solution pour diminuer ces coûts d'exploitation consiste donc à supprimer ce poste, c'est-à-dire à donner des durées de vie infinies aux pièces vitales. Aujourd'hui, une meilleure connaissance des efforts supportés en vol (progrès des méthodes de calcul théorique, progrès des méthodes de mesures et des méthodes de traitement de l'information) permet dès le stade de la conception d'un appareil une meilleure définition des pièces vitales travaillant à la fatigue. Cette meilleure connaissance des efforts, associée aux progrès réalisés dans la définition statistique des spectres de vol des appareils permet l'obtention de durées de vie très grandes, voire illimitées, dès le début de la mise en série.

Utilisation des équipements de grande série

En construction aéronautique, l'emploi d'équipements ou de matériaux dits de "qualité aéronautique" est, depuis longtemps de rigueur. L'homologation de ces équipements et de ces matériaux est faite sous la responsabilité des Services Officiels de l'Aéronautique. Certes pour assurer la sécurité d'une mission, il est normal d'utiliser des équipements sûrs, performants et fiables. Il serait impensable de monter sur des appareils des éléments qui n'auraient pas subi les tests et les contrôles correspondant à un certain niveau de qualité. Cependant le nombre d'équipements atteignant des prix exorbitants du fait de leur "étiquette" aéronautique et de leur fabrication en petite série. Exception faite des équipements très sophistiqués ayant un haut degré de spécialisation, il est certain qu'une quantité importante d'équipements pourrait être prise dans le secteur des grandes séries, l'automobile par exemple.

Nous pensons que des économies intéressantes pourraient être réalisées dans ce domaine car le poste "équipements" représente de 10 à 15 % des coûts directs d'exploitation et de 15 à 25 % du prix de revient d'un appareil de moyen tonnage et les écarts de prix entre le secteur automobile et le secteur avion sont considérables : la règle générale est un rapport de 1 à 5 ; on trouve quelquefois un rapport de 1 à 10 et même 1 à 20, il convient donc de changer la philosophie qui préside aux choix de ces équipements. En particulier, il convient de créer des catégories correspondant à la fonction remplie par l'organe et au rôle qu'il joue par rapport à la sécurité de vol. Plusieurs catégories pourraient être créées pour lesquelles la perte ou la détérioration de l'équipement : entraînerait la destruction de l'appareil ; occasionnerait l'arrêt ou l'écourtement de la mission ; n'aurait pas de répercussion sur la mission.

Ainsi les équipements ne mettant pas directement en cause la sécurité de vol, pourraient faire l'objet d'une homologation restreinte se limitant à quelques essais de bon fonctionnement sur banc d'essai ou sur appareil et n'entraîneraient pas d'importantes plus-values sur le prix de vente. De même, l'homologation des matériaux entrant dans la fabrication de certains équipements ou aménagements pourrait suivre la même procédure. Il faut néanmoins remarquer que beaucoup d'organes du secteur automobile sont en général plus lourds que ceux du secteur aviation, (mais aussi plus robustes) ; quand on sait l'importance que l'on attache au facteur poids dans l'aéronautique, l'on déduit qu'il sera nécessaire de faire un bilan Prix/Masse/Performances sérieux avant de procéder aux choix finals.

NOUVEAUX MATERIAUX

Laminés élastomère-métal

Sur les rotors principaux d'hélicoptères classique, les pales possèdent trois degrés de liberté au niveau de leur fixation sur le moyeu matérialisés par trois articulations orthogonales, autour desquelles elles oscillent périodiquement. Les articulations sont réalisées à partir de roulements à billes ou à aiguilles.

Le remplacement de ces éléments par des paliers lamifiés élastomère-métal, où le mouvement de roulement est remplacé par une déformation élastique, permet d'envisager un gain important sur le prix de revient :

- par une diminution du nombre de pièces (remplacement de 1 ou plusieurs roulements avec leurs joints d'étanchéité, leur système de lubrification),
- par une simplification des pièces (diminution du nombre d'usinages et de leur précision).

En allant plus avant, on peut concevoir des moyeux utilisant des paliers lamifiés de manière originale pour leurs qualités intrinsèques et non pas en remplacement d'éléments classiques. Le prix de revient peut alors être réduit beaucoup plus considérablement. Toutes ces simplifications entraînent également un gain de masse qui peut être très important. Au stade de l'exploitation, l'utilisation des paliers lamifiés entraîne une réduction des coûts spectaculaire. En effet, ces éléments n'exigent aucun entretien (au un ingrédient consommable, pas de problème de fuite). Par ailleurs, ils sont conçus pour avoir une durée de vie élevée et ne présentent pas de risque de rupture brutale. Un simple examen visuel est suffisant et peut se faire sans aucun démontage. En même temps que des économies sont ainsi réalisées sur l'entretien, la disponibilité de la machine augmente corrélativement, ces deux aspects entraînant une réduction considérable du coût d'exploitation.

Matériaux visco-élastiques

Les moyeux rotor principaux classiques sont équipés en général d'un amortisseur hydraulique pour contrôler le mouvement de traînée des pales. Sur les moyeux plus modernes, semi-articulés ou rigides, il est nécessaire d'introduire dans le plan de traînée une raideur parallèle avec l'amortissement, avec un rapport amortissement/raideur bien déterminé.

Par analogie avec le problème général des suspensions, on peut concevoir un système à fonctions séparées : ressort monté en parallèle sur amortisseur hydraulique. Une telle conception à partir d'éléments classiques, conduit à un système complexe lourd et encombrant. Les matériaux visco-élastiques ont apporté une solution élégante à ce problème, en permettant d'assurer les deux fonctions à partir d'un seul élément déformable. La valeur du rapport amortissement/raideur et la nécessité d'obtenir des caractéristiques sensiblement constantes dans la plage des températures d'utilisation, ont imposé le choix d'élastomère silicone.

Beaucoup plus simple que les éléments qu'il remplace, cet amortisseur visco-élastique permet un gain appréciable sur le prix de revient. De plus, on peut envisager d'incorporer cet élément dans un ensemble moyeu plus compact de conception plus simple et plus économique, perspective pratiquement impossible avec les systèmes classiques. Mais ici encore, c'est dans le domaine de l'utilisation de la machine que la réduction des coûts est la plus sensible : conçu pour avoir une durée de vie quasiment infinie, l'amortisseur visco-élastique ne nécessite aucune intervention, aucun entretien, un simple examen visuel est suffisant.

Les stratifiés "Verre-Résine" ou "Carbone-Résine"

Nous venons de dire que la simplification des rotors d'hélicoptère passe par la suppression des articulations classiques du moyeu montées généralement sur des roulements et leur remplacement par des éléments élastiques soit paliers lamifiés ou autolubrifiants soit éléments déformables de grand allongement en envergure : les matériaux métalliques usuels sont délicats à mettre en œuvre dans ce cas (contraintes élevées dues à la rigidité importante et sensibilité à l'entaille). L'utilisation des matériaux composites du type fibres de verre noyées dans une matrice en résine époxy a permis d'apporter des solutions légères et performantes du point de vue résistance à la fatigue de moyeux. Ont été réalisés suivant ce principe d'une part des moyeux semi-rigides à bras déformables d'autre part des pales souples en battement applicables à des moyeux rigides ou semi-articulés. Les stratifiés "verre-résine" ou "carbone-résine" allient en effet, souplesse (ou raideur) à remarquable résistance à la fatigue.

La fibre de verre est le matériau le mieux placé du point de vue du rapport résistance à la fatigue, module d'élasticité, pour la réalisation de structures travaillantes déformables : c'est ce qui explique que l'on ait obtenu sur les pales "Aérospatiale" de ce type une durée de vie infinie.

Le stratifié présente en outre les avantages suivants : orientation des fibres suivant les contraintes ; dégradation extrêmement lente progressive et apparente du matériau ; aucune sensibilité à l'entaille.

La fibre de carbone est utilisée de préférence lorsque l'on recherche une raideur importante pour une masse faible et des contraintes élevées. On l'emploiera pour raidir en traînée les pales principales par exemple ou pour réaliser des arbres de transmissions.

Dependant il faut savoir que des difficultés entravent encore le développement de ces matériaux, ce sont : d'une part leur prix, surtout pour le carbone, la fibre de verre étant relativement plus vulgarisée à l'heure actuelle, d'autre part leur mise en œuvre artisanale, un gros effort devant être fait sur ce point pour aboutir à des prix et cadences acceptables en série.

Les téflons armés

Le téflon est l'un des matériaux proposés par la chimie organique dont le coefficient de frottement est le plus faible, ce qui le prédispose à remplacer les roulements dans certaines conditions d'emploi : avec tous les avantages de gain de masse, et d'économie de maintenance que cela implique. Mais sa résistance à l'abrasement est faible. Deux procédés sont proposés pour lui donner les caractéristiques mécaniques souhaitées : d'une part, son utilisation sous forme de film très mince (quelques microns) ; mais cette technologie est relativement peu

usitée car elle est d'une application délicate, le revêtement est fragile, l'état de surface doit être très soigné pour appliquer le traitement ; d'autre part son utilisation sous forme de téflon armé c'est-à-dire sous forme d'une couche de frottement mince (5/100 à 1/10 mm) en matériau composite constitué d'une poudre, de fibres ou de tissus noyés dans une matrice en téflon.

Ces coussinets autolubrifiants peuvent être appliqués aux articulations de moyeux à faible débattement et lorsque les efforts de frottement ne présentent pas de répercussion sur le pilotage ou le comportement dynamique. C'est le cas des articulations de traînée. En ce qui concerne leur application aux articulations de battement ou d'incidence il convient d'être vigilant car d'une part les efforts de frottement sur un palier sec sont environ dix fois plus importants que sur un roulement à billes équivalent, ce qui intervient directement sur les efforts de commande ; d'autre part les mouvements sont plus importants et peuvent entraîner des échauffements parasites. De plus les paliers en téflon armé connaissent un "tassement" de la couche de frottement dans les premières heures d'utilisation, ce qui peut amener des jeux parasites.

CONCLUSION

Nous nous sommes efforcés de dégager quelques idées directrices sur le problème de la rentabilité des hélicoptères. Les limites de cette étude ont été imposées par :

- la nécessité de raisonner par rapport aux missions actuelles de l'hélicoptère,
- l'importance à priori du confort et de la sécurité dictée par les grandes tendances actuelles sans chiffrage précis de leur rentabilité propre,
- le raisonnement basé sur l'hélicoptère en lui-même sans référence à la rentabilité des autres moyens de transport concurrents éventuels.

Notre but était uniquement, dans un premier temps d'estimer le coût et les conséquences des moyens à mettre en oeuvre à partir de la notion actuelle de l'hélicoptère pour répondre aux besoins et aux nécessités qui se font jour, dans un second temps d'apprécier l'efficacité des nouveaux outils technologiques dont l'hélicoptère pourra disposer dans les années qui viennent.

Nous pensons pouvoir affirmer que l'hélicoptère entre dans une ère où il saura démontrer son efficacité et sa rentabilité.

DESIGN OPTIMIZATION OF THE VAK 191 B AND ITS EVALUATION BASED ON RESULTS
FROM THE HARDWARE REALISATION AND TEST DATA

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SUMMARY

The design optimization procedure for the fighter-type aeroplane VAK 191 B is being described and necessary reiterations are discussed. Based on hardware realisation and data derived from ground and flight tests, the procedure of design optimization is being evaluated by comparison of preliminary versus final design and test data.

RESUME

Le procede d'optimisation du projet pour l'avion de combat de type VAK 191 B sera decrit et les etapes d'iterations necessaires discutees. Le procede en projet utilise sera juge sur la base de la realisation du projet et par comparaison de donnees provisoires et definitives

1 INTRODUCTION

This paper deals with design approaches and procedures used for the development of the VSTOL fighter type Experimental Aircraft VAK 191 B (Fig. 1) and brings preliminary data in relation to final data, proven through hardware and test data. In various publications including AGARD papers (1, 2, 3, 4, 5, 6, 7) extensive technical details of the VAK 191 B have been presented, and therefore only a very brief description of the aircraft program will be given first for better understanding. The origin of the VAK 191 B dates back to 1962 when NATO issued first military requirements for a single seat VSTOL tactical close-support and reconnaissance fighter.

Due to a change in defence doctrine (1967) from massive retaliation to flexible response, the original prototype development program, which started in 1965, was changed to experimental, with 3 aircraft to be built.

From Sept. 71 to early 1972 all three planes have accomplished first hovering flights in Bremen. Flight testing was then continued from VFW Fokker flight test facilities on the German Air Force Test Center Manching.

Due to lack of funding by the German Government, VFW-Fokker had to stop the program in December 1972.

2. REQUIREMENTS

Key-points of the military requirements were.

- Type of aircraft
 - VSTOL strike/recce aircraft
- Basic mission
 - T_{VO} in S.L., hot day, 1250 lb payload
 - low level flight with .92 M for out- and inbound flight
 - radius of action 180 n.m.
 - 5 min combat in target area
 - 10% of internal fuel remaining after vertical landing
- additional tasks
 - armed recce and tactical recce missions
 - close support missions

– Special operational requirements and design features

- operation from dispersed semi-prepared sites
- independence from ground aids for engine starting and system check
- operation at day and night under limited bad weather conditions
- adequate manoeuvrability for CAS/strike mission
- internal stores bay
- 15 min. turn around time
- STOL capability
- A.U.W. 7 to 9 tons depending on mission

3. DESIGN PROCESS

When comparing the process of designing a VSTOL fighter for a given mission (Fig. 2) with that of designing a conventional type aircraft, it has to be considered that the propulsion system has to provide lift-off forces and additional power to propel the aircraft into the aerodynamic flight regime and decelerate it back from there to hover flight. In addition the propulsion system has to provide energy to generate control forces and this altogether should be done with minimum weight, minimum fuel consumption and minimum power loss which may result from hot gas recirculation, ground suction and/or jet induced down wash.

Therefore an optimum engine selection and engine/airframe integration with minimum negative interference effects becomes problem No. 1.

Mission analysis then in parallel leads to preliminary aerodynamic design, propulsion system evaluation and selection, subsystem lay-out studies and to the first approach to the aircraft configuration.

This way the VAK 191 B design process went as well.

3.1 Aerodynamic Design

3.1.1 Parametric studies

The aerodynamic design started with parametric studies of the wing borne flight regime since in a VSTOL strike aircraft wing size and configuration tend to be dictated by manoeuvre, cruise and transition requirements. Emphasis was therefore put on:

- low level high subsonic speed
- manoeuvre performance (clean configuration)
- transition speed range
- external stores loading capability, considering minimum wing span and influence on aircraft c.g.

Various trade-off studies in combination with point design studies were carried out, taking into account statistical data as well. Bearing in mind that the aircraft had to be designed for a high speed low level mission, the results of this approach were:

- wing loading at design weight $\sim 500 \text{ kg/m}^2 = 102 \text{ lb/ft}^2$
- max wing loading for transition $\sim 600 \text{ kg/m}^2 = 122 \text{ lb/ft}^2$
- low aspect ratio
- swept wing, considering a wing thickness between 5 and 7%, later on fixed at 5% at wing root, 6% at wing tip
- no complex high lift device to be considered.

3.1.2 Ride Qualities

Due to the fact that long duration high speed low level flying reduces pilot's efficiency considerably, ride quality investigations were initiated. A relevant gust spectrum was derived from statistical information. Then various wing configurations were evaluated from the angle of finding a wing shape which would provide for ride qualities with the least detrimental effect on pilot's efficiency upon arrival in the target area. The result of this study led to further updating of aerodynamic data, such as

- aspect ratio between 2.5 and 3.0
- leading edge sweep in the order of 40°

The compatibility of wing shape and external loading capability was then checked again.

3.2 Predominant VSTOL Features

Although a wide spectrum of design problems has to be considered and to be solved, only the predominant ones can be discussed here in more detail, such as:

- propulsion system selection
- control system lay-out
- undercarriage arrangement

3.2.1 Propulsion System

Various power plant concepts have been investigated. The finally contemplated four concepts (Fig. 3) have been evaluated in more detail in order to find the minimum weight aircraft for the given mission and so meet also other important aircraft design criteria, such as

- performance and growth flexibility,
- weight and cost sensitivity,
- pilot's safety,
- internal loading capability, and
- low overall systems complexity

in a suitable manner.

The four different approaches to power plant selection were:

- lift/cruise concept
- lift plus lift/cruise concept with symmetrical lift engine arrangement
- lift plus lift/cruise concept with asymmetrical lift engine arrangement
- direct lift concept (cruise engine plus lift engines).

Cruise engine cycle investigation, carried out in parallel to power plant integration studies, had already shown advantages in terms of thrust/weight and SFC versus Mach number for by-pass engines with a by-pass ratio of about 1.

For better comparison, A/C point design have then been worked out and a mission take-off weight for each configuration was estimated, using first step project weight estimation methods, where also the differences in subsystem weight have been considered to a practicable extent.

The resultant aircraft weight breakdown in to percentage of take-off weight is shown on the left in figure 4, and on the right the total aircraft weight for a given mission is plotted against percentage of lift engine thrust. A remarkable advantage was found for the lift plus lift/cruise configuration, with a thrust saving of about 50 to 50 percent between lift/cruise and lift engines.

Special attention was then paid to engine/airframe interference. In jet borne flight, while the aircraft is picking up forward speed, inlet momentum as well as jets and their induced down wash may create forces and moments which can be in the same order of magnitude or even higher than the available aerodynamic forces. Particular attention has here to be paid to sideslip in transition, and the designer, to minimize secondary forces and moments, has to configure the aircraft accordingly.

Figure 5 shows the specific characteristics of jet induced ground effects for the before mentioned power plant concepts.

This graph presents qualitative tendencies which have been derived from a variety of model test data and which have been used for the power plant selection. The graph shows an advantage to be expected from the symmetrical lift plus lift/cruise concept.

Crew safety at engine failure in low level hovering was investigated by means of simulation, using expected engine response and control system characteristics. With the different power plant concepts, the changes of aircraft attitude after engine failure showed dangerous pitch attitudes with the asymmetrical lift plus lift/cruise configuration, whereas symmetrical lift plus lift/cruise and direct lift configurations, due to smaller disturbing moments could be kept within acceptable limits for safe pilot escape.

The result for the single lift/cruise engine configuration looked different because there is no control power left after engine failure, but the thrust vector is practically in line with the aircraft c.g. and thus the disturbing moment created by engine failure is relatively small. So when the aircraft is in level attitude at engine failure, but only then, the pilot should have enough time to bail out. A summary of all these trade-off studies is shown in Fig. 6 which led to the selection of the symmetrical lift plus lift/cruise power plant concept. The preliminary design of the VAK 191 B derived from this first approach is shown in Fig. 7.

3.2.2 Flight Control System

The military operational requirements and the selected aircraft configuration led to basic design criteria for the flight control system, which are given in Fig. 8. Following these criteria and considering pilot work load and aircraft handling which were simulated for various flight control concepts, it was concluded that an electro-hydraulic system with a minimum of mechanical complexity offers dominant advantages over a mechanical system. Fig. 9 shows a scheme of the pitch channel as installed in the three experimental aircrafts.

The system is of redundant electro-hydraulic design, where pilot commands are picked off as analogue voltages from potentiometers and fed, via the triplex command and stability augmentation system (CSAS), into the electro-hydraulic duplex servo actuators.

The CSAS provides attitude control (pitch, roll) and rate control (yaw) in jet borne flight and reverts free of transients to damper mode in wing borne flight. The switch over from one mode to the other is performed either automatically as a function of speed or the pilot can operate manually. Reliability is achieved by triplicating the electrical part of the flight control system – with mutual monitoring and by duplicating the electro-hydraulic servo-actuators – with self-monitoring. After two failures in the system a mechanical back-up system is automatically engaged. Stick forces in all three axes are controlled by preloaded springs. Trim in pitch and roll is performed by shifting the springs; in yaw a voltage corresponding to the trim signal is added to the CSAS output. Flap position is also controlled electrically with the possibility to feed CSAS signals into the flap servo.

The flight control system is supplied by redundant hydraulic and electrical power systems.

The control moments in jet borne flight are generated by compressed air, which is tapped from all engines and expelled via two independent systems through control nozzles at the front and rear fuselage and at the wing tips. A scheme of the system is given in fig. 19. For high demand pitch manoeuvres lift engine thrust is modulated.

3.2.2 Landing Gear

The landing gear design, a tandem type main gear with supporting wing tip mounted outriggers (Fig. 17), was mainly dictated by the type of power plant installation and the large internal bay.

As an alternative a conventional tricycle layout has been investigated but finally was found not suitable for the following reasons:

- housing the main gear units in the fuselage was impracticable because of interference with main engine jets as well as with underwing stores. The necessary fuselage cross section would have been increased;
- stowing the gear units and low pressure tyres in wing nacelles was inconvenient due to unacceptable drag penalty and interference with external stores and wing trailing edge controls.

The undercarriage has been designed to cater for the complete range of motion at touch down, ranging from purely vertical landings with drift in all directions on semiprepared grounds to emergency conventional landings on concrete runways. Little information with regard to undercarriage loads was available and therefore detailed simulations had to be worked out to establish the undercarriage maximum load cases. In order to enable aircraft operation from semiprepared to unprepared grounds, a C.B.R. of 6 and a number of 50 passes were chosen for tyre selection.

The energy absorbing portions of the legs in combination with the remaining tyre deflections under static conditions were to be designed in such a way that local bumps of up to 15 cm which may occur in unprepared fields could be overrun without difficulty while taxiing.

For aircraft manoeuvring on the ground a nose wheel steering is provided which is controlled by the rudder pedals. The steering angle is normally limited to $\pm 17^\circ$, but can be increased to $\pm 45^\circ$ for taxiing.

For the best braking efficiency antiskid brakes have been installed in both, forward and aft wheels.

4. RESEARCH AND TEST PROGRAM

With respect to the advanced technology required in the VAK 191 B development, a rather extensive research and test program was carried out to back up aircraft aerodynamic and structural design and engine airframe integration. This program served for optimization and validation of the overall design and some of the results of course led to a number of design iterations.

The test program included

- 7850 hours wind tunnel tests in subsonic and transonic regime;
- 2000 hours testing of secondary aerodynamics such as ground and jet induced effects;
- 2350 hours subsystem tests with regard to function and reliability by use of special ground test rigs;
- 995 hours soft and hardware simulations with and without pilot by use of a fixed base simulator with hybrid computation;
- 390 tests with a flying rig including 141 free flights for testing and optimizing the flight control system, judgement of handling qualities and development of suitable displays;
- investigation of sonic and thermal load distribution on the aircraft;
- airframe assembly strength and temperature tests;
- static structural testing of the complete airframe.

Fig. 12 shows the flying test rig which was flown for two years and operated by 14 pilots from Germany, Italy, USA and UK and which has delivered very encouraging test results.

A few characteristic examples from the development test work are presented in the following figures (predictions).

- Fig. 13: Engine intake temperature increase due to engine exhaust gas ingestion. Different means to get rid of rear lift engine recirculation problem are presented, from which a 30° outboard tilt of the rear undercarriage doors has been selected as the final solution.
- Fig. 14: Temperature impact on air time in vertical take-off condition.
- Fig. 15: Jet induced ground effect versus aircraft height above ground and jet induced downwash during transition.
- Fig. 16: Intake and jet influence on longitudinal and lateral motion. The latter can cause critical conditions in the lower speed range due to yaw-roll couplings.
- Fig. 17: Sonic loads on the airframe.

All these test results have been of fundamental importance to reduce the development risk, let alone that a smaller number of uncertainties still remained to be clarified by aircraft flight testing.

5. FINAL AIRCRAFT DESIGN

The VAK 191 B design (Fig. 18 and 19) was finally frozen with:

Aerodynamics

- swept wing in high position with relatively high wing loading and low aspect ratio.
- trailing edge flaps and symmetrical deflectable ailerons for lift increase (flaperons),
- fixed fin with rudder and low mounted all flying tail and
- wing fuselage combination for optimum steady state transsonic flying; high angles of attack are only limited by lateral stability ($\alpha \sim 25^\circ$); no pitch up tendency.

Structures

- primarily of heat resistant aluminum alloy,
- steel and titanium for high strength high temperature areas,
- fibre construction access panels.

Propulsion

- Rolls Royce/MTU swivelling nozzle lift/cruise engine, 10,000 lb nominal static thrust,
- Rolls Royce light weight lift engines, 6000 lb nominal thrust each, deflectable by outlet doors and thus providing for emergency flight capability after main engine failure (get you home),
- variable geometry main engine air intake (slides forward and forms semicircular slots, acting as auxiliary intakes) provides satisfactory intake performance over the complete speed range,
- lift engine air intakes are of bellmouth type, no additional ram devices are necessary,
- power plant arrangement allows for full utilization of future thrust improvement either of the lift or lift/cruise engine.

Subsystems

- advanced fly-by-wire control system with integrated CSAS provides excellent handling qualities in jet borne and wing borne flight,
- BITE allows for automatic flight control system check-out, both on ground and in flight;
- 4000 psi hydraulics for minimum weight and volume of system and components,
- secondary power system with auxiliary power unit (APU) providing for independence from ground power supply, both for engine start and systems operation.

Crew Station

- excellent visibility over nose and to the sides,
- conventional controls,
- zero-zero rocket supported ejection seat,
- adequate displays

6 FINAL AIRCRAFT DESIGN

After ground testing the aircraft was mounted onto a pedestal (Fig. 20) to undergo final engine runs and to have the flight control system tested and finally adjusted. This tethering tool is capable of lifting the aircraft 6 ft off the ground and does allow rotation in pitch and roll axis over a range of $\pm 15^\circ$ and in yaw over the full 360°

Only 5 tests for aircraft No. 1 were necessary which were followed by lift-off tests within the undercarriage stroke to get the pilot adjusted to handling and engine response before the first vertical take-off was performed. Several flights were conducted in the hovering arena (Fig. 21) followed by accelerating and decelerating transitions including lift engine shut down and relight tests and wing borne flights up to 300 KIAS.

The flight test results which have been achieved until the program was terminated can be summarized as follows:

- the aircraft shows outstanding handling qualities throughout the flight envelope, opened so far, and very precise flying was demonstrated while the aircraft is jet borne;
- there is a positive ground effect up to 10 feet above the ground and the aircraft is virtually free of engine exhaust gas recirculation;
- engines, aircraft energy supply and subsystems operate in a satisfactory manner within their specified range, and
- there were no structural problems due to sonic or temperature effects.

7 COMPARISON OF PRELIMINARY VERSUS FINAL DESIGN AND TEST DATA

During aircraft construction and all test phases a number of changes were necessary.

None of these changes have been of significant influence to the program, but a few ones are worth to be mentioned.

a) During development and construction:

- Improved access to main engine accessories was required and led to an additional wing tilt of 0.5°. This was followed by redesign of wing fuselage combination, influencing stability of the aircraft with a noticeable trend towards pitch-up at low angles of attack ($6^\circ < \alpha < 8^\circ$) due to wing tip vortex interfering with tailplane. Pitch-up-trend was then cleared by 8° dihedral of the horizontal tail and original stability was regained by the addition of extended wing tips outside of the outrigger nacelles.
- The production of the high pressure bleed air ducts (bleed from the lift/cruise engine) which were formed from sheet material, caused a considerable welding problem, because the selected heat resistant light weight material, a titanium-aluminum-tin-alloy, showed brittleness along the welding zones. This problem was solved by use of steel alloy (nimonic) connected with some weight penalty.

b) During aircraft ground and flight testing:

- First transition trials showed inconvenient aircraft overshoots in roll axis. The magnitude of these had not been noticed during pedestal and hover tests and led to modifications of the CSAS amplifier circuits until an acceptable adaption was found.
- A greater safety margin for side slip manoeuvres in transition was desirable and therefore the roll control nozzle exit areas have been increased by 20 %. The new set of nozzles has not yet been installed.
- Because of inadequate free play in the mechanical linkage for lift engine control, the engine response to inputs from the CSAS was insufficient when thrust modulation was engaged. A partial redesign of the system was necessary to get rid of that problem.
- The aircraft empty weight is about 400 kg more than it had been when final design was started. Part of this increase is the tribute which had to be paid when the program was turned over to experimental. Therefore several additional tests and further refinement work originally planned for weight optimization had to be abandoned. The additional weight falls to the airframe and to aircraft subsystems as well.
- The thrust balance included a safety margin for uncertainties such as recirculation which did not occur during aircraft testing and thus the weight increase could be balanced for vertical take-off.

8 CONCLUDING SUMMARY

The VAK 191 B design approach and the technical development procedure have been described and several examples were given to demonstrate the interaction of different technical disciplines. Due to the high level of technology required, the program had to be characterized as a high risk program, which could only be mastered by consequent control of the various design steps and significant back up by means of test work.

The design process which started out with mission analysis, initial theoretical investigations and parametric studies resulted in various configurations, which then have been evaluated in more detail in order to select the most suitable ones, which could be realized with the available technology. As a result of the studies and supported by first model test data a first preliminary design of the aircraft could be derived.

A draft specification and a detailed work breakdown structure with built-in iterations was then set up for further definition.

In course of the development, design optimization was achieved by further trade-offs and continuously refined model tests and simulation, using analogue and hybrid computation.

Finally, test runs were conducted with integrated hardware equipment and structural components as well as complete aircraft subsystems. The test results have continuously been used for updating the design.

By this way, the development risk could be reduced step by step to an acceptable limit. Final risk, which remained for flight testing was reduced by very careful monitoring with quick look during test runs and detailed data reduction after each flight. 31 flights were performed and the flight envelope was opened up to 300 knots. Most of the tests were devoted to VTO, hover performance and transition.

The total program cost were kept to 530 Mill DM, including engine and subsystem development and flight testing as well, and part of the program was under fixed price contracts.

Looking back from where we are today, the design procedure and thoroughness to reduce the risk, has proven to be right. Of course the design methods have meanwhile been improved, more prediction methods and back-up data are now available and thus the number of iterations for future V/STOL designs would certainly be reduced.

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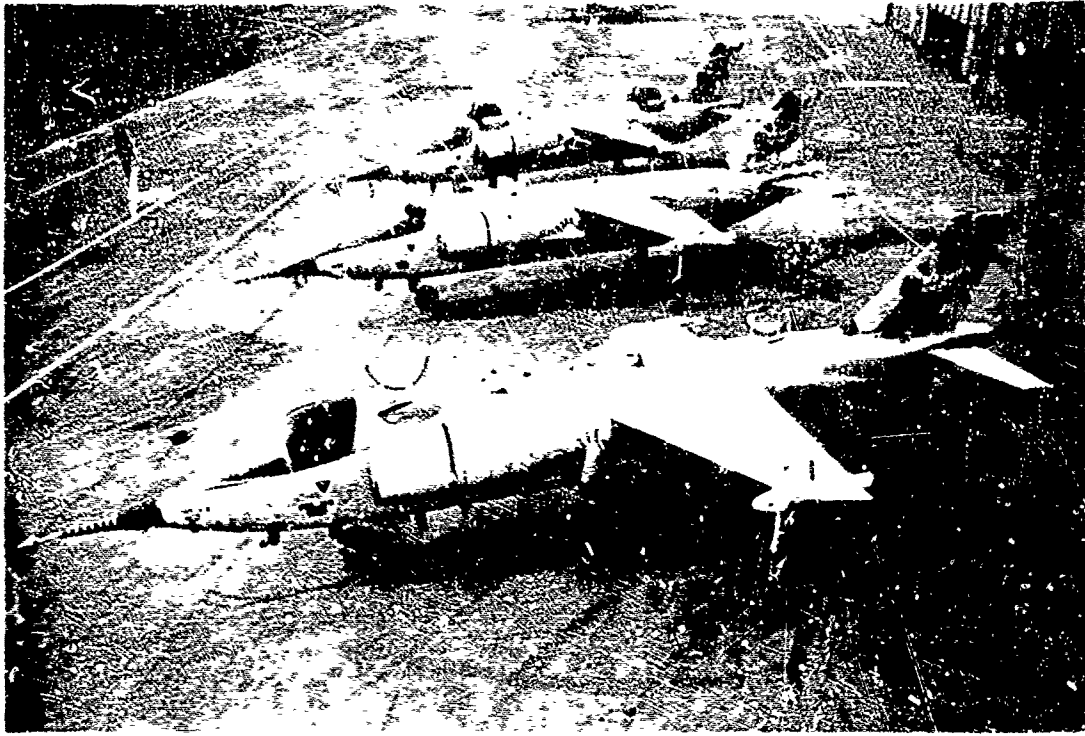


Fig. 1 VAK 191 B

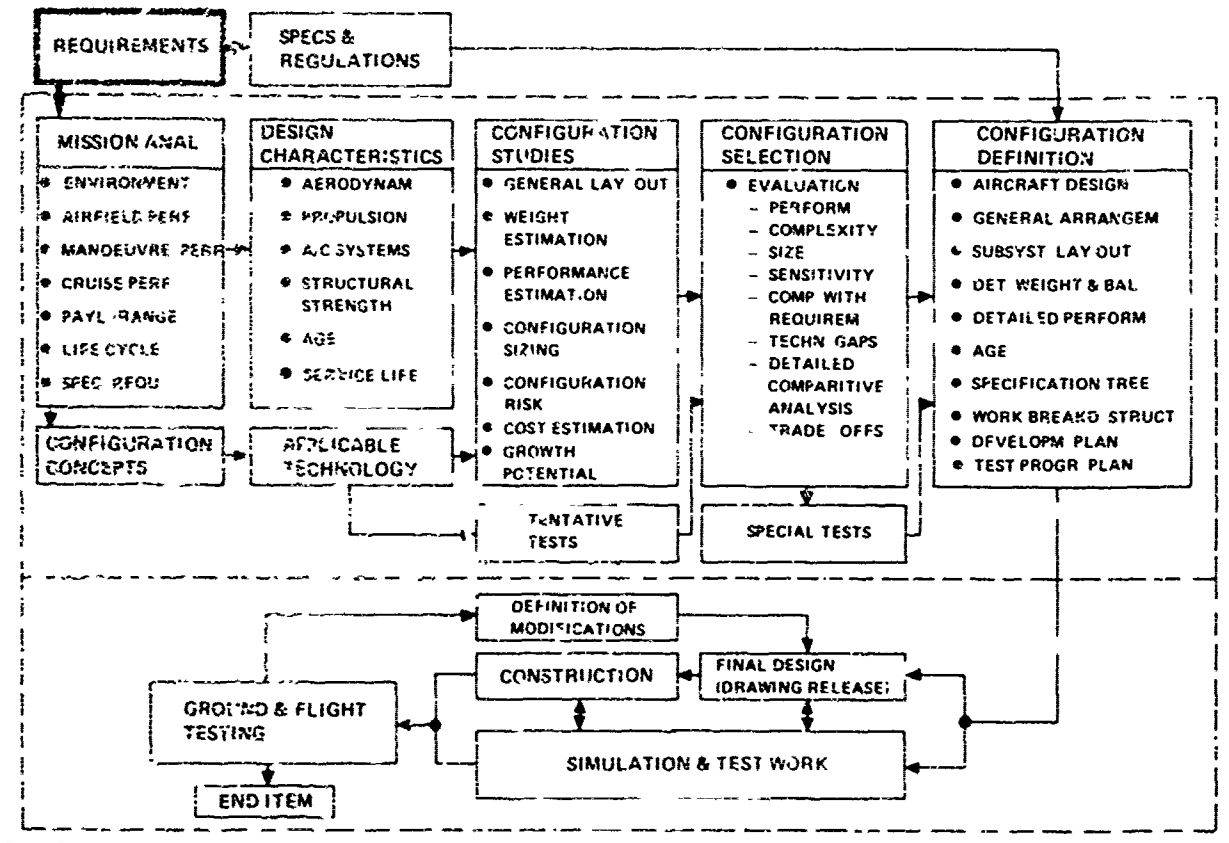
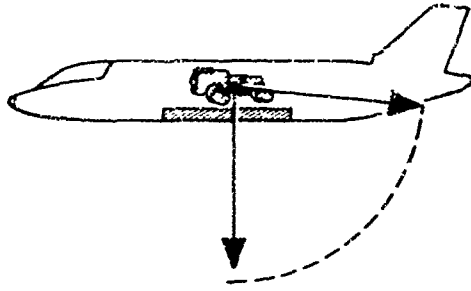
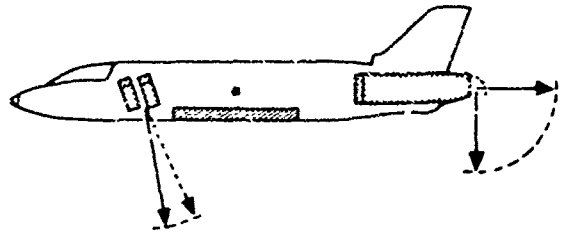


Fig. 2 DESIGN FLOW CHART

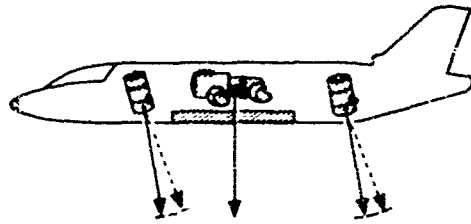
LIFT/CRUISE



LIFT + LIFT/CRUISE
ASYMMETRICAL



LIFT + LIFT/CRUISE
SYMMETRICAL



DIRECT LIFT + CRUISE

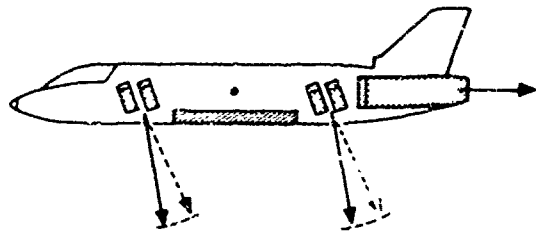


Fig. 3 POWER PLANT CONCEPTS

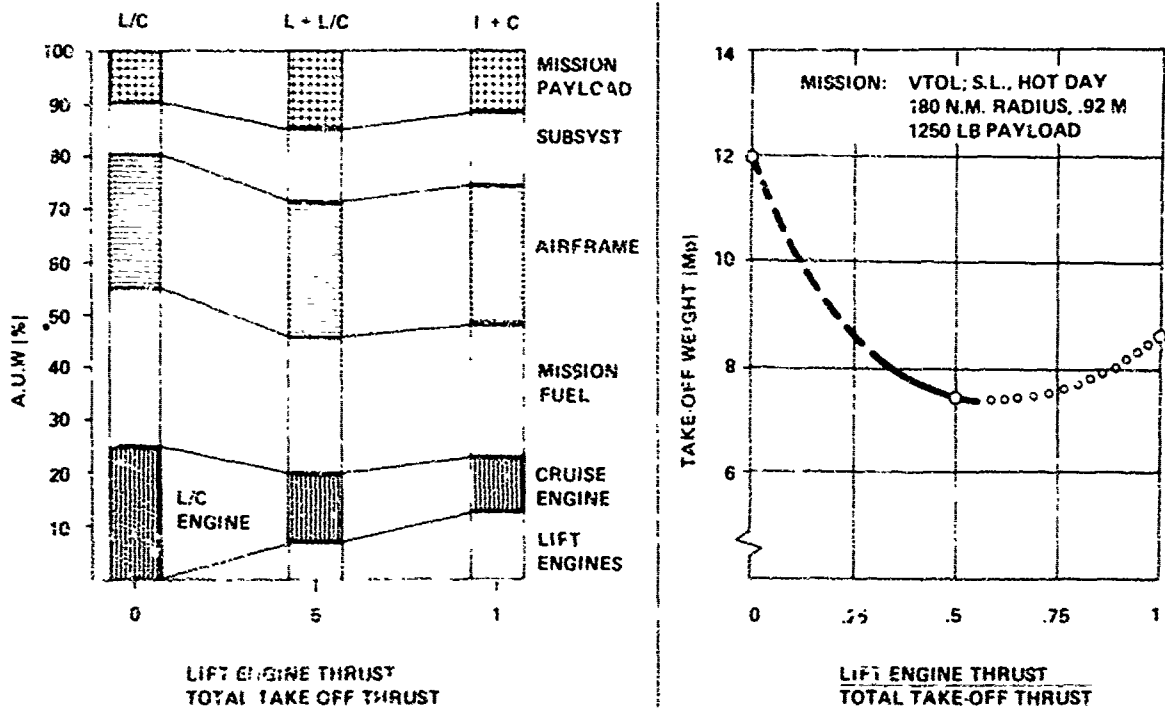


Fig. 4 VSTOL CONCEPTS WEIGHT CONSIDERATION

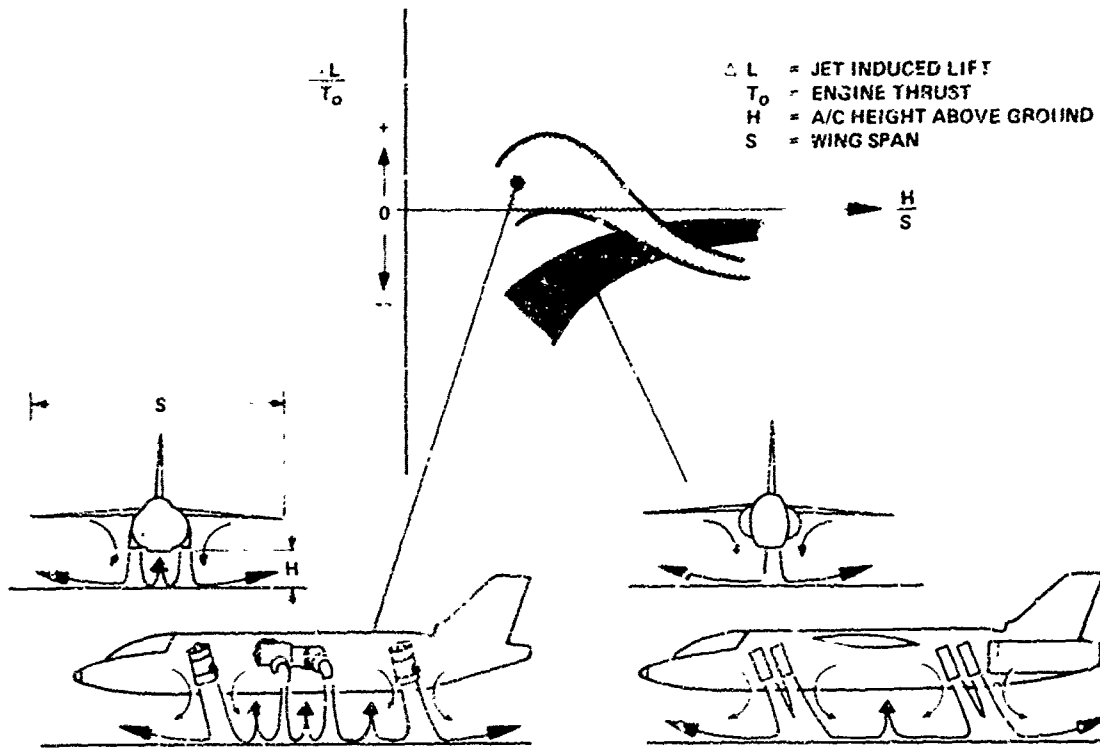


Fig. 5 JET INDUCED GROUND EFFECT

POWER PLANT CONCEPT					
EVALUATION ASPECTS	THRUST ①	4.5	6.2	6.2	5
	POWER PLANT WEIGHT	4.5	3.15	3.1	2.5
INT. LOAD BAY		NO	YES (FITS BEST)	YES	YES (INFLUENCE ON FUS. LENGTH)
CREW SAFETY (ENG. FAILURE)		LOW - ACCEPT.	UNACCEPTABLE	ACCEPT.	ACCEPT.
EMERG. FLIGHT AFTER ENG. FAILURE		NO	NO	YES (I.E. THRUST VECT.)	YES (I.E. THRUST VECT.)
GROUND EFFECTS		POSITIVE	NEGATIVE	POSITIVE	NEGATIVE
HANDLING AND WORKLOAD IN TRANSITION		ENGINE THROTTLING - NOZZLE ROTATION	SEPER. ENG. THROTTL. + NOZZLE ROTATION AS FUNCT. OF POWER + L.E. OPERATION	COMB. ENG. THROTTL. - NOZZLE ROTATION + L.E. OPERATION	SEPER. ENG. THROTTL. + L.E. OPERATION
MANOEUVRE THRUST VECTORING		YES	NO	YES	NO
ESTIM. A/C MISSION TAKE-OFF WEIGHT		12 MP (26 430 LB)	7.5 MP (16 520 LB)	7.5 MP (16 520 LB)	8.5 MP (18 720 LB)
ADVANT. OF ENG. GRWTH	LE	-	NONE ②	VTO	VTO
	MAIN C	VTO AND CRUISE	CRUISE	VTO AND CRUISE	CRUISE
OVERALL COMPLEXITY		LOW	HIGH	MEDIUM	MEDIUM - HIGH

① ESTIM. THRUST/WEIGHT
L.E. = 10, L/C = 4.5, STRAIGHT ENG. = 5

② IMPROVEMENT OF VTO PERFORMANCE
NECESSIT. THRUST IMPR. OF BOTH ENG. GROUPS

Fig. 6 EVALUATION OF POWER PLANT CONCEPTS

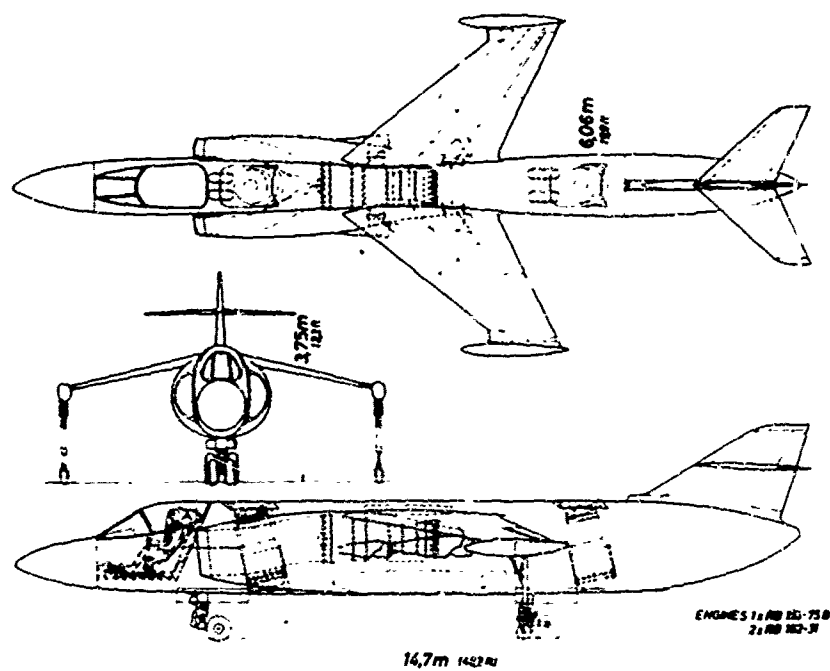


Fig. 7 PRELIMINARY DESIGN

- IMC CAPABILITY IN ALL FLIGHT PHASES
- AT LEAST ONE FAILURE IN THE SYSTEM MUST BE OVERCOME WITHOUT REDUCTION OF HANDLING QUALITIES
- IN CASE OF ENGINE FAILURE, AIRCRAFT ATTITUDE CHANGE HAS TO BE KEPT IN LIMITS TO PERMIT SAFE PILOTS ESCAPE
- BUILT IN TEST EQUIPMENT FOR QUICK TEST AND QUICK CHANGE OF LRU's

Fig. 8 FLIGHT CONTROL SYSTEM DESIGN CRITERIA

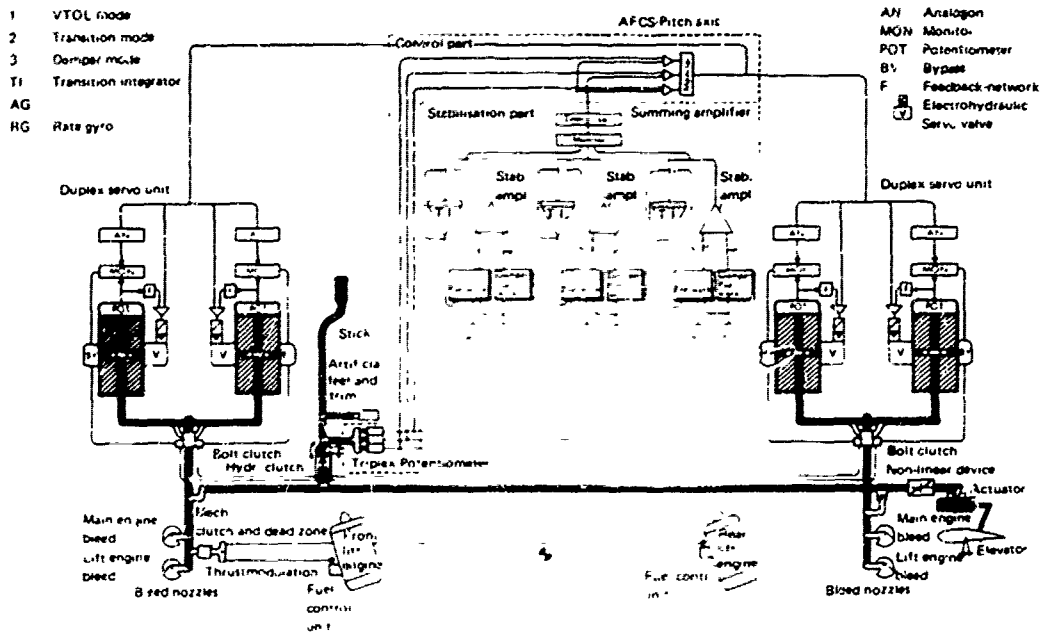
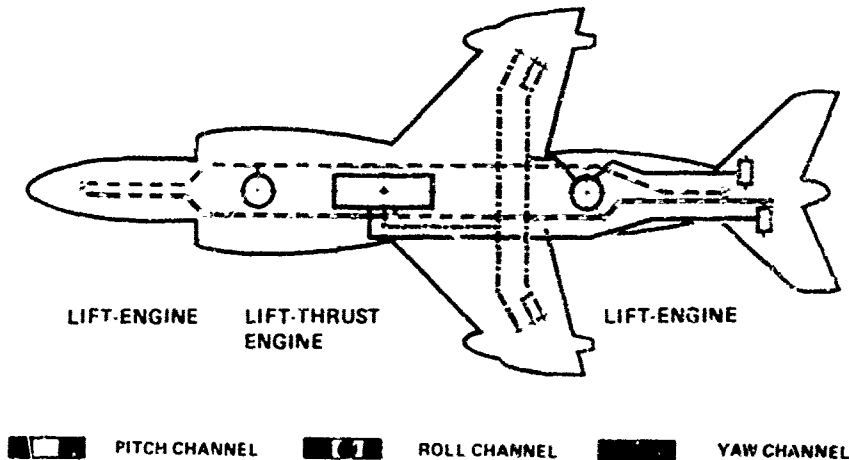


Fig. 9 AUTOMATIC FLIGHT CONTROL SYSTEM SCHEME OF PITCH CHANNEL



CONTROL CHARACTERISTICS: ROLL : 1.4 RAD/S²
 PITCH : 1.2 RAD/S²
 YAW : 0.35 RAD/S²

Fig. 10 BLEED CONTROL SYSTEM SCHEME

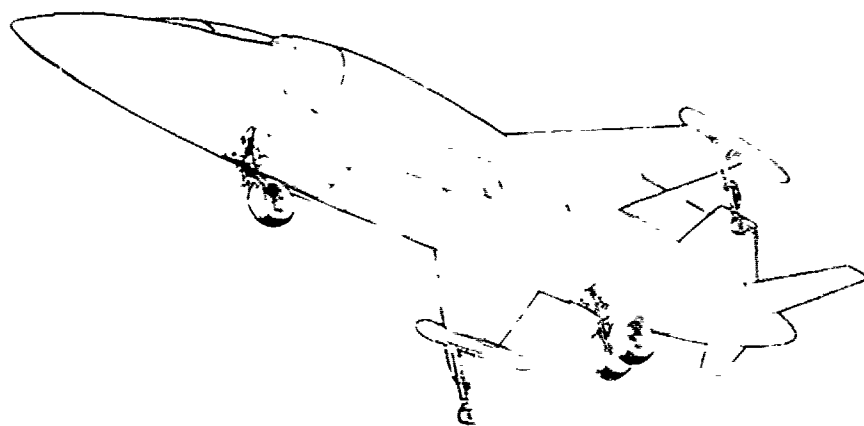
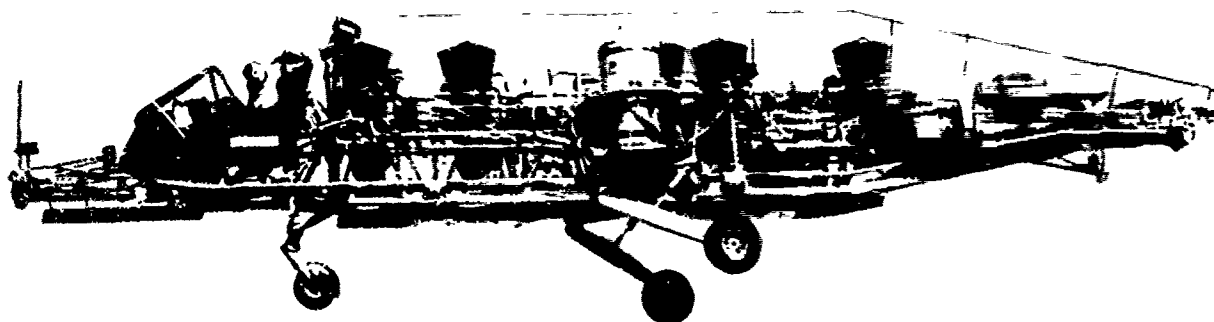


Fig. 11 UNDERCARRIAGE



WEIGHT	3500 kp
LENGTH	14,7
MAX. WIDTH	5,8 m
MAX. ENDURANCE	12 Min
POWER PLANTS	5x ROLLS ROYCE RB 108
AUXILIARY POWER UNITS	2xBMW 6012
EJECTION SEAT	MARTIN BAKER Mk GH G
FREE FLIGHTS	141
TETHERED FLIGHTS	171
GROUND TESTS	80

Fig. 12 FLYING TEST RIG

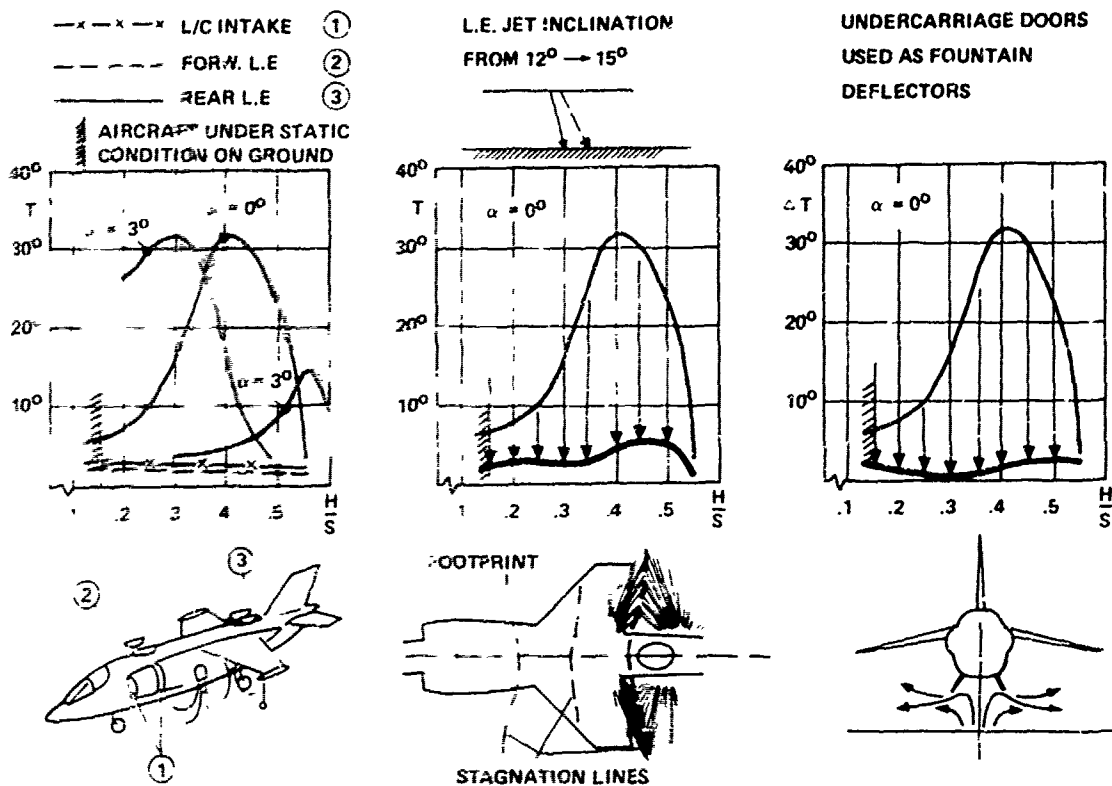
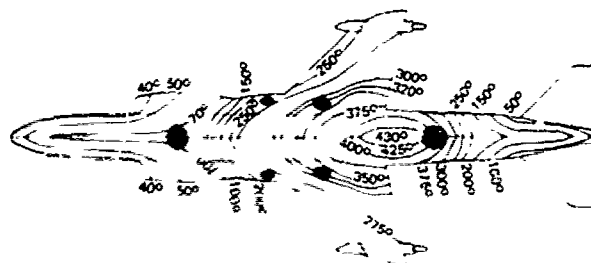
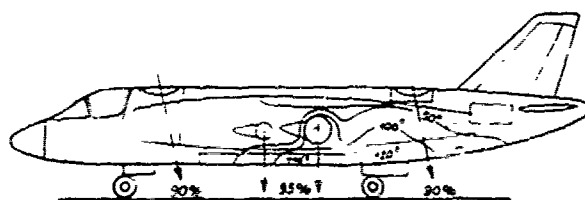


Fig. 13 INTAKE TEMPERATURE INCREASE (EXHAUST GAS INGESTION)



TEMPERATURE DISTRIBUTION
ON WING AND FUSELAGE LOWER SIDE



STRUCTURE TEMPERATURE AFTER 40 SEC.
OF ENGINE OPERATION

Fig. 14 TEMPERATURE IMPACT ON AIRFRAME (VERTICAL TAKE-OFF CONDITION)

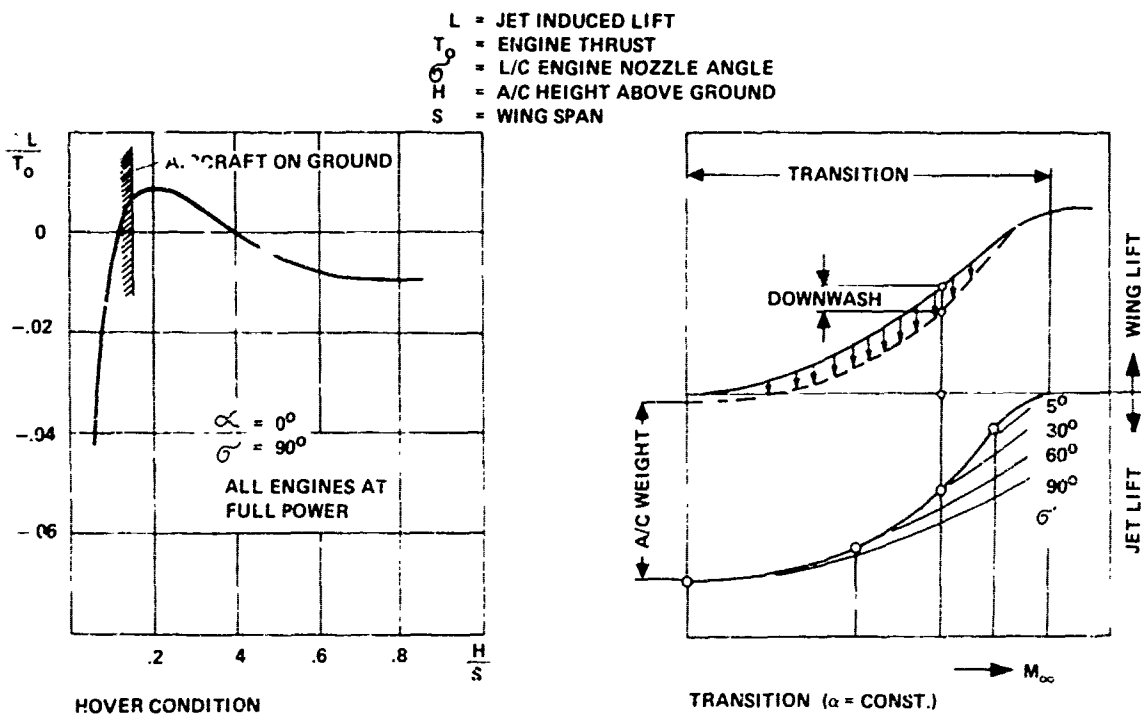


Fig. 15 JET INDUCED LIFT IN HOVER AND TRANSITION

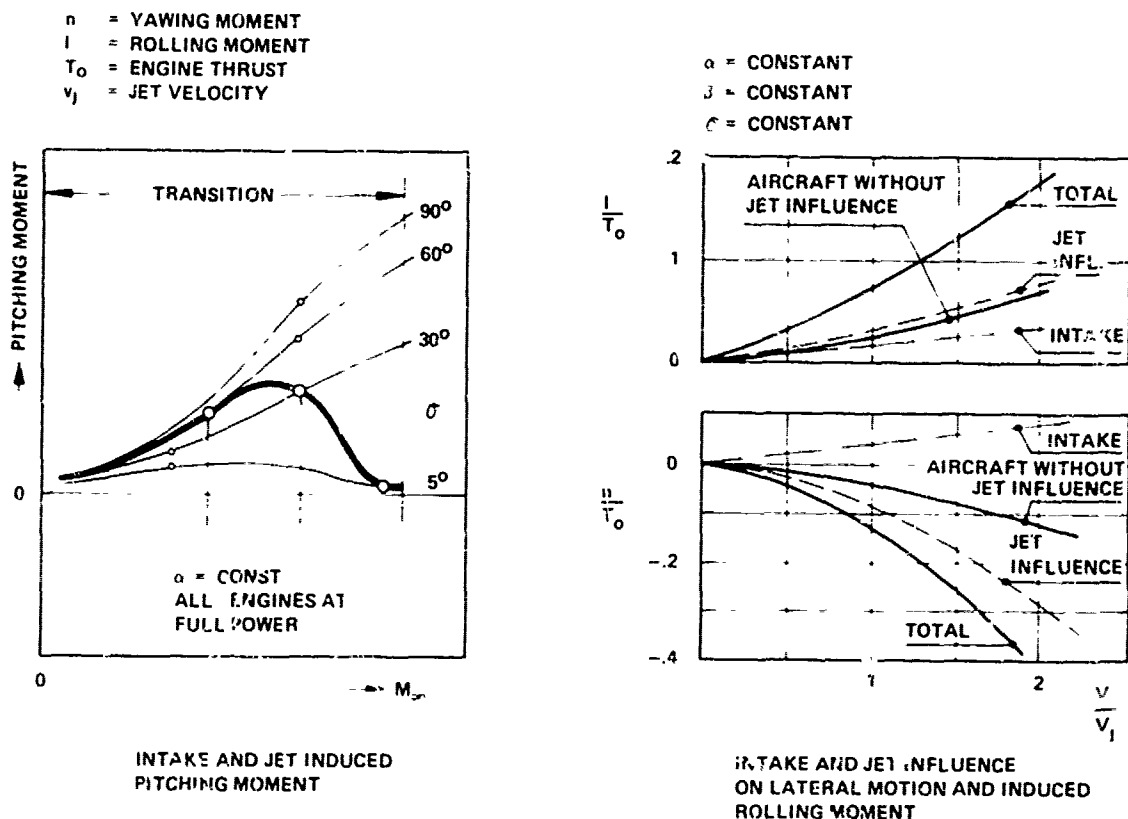
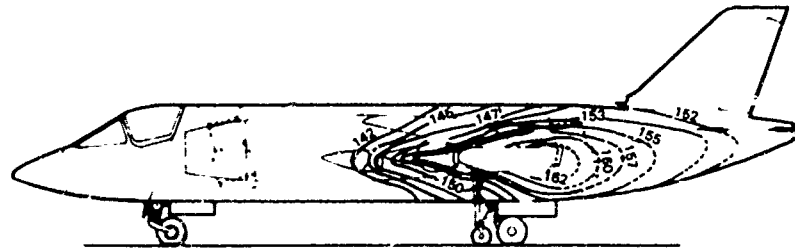
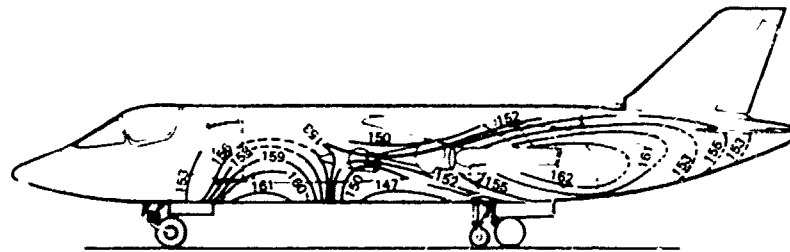


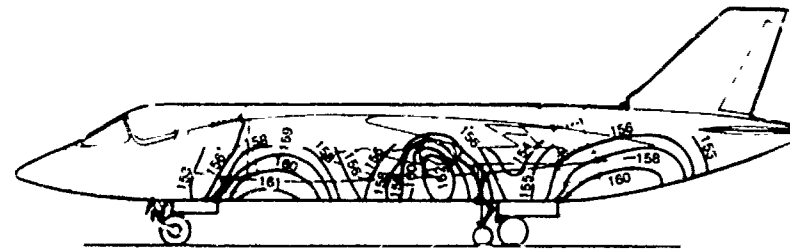
Fig. 16 INFLUENCE OF INTAKE MOMENTUM AND JET ON AIRCRAFT MOTION



CONVENTIONAL TAKE-OFF

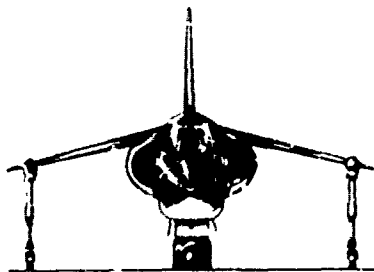
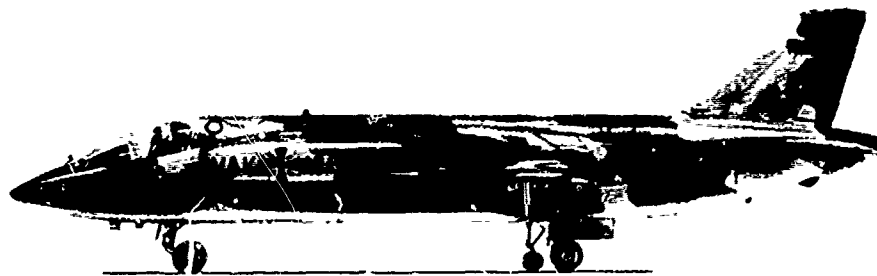


SHORT TAKE-OFF



VERTICAL TAKE-OFF

Fig. 17 SONIC LOAD ON AIRFRAME [dB]



DIMENSIONS:

WING SPAN	6.18 m
OVERALL LENGTH	14.72 m
HEIGHT ON THE GROUND	4.295 m
HEIGHT IN FLIGHT	3.495 m
WING AREA	12.5 m ²
DIHEDRAL ANGLE OF WING	-12.5°
DIHEDRAL ANGLE OF HORIZONTAL TAIL	-8°

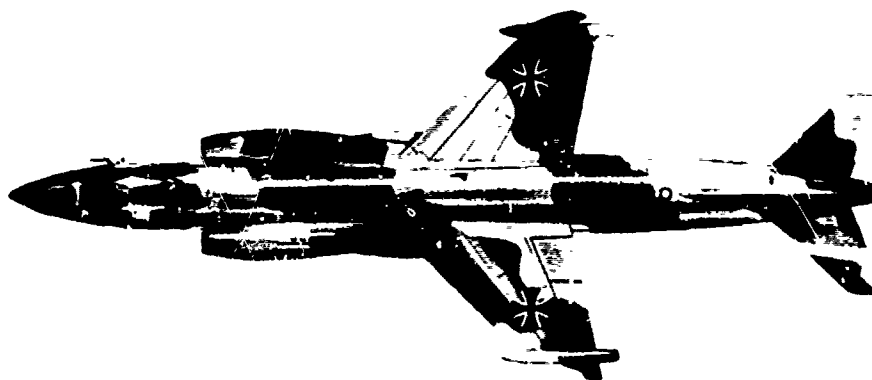


Fig. 18 THREE SIDE VIEW

AN APPROACH TO DESIGN INTEGRATION

BY

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SUMMARY

This paper discusses the nature of an aircraft design team, and how new techniques can be integrated into the organisation to improve design efficiency. In particular the structure and implementation of new techniques are described as they have been applied to the preliminary stages of design in Hawker Siddeley Aviation.

1. INTRODUCTION

For some decades now aeronautical engineers have steadily improved the accuracy of calculation in each specialist area in an attempt to improve the overall quality of design, i.e. the balance of resources has been such that relatively slight effort has been applied to the problem of integrating these elements of the design process. And yet this is of comparable importance as close observation of any design team will show. Any one designer spends a significant, sometimes dominant portion of his time collecting, transforming and disseminating data, and repeating calculations due to modifications in the design. A solution to this problem might pay off handsomely in terms of reduced elapsed time, manpower and frustration and should provide management with a much improved picture of the state of the design at a given time.

In Hawker Siddeley Aviation, as in most other aerospace companies, we have the added complications of several geographically remote design teams, collaborative ventures with other companies, and a wide range of aircraft types. Any attempt to improve the design efficiency of a company must evidently solve the tendency towards divergence which these factors can introduce.

Systematic studies of what help could be provided by recently developed techniques were initiated by H.S.A. in 1970, when the Company began a joint study with the Department of Trade and Industry's Computer Aided Design Centre and the University of Cambridge (through the Engineering and Computing Laboratories). Inevitably the power of the computer was an important factor in these studies, although design and organisational methodologies were also taken into account.

During the first two years, the more difficult areas were the subject of theoretical and experimental studies (Reference 1). A proto-type system was created and evaluated, and proved the feasibility of the approach. As a result, the decision was taken to mount a working system in the Future Project Department at each of H.S.A.'s four design teams. It is this system which forms the basis for this paper, although we are also applying such techniques in designing systems which will do a comparable job in the later design stages.

2. AREAS OF DESIGN

The total design process can broadly be split into four levels of complexity as follows:-

1. The first stage is the Feasibility Study, during which many novel solutions may be tried in an effort to reach a configuration which could be effective. Research will be initiated into areas which are beyond existing experience in an effort to quantify the resulting effects on the aircraft, but inevitably in some areas the estimation of aircraft effectiveness will be based on slender evidence and simple mathematical models. The manpower involved and the quantity of data generated will be relatively small, and the results to be expected from the study are whether or not a workable solution exists and the approximate dimensions for optimum performance. A thorough optimisation of the design at this stage is of limited value, - if the design is conventional then experience of the active constraints will normally give an adequate approximation to the optimum. On the other hand if the design is unconventional then the best mathematical model will be too crude to warrant rigorous treatment and will in any case change rapidly with time.
2. Then follows the Project Study during which the possible configurations from the Feasibility Study are subjected to a much more detailed analysis with the aim of finding the optimum values for the best configuration, and the associated commercial prospects. This is the last stage of design at which all the major parameters are still variable and hence meaningful optimisation of the whole aircraft can only take place at this stage. If simple empirical methods are not adequate for the type of aircraft being considered then more substantial fundamental methods must be applied (such as finite element structural analysis) or expensive testing undertaken (for example slender delta aerodynamic characteristics). The resources applied are obviously becoming substantial and the period of this study will be of the order of one year.
3. The Project Definition phase is then entered with the intention of refining the estimates by taking the design to a greater level of detail. All the structural elements will be outlined and the external shape defined. Detailed calculations will be backed up by extensive tests in an attempt to minimise the risks involved in quoting guarantees to customers. There is little opportunity at this stage for reoptimising the aircraft as many important decisions will have been taken and dimensions frozen. At least 100 man years will be expended during the year or so spent in this phase, and a considerable quantity of data will be generated. During this period it is at the component level that optimisation takes place, and the overall performance is updated as estimates are refined.
4. Finally, with the aircraft committed to production, the Detail Design stage is entered. All of the important design decisions have already been made and the task is basically to produce suitable instructions for the production department. This will take about two years and involve perhaps 1,000 man years of effort.

The second of these stages was chosen by Hawker Siddeley Aviation as most suitable for applying the results of research mentioned in the introduction. The scale was such that the implementation costs were reasonable, but there still existed in microcosm all the specialists and multiple projects of a complete design team.

3. AIMS AND CONSTRAINTS

The aim in producing the system, known as the Future Projects Design System (FPDS) could be generally termed as the improvement of design capability and efficiency within the Future Projects teams by any means which did not infringe the following formidable constraints:

- Benefits had to be seen to accrue soon after project initiation and with a low manpower hump. This latter requirement is an acknowledgment of the fact that we required the best engineers to help design and implement the system and that we could not expect to monopolise their effort over a substantial period in the presence of many live projects. It also implied a major effort to make the system easy to understand and to provide tools which made it relatively quick to build. No major new software could be created in the timescale (such as database management or operating systems).
- Existing computer hardware was to be used. (The mainframe computers throughout H.S.A. are I.C.L. 1900 series.) Advanced operating systems are available which provide compatible on-line and batch access, database control and a flexible simple command language.
- It was recognized that introduction of the system into use could not be allowed to affect too much of the work of the design team. Thus the concepts and details of its use had to be very easy to learn. Of course considerable resistance could be expected if individuals found its use to be less pleasant than alternatives.

4. SYSTEM ARCHITECTURE

It was apparent from the earlier research (Reference 1) that the system should fulfil a number of functions:

4.1 DATA STORAGE

- A single computer-stored data area should exist for each aircraft. Controls should be available such that the contents of this data area are secure, and could be administered by a senior designer, or logically subdivided for control by delegated specialists.
- A further series of data areas should store all data relevant to more than one aircraft. This includes data such as wing section properties, engine data and standard parts, and is called 'experience data'. These data areas again have suitable controls for administration by specialists.
- The methods applied by specialists are stored in the form of programs, and controlled by themselves.

These three types of data area can be seen from figure 1 to conform to the usual 'matrix organisation' for a typical design team. The collection of data areas is called the design database. At any time, there will in general be more than one project passing through the design team, with several at each stage. One man will have responsibility for each project and will obtain his manpower resources from the specialist departments. Each individual designer is responsible to a chief specialist for technical methods and standards, and to a type designer for producing results for a particular aircraft.

This same organisation can be drawn to illustrate a complete design team or (as applied to this paper) to represent a project design group which is a part of the larger team.

Thus, although all the data is stored on a single computer, control of the contents of the database and who can read or modify the data is securely in the hands of those who control that data in a manual system. The structure of the database is largely independent of the applications programs, as the data necessary to describe an aircraft is not dependent on the methods used to design it. Thus, whilst one would expect the database format to change little with time, the programs are considered to be interchangeable modules which can be replaced by new versions as technology progresses. Thus a simple structural analysis module can be replaced by a finite element method, and a conventional take off analysis can be replaced by a STOL or VTOL module. Where alternatives exist the designer can choose for himself.

It is expected that before long the running of these programs could be controlled by a single type designer for a particular aircraft, and that specialists would only be needed to monitor the calculations. These specialists will then be able to apply far more effort to research, to developing and refining methods, and to implementing the resulting methods as program modules.

The database is structured internally and externally as a tree structure (Figure 2). At each node there is a set of information describing who may read or modify the data at that level in the tree. Passwords, user numbers and project codes can be used to restrict access to individuals or groups of designers. For example the type designer of aircraft Z might be given ownership of node B by the manager who owns node A. The type designer can in turn pass ownership of the geometry files to a designer at node C. It is then up to this specialist to decide which other designers may read versions of the files 'fuselage', 'wing', etc. Only the specialist is likely to be allowed to modify any of his files.

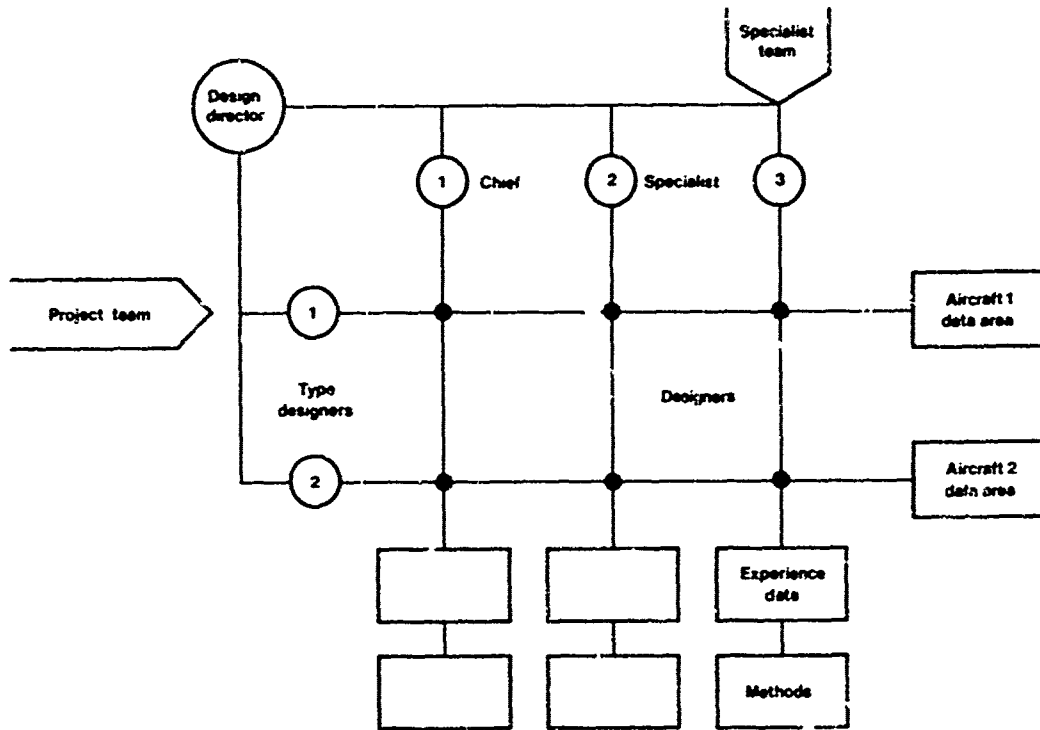


Figure 1. A design team-matrix organisation and data areas

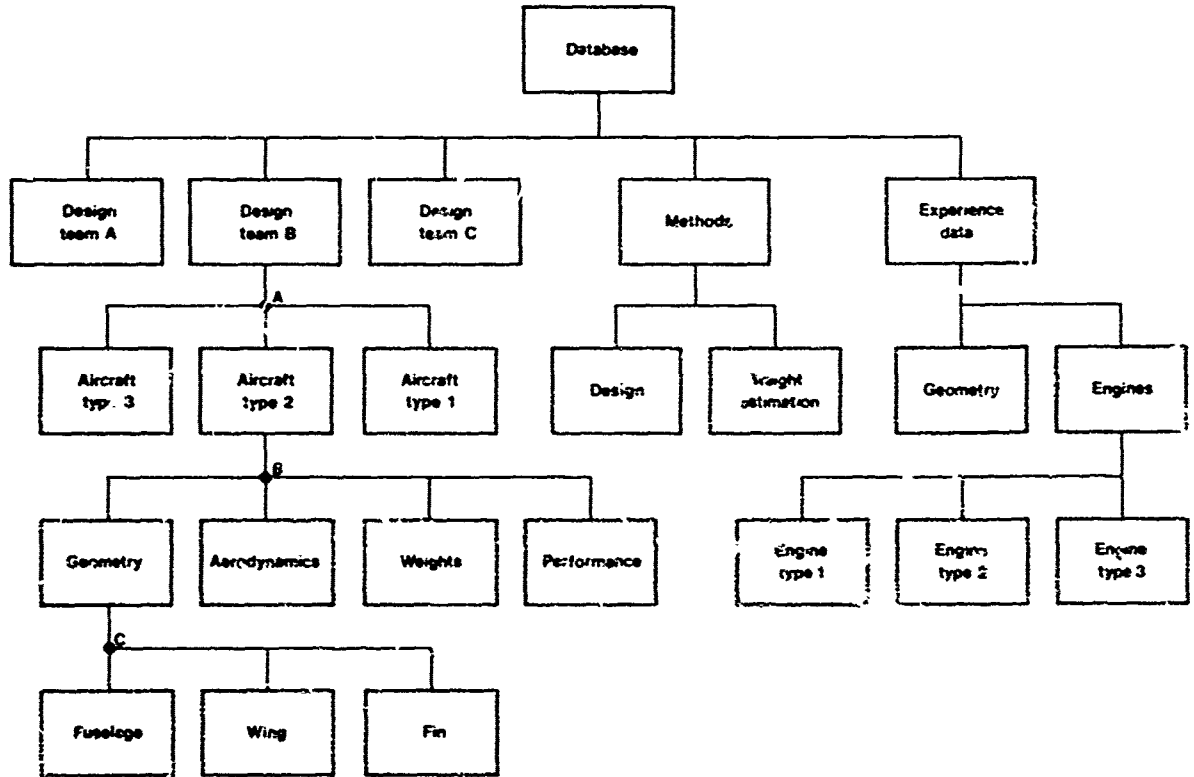


Figure 2. A design database

We would have liked to be able to construct and maintain our database using the latest ideas, which involve complete separation of the formats used for storing data from the format as assumed by the applications program. This entails the use of techniques which transform the data automatically between one format and the other. The examples illustrated in figure 3 shows

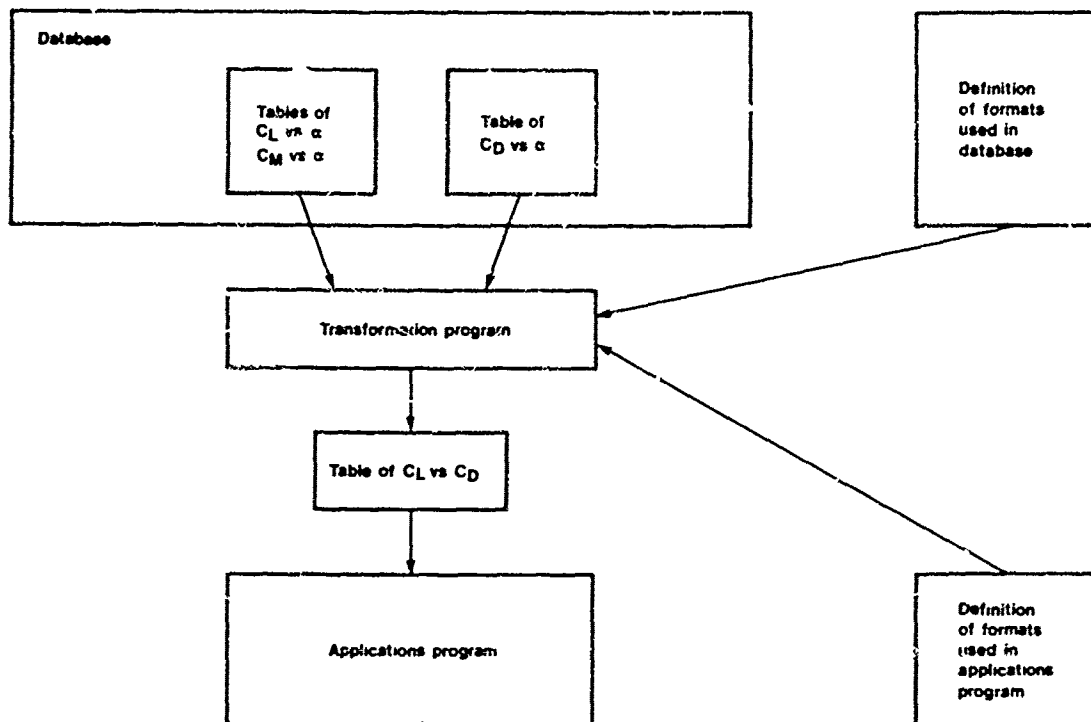


Figure 3. An example of data/program independence

how values of lift and drag coefficient can be stored separately against values of incidence in the database, but are automatically transformed into the required format for a particular application. Then if it becomes necessary to modify the database or an applications program, only the relevant format definition has to be altered. This makes the whole system dramatically easier to maintain. Note also that the transformation can include unit conversions.

Such techniques are likely to be viable in a few years, but for the present their efficiency is not adequate and hence the cost is too high. For now a much simpler approach has to be used, which places increased reliance on getting the database design correct at the start. The aids which we are using are outlined in figure 4.

The database consists of a collection of files each of which contain named items of data. For each applications program there is also a definition of which files are required, and which data items are needed from each file. When the applications program is run, a set of data access routines find the appropriate files, select the named data items and present them to the program. In general, detailed transformations of the data are not available, but data can be selected down to a low level. The exceptions occur where data is compacted to reduce storage space.

4.2 COMMAND LANGUAGE

The commands used by the designer are set up using a facility available under the operating system. This allows any collection of detailed operating instructions to be called by a single name. If any designer uses this command, the appropriate list of operating instructions is processed. This collection may include various conditional clauses ('if the program fails, then do as follows ...') and the result is a surprisingly powerful tool which eliminates the hieroglyphics talked by most computers. The designer will use commands such as TAKEOFF to initiate

a takeoff performance calculation and ERASE WINGGEOM (4) to erase version 4 of the wing geometry file. An example is given in Appendix 1; the most frequently used commands are seen also to have an abbreviated form (ER is short for ERASE).

Again a better tool is available at extra cost in overheads, as detailed in References 1 and 3. In general such a 'command interpreter' can be made to perform all of the arithmetic and data manipulation operations of a high level language such as FORTRAN, although it interprets the commands line-by-line rather than compiling a collection of commands before running them as with normal FORTRAN. The same language can also provide all of the commands necessary to drive highly interactive programs, such as those that can be used with a graphics terminal.

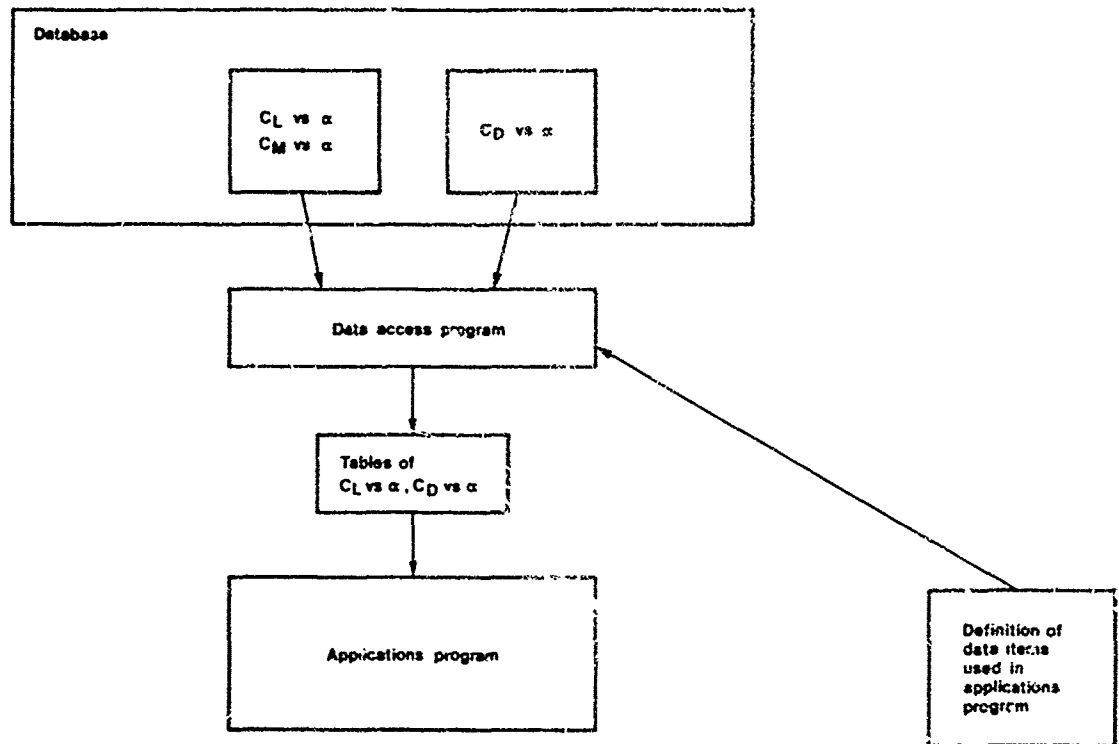


Figure 4. F.P.D.S. data access mechanism.

4.3 APPLICATION PROGRAMS

The actual calculations are carried out by a series of computer programs designed with a standard form of interface to the rest of the system, across which data and running instructions are passed. These programs are constructed such that they can be run individually (e.g. a profile drag analysis), can be used as a suite under manual control, or can be run in groups under the control of a higher level program. This latter program might perform an automatic optimisation or parametric study. So the order of use of the applications programs may be controlled manually or automatically and during a normal interactive design process a number of these programs will be used until convergence is obtained.

The system can then be represented as in figure 5, where a sequence of applications programs operate upon the aforementioned aircraft and experience database.

In general, a program will be initiated by a command from a designer or a higher level control program, and may be entered at various points depending on the path to be taken through the program. It will then access data as required from various files on the data base and direct from the designer or control program. During and after the calculation it will output data to several files on the database and perhaps at a terminal, line printer and/or plotter.

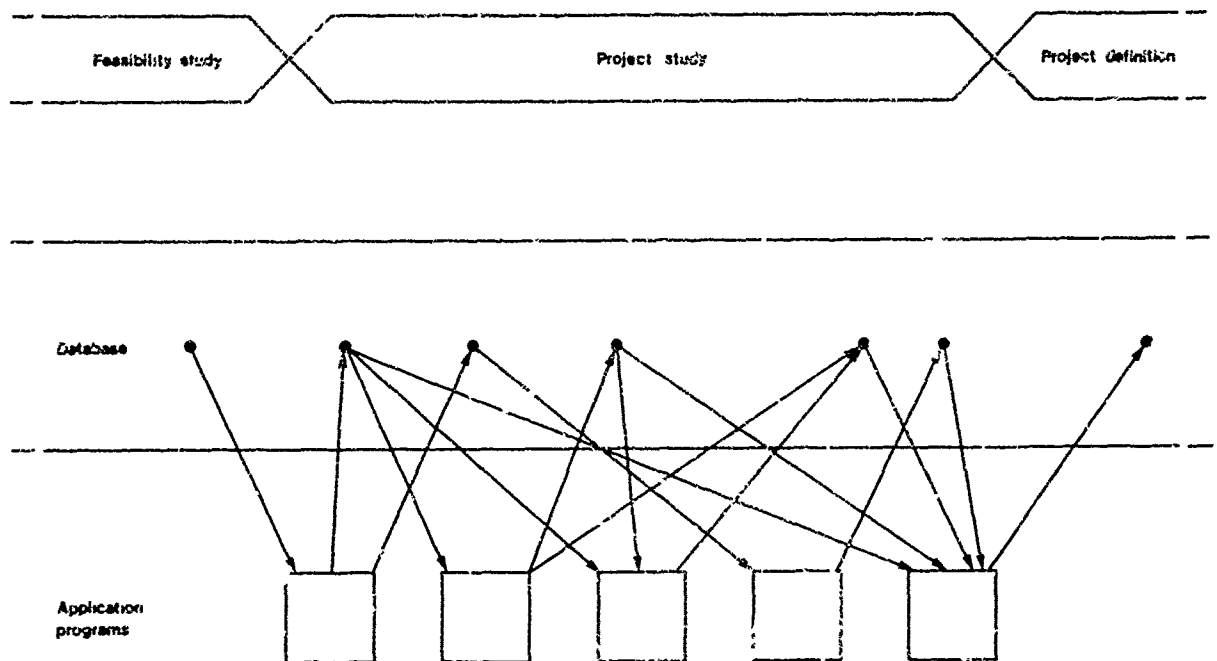


Figure 5. The interaction between applications programs and the database

5. IMPLEMENTATION

There are two basic stages in the creation of any complex system, - the first in which the outline of the design is sketched, and the second in which the details are filled in and the system brought into use. The first stage is technically difficult but organisationally straightforward, and many companies have successfully completed it. Where almost every starter has fallen is in mounting the second and far more difficult hurdle, - the successful implementation. By this is implied a reliable system in use by all those who should make use of it, which can be updated and maintained for a significant time. During this period a large number of people become involved and while the technical problems are straightforward the organisational problems become severe. A sense of involvement, culminating in widespread acceptance of the system is essential, and this is only possible with excellent communications. A study of various systems which had failed (some of them our own) revealed an apparently obvious but very difficult set of conditions which are seldom fulfilled:

- An experienced designer must be appointed as fulltime project leader, with responsibility for forward planning of technical priorities and resources, detailed design and implementation of the system.
- For any particular specialist program the most experienced specialists must be available to decide which methods should be used. If more than one design team is involved, then the methods must be completely agreed before programming begins.
- The above conditions cannot be fulfilled unless the whole project is initiated at the highest management level, with extensive support from relevant staff at lower levels. Hence the project has to be 'sold' to a large number of managers and prospective users. This commitment is not just for the year or two required to build the system, but also for the extensive maintenance required for the life of the system. For every five man years spent on the original programs, one man will be required permanently to update and maintain them and in the volatile life of a project office new requirements arise continuously, so a project leader will be needed indefinitely to prevent the system from diverging due to local developments.

With F.P.D.S. these conditions have and still are being met, despite the inevitable pressures to use the best designers on live projects. From early studies it was evident that the majority of methods could be made common for transport, strike and other types of aircraft, i.e. all four of our design teams could and should use the same method. Evidently this should lead to the best possible methods being made available to everyone, and to consistent results from competitive inter-site studies. (Where this was not sensible only two or three teams would be interested in a particular method.) Hence a 'specifier' is appointed at one site for one or a group of methods, and a 'consultant' at each of the other relevant sites. Together they agree a specification which is then programmed at the specifier's site. The tasks of detailed specification and programming have been considerably eased in two ways:

- Special documentation has been produced which rigidly defines the steps required to produce a program. The usual glut of computer manuals are not used as the difficult operations of data input/output and describing (to the operating system) how the program will be run are closely defined (to the extent that each programmer starts work with a standard, partially completed program which contains all the awkward statements. This approach is desirable to obtain a set of programs from many individuals (many with little previous programming experience) which subsequently appear compatible to the designer.
- Almost all program development is done online via a teletype. This perhaps halves the manpower required and reduces the elapsed development time by order of magnitude.

In this way we have almost completed our first batch of 41 programs which cover a wide range of subsonic and supersonic aircraft, both military and civil. We are now steadily filling the gaps and modifying or replacing programs to fulfil new requirements. We expect that the system will eventually level out at about 70 or 80 programs, although there will always be a steady turnover as methods become obsolete and new ones arise.

To give an indication of the structure of programs and data, the example detailed in the appendix is shown in part in figure 6. In this case a passenger transport is being designed. Firstly the fuselage cross-section is designed to provide the required accommodation standards, and then the remainder of the fuselage is synthesised.

The wings and their spars are added, and then the operating speed boundary is calculated, followed by the lift curve slope. If space had allowed, the remainder of the design (fin, tail, etc.) would have been completed, and drag analysis made. A further design session might have extended this through the remaining aerodynamic analysis (flap effects etc.) weight, stability, control and performance analysis.

In practice many iterative design loops would exist. - a number of attempts might be needed to obtain a suitable fuselage layout, and some manual work would be necessary to check that the allowances for toilets, catering and other areas were adequate. Similarly centre of gravity and stability checks normally indicate shortcomings which require another loop back to the geometric programs.

Apart from technical areas where our programs are as yet sparse, we see the following lines of development:

- We are implementing a 3D, device independent graphics package, developed by the Department of Trade and Industry's Computer Aided Design Centre from work done at Cambridge University (Reference 4). With very simple Fortran statements we can generate graphs, diagrams and pictures which can be sent to any plotter or display device. Similarly any normal cursor or light pen can be used to select data for interaction with a program.
- As computing power reduces in cost we expect to introduce as modules, methods which at present are used only in the Project Definition and Detailed Design stages. For example these will eventually include finite element stress analysis and vortex lattice aerodynamic analysis. Even though not required for conventional designs for which correlations are available, such techniques are already necessary in novel situations e.g. with carbon fibre structures. The existing design framework will make the practical implementation of these methods very much simpler.
- We have not yet used any higher level control programs to run groups of programs. This is a conscious part of our development strategy, in that we wanted to obtain a quick return on our investment by being able to run each program as it arrived, and to test each program very thoroughly before grouping a number together. However, we shall soon be ready to apply general purpose parametric and optimisation control programs (e.g. Reference 5), initially with small groups of programs and later with larger groups.
- A similar approach is being taken in the other areas of design (feasibility and project definition) and links will soon appear upstream and downstream. These links will occur as compatible data files: for example it is very simple to modify a program such as that described in reference 6 (used in the feasibility study) automatically to load the data input files for F.P.D.S.

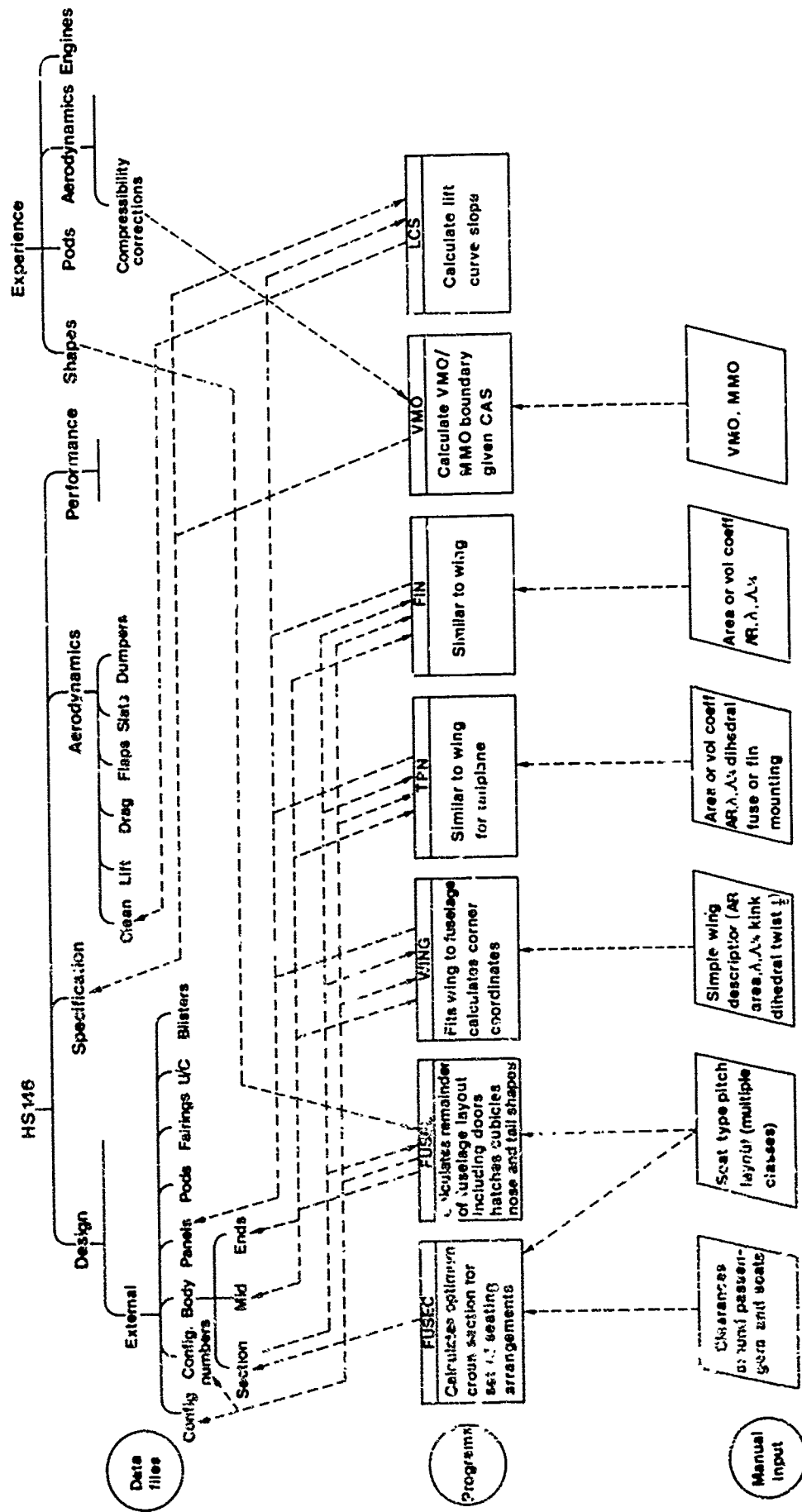


FIGURE 6.

- We have not yet seriously approached the use of F.P.D.S. during the design definition phase for the frequent and rapid assessment of the situation by the type designer. The intention is that the latest estimates of drag, weight etc., (produced from a mixture of tests and predictions) should be fed periodically into a fast evaluation program, which would indicate guarantee margins and enable the type designer to reoptimise any remaining design variables that are not frozen (Reference 7).

- A simple device will be incorporated greatly to improve project control and security. Every time a program is run, information will be appended to an aircraft file giving the date, time, program name and version, input and output data file names and versions, and other relevant data. Over the period of the design, a complete record will automatically be compiled which will give a complete description of the methods and data used in the design. This could be used to reconstitute lost data, to clarify how answers were obtained and to give the type designer information on whether or not work has been done by specialists.

The philosophy of minimal, but specialised documentation is also employed in equipping the designer to use the system. All a potential user requires is:

1. A short tutorial on the basic use of terminals.
2. A user manual, which describes the system philosophy, how to run programs, how to maintain the database and particular running instruction for each program.
3. Reference manuals containing a detailed description of the methods used in each program and of the data file formats.

With the simple interface provided to the designer, education time is measured in hours rather than days, and the main topics are how to make the best of the new techniques (more iteration, data base management) rather than the basic mechanics of the job.

7. COST AND BENEFITS

In designing the system one naturally has to look at the complete operation of the relevant departments. From this study were extracted simplified models of typical design operations such as that shown in figure 7. From these it was possible to evaluate how a system of given characteristics would be likely to affect the performance of the relevant designers, and how it would affect their work pattern and environment. We were not in business merely to speed and increase the flow of useless data to the designers, nor to bury them in a sea of computer cards. Also we did not want to see them reduced to a neurotic state by clattering teletypes nor see them turned into punch clerks. The system has four major aims:

- To reduce the time and effort required to carry out standard calculations, and to do them more reliably and consistently. This of course was the original use to which technical computers were applied.
- To store data cheaply and securely, and to make possible the rapid access of any item of data from a large database.
- To provide a system for the passing of data between specialists with minimum fuss.
- To minimise the effort required to develop and implement new design techniques, and to make these new techniques available to all designers quickly.

The chosen solution is very much a computer based system, to the extent that there is little need for the designer to keep paper records (all data can be rapidly recalled via a terminal). As terminals are still relatively expensive it may initially be cheaper in many cases to maintain a paper file, but this is not the master copy of the data, and can be discarded as terminals and on-line computing become cheaper.

It was found that slightly more than half the members of our project groups would be directly affected by the introduction of the system - the remainder are involved with 'one-off' research and development items (mostly particular problems which exist on a configuration under study) and with administration. The savings in effort realised on a relatively conventional study (where programs exist for all the important calculations) are of course diluted when a novel aircraft is being studied. The dilution is reduced by designing the system such that a particular calculation can be done by hand, with a minimal effort being required to extract the necessary data from the database and to return the results to it later.

These studies led us to the conclusion that, compared with existing methods we should save 30% of the elapsed time, 35% manpower and 20% of cost for a given typical task. These figures should improve with time as more automated control programs become available, but are still dependent on the degree of novelty of the project. The cost savings alone produce a handsome return on investment, with the positive cash flow starting after about three years.

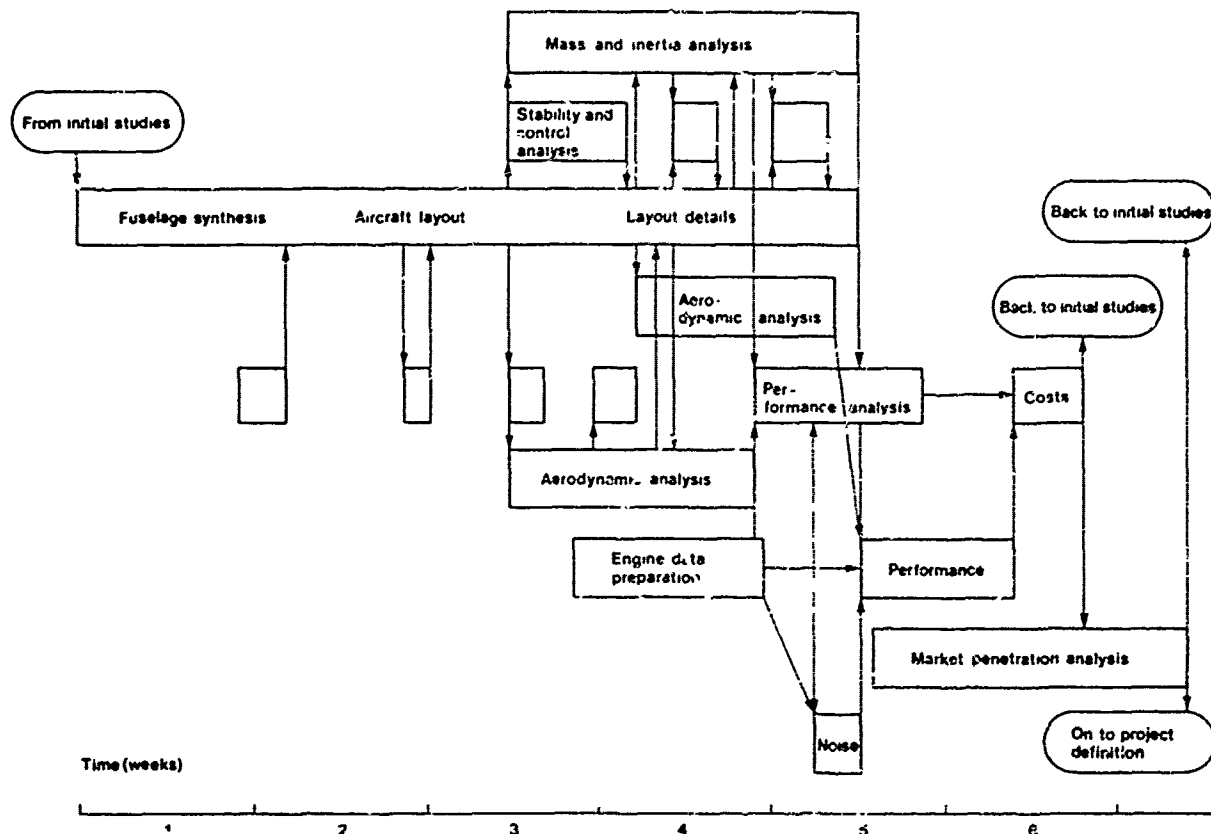


Figure 7. Flowchart of typical project study activities.

In addition there are other less quantifiable benefits which may prove to be more important:

- The designer can, at last, obtain reasonably accurate answers in a realistic time scale to such questions as 'what happens if I move the engines outboard by 0.5 metres'. The usual argument that the designer will 'lose his feeling for the basic calculations' is more than counteracted by the fact that he will gain a substantial feel for the total calculations. In any case, the basic calculation method will be better documented than ever before.
- Each individual program will have been carefully developed and widely tested, so that the reliability of the answers will be greatly improved. We have all experienced programs which have little or no documentation and testing, which seem to work when driven by the right person, but upon which we never care rely too heavily. Also, as many different studies will now use the same calculation method the comparison of results will be vastly easier. This consistency will reduce doubts and shorten timescales.
- The work pattern of a typical designer should improve, as a great deal of clerical work (some calculations, but also most data collection and dissemination) will disappear. Instead more time will be devoted to digesting results and to directing the course of a study, and the specialists will have far more time to investigate the particular problems of a configuration and to develop new methods. We especially hope that these particular benefits will accrue, as it is all too rare to see 'automation' used deliberately to improve the 'quality of life' of hard-pressed designers (Reference 8).

8. CONCLUSIONS

Into this paper are condensed some of the more important results of a major attempt to improve design efficiency in one particular area. Thus far the attempt has proved to be successful, due to technical and managerial approaches from which we believe others may stand to gain.

Within our constraints, we have built a system which should support our Future Projects teams in applying the new generation of design techniques for the foreseeable future. Not only can the designer perform a single calculation, but he can also perform more complex processes and interrogate or administer the database using the same language.

The Project Study stage of design has been chosen for the first implementation of such a system, but the techniques are equally applicable to other stages and are being applied.

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ACKNOWLEDGEMENTS

The work reported in this paper is the outcome of effort and involvement of many managers, designers and computing experts within Hawker Siddeley Aviation Limited. Many of them have also been involved in the preparation of this paper, for which we are sincerely grateful. We acknowledge the permission of Hawker Siddeley Aviation Limited to publish this paper, but the views and opinions expressed are personal, and not necessarily those of the Company.

APPENDIX 1

This is a short example of the use of the 'Future Projects Design System' designed to demonstrate the facilities available rather than a typical design session. The commands are all in their shortest form (as a practiced designer would use them). The quantity of input is seen to be greatly reduced once the 'synthesis' stage is completed and the 'analysis' phase begins, reflecting the gains introduced by the automatic accessing of previously generated data. The time involved is about 35 minutes.

(user input is underlined>)

```

time
invitation to type
16.27.20- IN SEATLAYOUT(+1),T////
- *N 3
- *NS+0,8
- 14 122 0
- 0 152 0
- 0 0 180
- *SA+0,11
- 2 2 0 0 0 0
- 3 3 0 0 0 0
- 5 4 0 0 0 0
- *AA+0,14
- 20 0 0 0 0
- 15 0 0 0 0
- 15 0 0 0 0
- *SP 0,20
- 39 31 0
- 0 30 0
- 0 0 28
- *KI+0,2
- 23 19 18
- *BI +0,2
- 6 2 1
- ////
16.30.32- IN SEATSIZE(+1),T////
- *SW+0,11
- 25 53 81 109
- 21.5 40.1 58.7 77.6
- 18.6 35.7 51.1 67.1
- *CH+0,5
- 15 15.5
- 15 15.5
- 15 15.5
- *SL+0,2
- 28 25 24
- ////
16.32.08- IN CLEARANCES(+1),T////
- *CL+0,11
- 4 1.75 17.09 3
- 3.35 1 8 3
- 1 -1 6 3
- ////

```

Input the file SEATLAYOUT, with a version number 1 greater than the previous highest, to be Terminated by ////

- the data item *N is given the value of 3.
- *NS has elements 0 to 8, with values 14, 122, 0, 0, 152, 0, 0, 0, 180. NS is a mnemonic for Number of Seats. The values correspond to the number of 1st, 2nd and 3rd class seats in three alternate layouts.

- end of file

- Seat Widths for single, double, triple and quadruple seats in each class

- seat Cushion Height, compressed and uncompressed

- Seat Lengths, for each class

Run the FUSELAGE SECTION design program. The program automatically accesses the latest version of the files SEATLAYOUT, SEATSIZE and CLEARANCES.

```

16.33.28- FUSEC
16.33.52 JOB IS NOW FULLY STARTED
- E Z
-
1ST CLASS- 4-ABREAST WITH 1 X 20.0" AISLE(S)
TOURIST- 6 " " " 1 X 15.0" "
ECONOMY- 7 " " " 1 X 15.0" "
- standard output from the program
to the terminal
SECTION MAX. EXTERNAL WIDTH 12FT- 4.0IN (12.33FT)
FRAME LEPTH + SKIN THICKNESS 3.1IN (0.048IN+ GAUGE SKIN)
TRIM THICKNESS 0.8IN
INSIDE RADIUS 76.1IN
FLOOR DEPTH BELOW CENTRE 18.5IN

```

at this point values from the input files may be overwritten. In this case no changes are required, so E and Z are input to Enter the program and then stop.

```

DISPLAY: DESEXTBODSEC(-0) CREATED
DISPLAY: FUSEL(-0) CREATED
END OF MACRO
END OF MACRO
16.35.37- FUSEL
- *CL-2 16 E Z

```

a new version of the file DESEXTBODSEC is created, which contains the fuselage section description.

associated with each program is a file which contains additional output details, and which can be listed at the terminal or lineprinter as required.

```

1ST CLASS- 4-AIRLIFT WITH 1 X 29.0" AISLE(S)
TOURIST- 6 " " " 1 X 15.0" "
ECONOMY- 7 " " " 1 X 15.0" "

```

the same program is rerun, this time with element 2 of Clearances modified to 16.

```

SECTION MAX. EXTERNAL WIDTH 12FT- 2.61N (12.02FT)
FLOOR DEPTH + SKIN THICKNESS 3.11N (0.046IN. GAUGE SKIN)
DILL THICKNESS 0.82N
WHEEL RADIUS 09.41N
FLOOR DEPTH BELOW CENTRE 19.91N

```

```

DISPLAY: DESEXTBODSEC(-0) CREATED
DISPLAY: FUSEL(-0) CREATED
END OF MACRO
END OF MACRO

```

```

16.37.55- FUSEL
- E Z

```

- run the FUSELage Layout program. This uses the same input files as FUSEL, together with the newly-created DESEXTBODSEC.

```

FUSEL MAX. WIDTH 12FT- 2.61N

```

```

CASE 1:-
FUSELAGE LENGTH 116FT- 11.71N
COMPOSING:-
LIGHT DECK 12FT- 6.01N
CABIN 86FT- 1.81N (SEATING 73FT- 3.01N)
T/WL CONE 18FT- 3.91N
DIVIDED STRUCTURALLY INTO:-
FORWARD 21FT- 8.71N
CENTRE (CYL.) 58FT- 7.21N
REAR 36FT- 7.81N

```

```

CASE 2:-
FUSELAGE LENGTH 114FT- 1.71N

```

```

CASE 3:-
FUSELAGE LENGTH 109FT- 2.01N
DISPLAY: DESEXTBODEND(-0) CREATED
DISPLAY: DESEXTBODMID(-0) CREATED
DISPLAY: DESEXTBODFG(-0) CREATED
DISPLAY: DESEXTBODGNM(-0) CREATED
DISPLAY: FUSEL(-0) CREATED
END OF MACRO
END OF MACRO

```

```

16.40.51- IN WINGDATA(+1),1//// - another data file, as required by the WING
design program

```

```

- *TYPE 1
- *S 1493
- *M 6.44
- *R 2.85
- *SLEEPC 35
- *FLIK .34
- *GAMMA 0
- *GAMMA2 2.6
- *XLIST
CANCEL *TWIST 6.5
- *XCATM 58.5
- *WEP 2
- *ICBE .147
- *TCY .131
- *YCT .108
- ////

```

- an incorrect line can be corrected by the CANCEL key.

```

16.43.33- WING
- E Z

```

```

NUMBER OF PANELS = 2
INPUT AREA = 136.704 SQ.M 1493.00 SQ.FT
INPUT ASPECT RATIO = 6.440
WING SPAN = 29.857 M 98.05 FT
MAX. WIDTH OF FUSEL. = 3.724 M 12.22 FT

```

```

IN PLAN VIEW
DISPLAY: DESEXTPAN(-0) CREATED
DISPLAY: ADCEFGWNG(-0) CREATED
DISPLAY: ADDCFGNUMWNG(-0) CREATED
DISPLAY: ADDDERWNG(-0) CREATED
DISPLAY: WINGP(-0) CREATED
END OF MACRO
END OF MACRO

```

the ADD files are those which may be appended to existing files. This is not done except by explicit use of the ADDWING macro or a Copy command. This avoids the appending of unwanted data to an existing correct file.

16.46.29 - ADDWING - a single character can be erased by a ←
 END OF MACRO
 16.47.01 - CY ADDDERING(-0), ELSDE(WNG(+1)
 16.47.32 - SPAR1 - run the SPAR design program
 - *STRT . 196 - all the necessary new data is input at
 run time.
 - *Y1 .971 *Y2 .4265 *Y3 .971
 *R1 .1126 *R2 .324 *R3 .54
 *T1 .022 *K2 .33 *T3 .52 E Z

DISPLAY: DESINIPANSI(-0) CREATED
 DISPLAY: SPARPC(-0) CREATED
 END OF MACRO
 END OF MACRO

16.49.55 - LE SPARPC(-0) - the extra output is Listed from File SPARP

NO. OF SPARS = 3
 SPAR1 HAS 0 KINK(S)
 X(FT) Y(FT) REL. 10 WING LATUM
 INNER: 12.65 -0.25
 OUTER: 18.64 41.50
 SPAR2 HAS 0 KINK(S)
 X(FT) Y(FT) REL. 10 WING LATUM
 INNER: -7.23 -0.25
 OUTER: 0.89 4.80
 SPAR3 HAS 0 KINK(S)
 X(FT) Y(FT) REL. 10 WING LATUM
 INNER: -1.7 -0.25
 OUTER: 20.81 41.50

16.50.48 - MMO - the Maximum Operating speed program is run
 - *MMO 365 *MMO .87 E Z - new data is input at run time

SPEEDS ACCEPTED
 MMO = 365.00 KT.CAS MMO = 0.870
 KINK ALTITUDE = 25714.34 FT. V = 345.08 KT.EAS
 DISPLAY: SUMSPC(-0) CREATED
 DISPLAY: VMOP(-0) CREATED
 END OF MACRO
 END OF MACRO

16.54.30 - LE VMOP(-0) - the extra output is Listed from File VMOP
 MMO = 365.00 KT.CAS MMO = 0.870
 HEIGHT(FEET) V (KTS.EAS)

0.0	364.7
1640.4	363.8
3280.8	363.0
4921.3	362.2
6561.7	361.2
8202.1	360.2
9842.5	359.3
11482.9	358.1
13123.4	356.9
14763.8	355.6
16404.2	354.2
18044.6	352.8
19685.0	351.4
21325.5	349.7
22965.9	348.0
24606.3	346.3
25714.3	345.1
26246.7	341.0
27887.1	328.8
29527.6	316.9
31168.0	305.2
32808.4	293.9
34448.8	282.8
36089.2	271.9
37729.7	261.4
39370.1	251.3
41010.5	241.6
42650.9	232.2
44291.3	223.3
45931.8	214.6
47572.2	206.3
49212.6	198.4

16.56.41- LCSP - run the Lift Curve Slope program
 AIRFIELD: MACHNO .2, LIFT SLOPE 4.34 /RAD
 CAS/MACHNO INTERSECT: MACHNO 0.870, LIFT SLOPE 5.69 /RAD
 DISPLAY: AERCLN(-0) CREATED
 DISPLAY: ADDESDEHWNG(-0) CREATED
 DISPLAY: LCSP(-0) CREATED
 END OF MACHO
 END OF MACHO

16.58.12- LF LCSP(-0)
 EQUIVALENT WING:
 SPAN = 98.06 FT
 AREA = 1546.90 SQ FT
 ASPECT RATIO = 6.22
 TAPER RATIO = 0.170
 THICKNESS: CHORD RATIO = 0.135
 QUARTER CHORD SLEEP = 32.7
 S.M.C. = 15.87 FT
 L.M.C. = 18.54 FT
 YEAH = 18.72 FT
 XEAH = 21.15 FT
 DIAMETER = 7.59 FT
 LIFT CURVE SLOPE V.S. MACHNO:

3.396	0.00
4.156	0.05
4.244	0.10
4.297	0.15
4.341	0.20
4.384	0.25
4.429	0.30
4.470	0.35
4.535	0.40
4.598	0.45
4.669	0.50
4.751	0.55
4.846	0.60
4.955	0.65
5.082	0.70
5.352	0.75
5.941	0.80
5.932	0.85
5.332	0.90

16.59.53- CY ADDESDEHWNG(-0), DESDEHWNG(-0) (APPEND)

17.00.42- ER FUSECP - ERase unwanted files - the extra output files
 17.01.11- ER FUSECP are not part of the data base, and are normally
 17.01.18- ER FUSELP erased.
 17.01.26- ER WINGP
 17.01.38- ER SPAHP
 17.01.46- ER WMDP
 17.01.55- ER LCSP
 17.02.05- LI Log out

**DESIGN EVOLUTION OF THE
BOEING 2707-300 SUPERSONIC TRANSPORT**

Part I
Configuration Development, Aerodynamics,
Propulsion, and Structures

by

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SUMMARY

Detailed cycling of a wide range of potential aircraft configurations against design requirements and criteria characterized the evolution of the preliminary design of the Boeing 2707 supersonic transport. The configuration selection process consisted of competitive evaluation of many variable-sweep arrow-wing aircraft, fixed-sweep modified delta, and fixed-sweep modified arrow-wing configurations. Each broad configuration concept evaluated a multitude of variables with regard to such major items as wing planform, nacelle type and location, and canard and/or aft tail arrangement and geometry. Promising configurations were thoroughly analyzed with comprehensive wind tunnel testing and in-depth engineering analyses within all technical disciplines. The preliminary design process deeply involved the development of new analysis methods and the adoption of new approaches to commercial airplane design to achieve maximum performance and operating efficiency. The final configuration selection represented the best compromise between achievement of optimum supersonic cruise L/D, minimum structural weight, optimum low-speed performance and minimum community noise.

This paper will trace the history of this design activity, noting areas where failure and/or misconception occurred due to insufficient knowledge, and relating how such shortcomings were corrected or avoided in subsequent configuration evaluations. Finally, attention is invited to selected areas where preliminary design tools could be improved.

INTRODUCTION

Any review of current preliminary design methods for modern multipurpose aircraft must include the cancelled United States supersonic transport prototype program, particularly the methods and constraints which led to the Boeing 2707-300 configuration. The unique reasons for the importance of a review of the methods used in this configuration selection are (1) this program was a classic example of a purposely chosen high-risk technology effort, and (2) the program would have consumed over \$1 billion in funds in the prototype phase, and did expend over 26,000 wind tunnel hours, and a considerable amount of one contractor's energy for a period of over 14 years. The Boeing engineering manpower alone at the time of program cancellation was approximately 3000 employees, and 70% structural drawing release to manufacturing had been readied, with the first major aircraft structural wing panels already completed. Thus, although the program was unfortunately terminated, a case history of the preliminary design activity which led to the 2707-300 configuration is felt to be of value when considering possible revisions to methods in use today to exploit advanced technology into new complex aircraft systems.

BRIEF HISTORY

The Boeing SST effort began in 1957 at the conclusion of the B-70 competition, and at a time when the B-70 was first being readied for commercial service. Initial efforts were at Mach 3.0, using a fixed-delta all-steel configuration with a canard for longitudinal control. As shown in figure 1, this effort was soon expanded to encompass the arrow-wing and variable-sweep concepts in early 1960 and to broaden the speed range between Mach 2.6 and 3.0. Titanium was introduced during this period to offset at least a 12% weight penalty required by high-strength steel alloys. At President Kennedy's suggestion in mid-1963, NASA, with assigned ground rules, started initial efforts in a national program through competitive studies of four uniquely different configurations shown in figure 1. A year later, President Johnson designated the FAA (later the Department of Transportation SST Program Office) as the responsible agency to conduct a competition for a full-scale preproduction prototype program. Boeing chose to concentrate mainly on the variable-sweep concept at that time because of its opportunities for improved low-speed performance and consistency with existing airline airport operations.

During the next 3 years, many three- and four-engine arrangements were compared, with substantiating wind tunnel data, against continually revised design criteria and operational requirements, as noted in figure 2. Smaller competitive efforts in modified delta- and arrow-wing configurations were carried out at the same time.

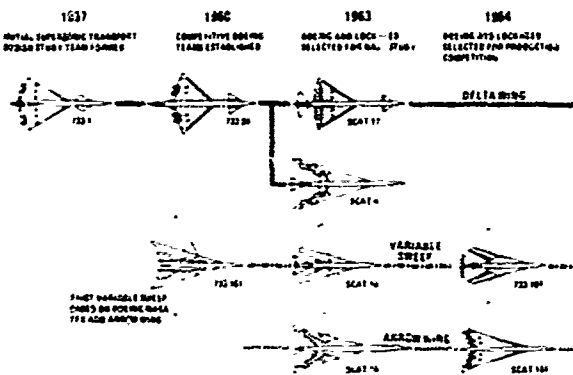


Figure 1.—Boeing SST Program (1957-64)

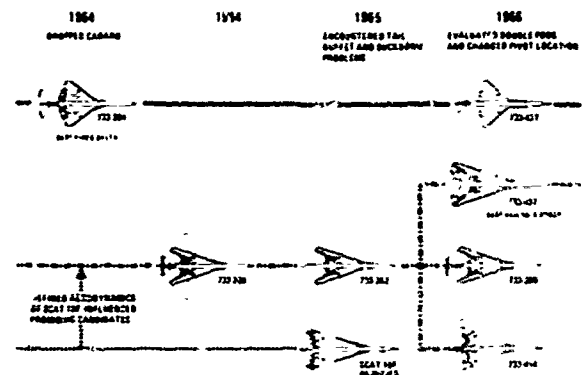


Figure 2.—Boeing SST Program (1964-66)

Finally in late 1966, the competitive variable-sweep configurations were narrowed down at Boeing to one integrated wing-tail arrangement, the 2707-100 with engines aft (see fig. 3). This concept evolved from hot-gas tail impingement at high angle of attack at supersonic conditions, tail suckdown at takeoff, and locked-in pitch-up experiences encountered with body- or a wing-mounted engine installations on all previous configurations which employed horizontal tails. Because of strict adherence to handling qualities and stability established by subsonic jet design practice, it was found impossible to avoid either a horizontal tail or a canard with the long-body 250-plus-passenger configurations under consideration.

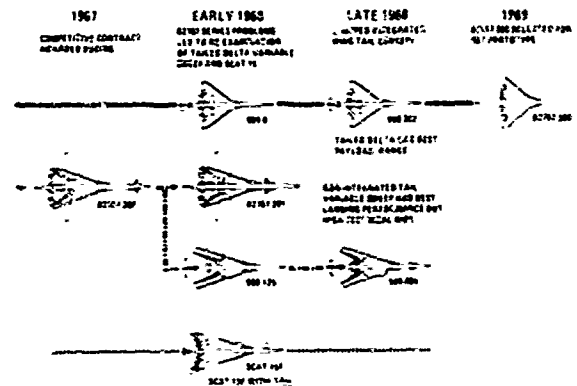


Figure 3.—Boeing SST Program (1966-69)

When the 2707-100 configuration was selected by the U.S. Government in 1966 for further development, a period of 9 months was established to validate the concept before a design go-ahead was to take place. Detailed refinement took place, supported by testing in all critical areas, including a full-scale fail-safe pivot. For the first time, a fully integrated aeroelastic analysis of an SST configuration was executed in sufficient depth to obtain realistic structural sizing needed to support releases to manufacturing. At this time, Boeing was designing a very slender aircraft over 300 feet long (football field length). Aeroelastic deflections under limit load were found to cause changes in pitching moment of the same order as the rigid data itself. Hence, it proved necessary to account for the complete impact of all significant inputs to the aeroelastic analysis, including hydraulic fluid elasticity, before proper fixed and movable surface sizing could be made. The ineptness in the existing computer systems became evident. There were lengthy turnaround delays for a single cycle of critical data intended to establish whether the configuration aeroelastically converged, due to design changes, within the established design criteria.

It became apparent after the allotted 9 months for configuration firm-up that even with the addition of a canard (with its attendant weight, complexity, and added body stiffness) to achieve reasonable aeroelastic forgiveness, the weight of the final product yielded a noneconomic commercial aircraft when compared to the original Boeing SST program goals. With DOT-SST project office concurrence, it was decided to reopen the configuration selection process for 1 year under rigid government and Boeing review controls, and to accelerate the ability to cycle given designs and conduct trade studies on configuration alternatives. It also was decided to separate the prototype and production programs to allow the prototype design to proceed with added technical risk to support a better performance objective. This decision was based on a sober understanding of the aeroelastic influence on ultimate design.

During 1968, a comprehensive reevaluation of configurations was conducted in a "hot room" environment. This included modified arrow-wing concepts (SCAT 15F), modified deltas (969-6), and variable-sweep alternatives (both inboard and outboard pivot arrangements) aimed at reduced weight. An instantaneous status of each configuration, compared to the competition, was available. Each concept was evaluated against the constantly revised design and safety criteria and operational requirements.

Finally, the competition narrowed down to a moderately outboard-pivot twin-pod design (969-404) and a modified delta (969-302) which employed separate pods. Both configurations by this time had chosen artificial stability in all axes as needed to maximize performance. In the ultimate, due to its simplicity and better understanding, the 2707-300 was born—approximately 12 years after the Boeing SST effort started.

The low-speed operational goals of the arrow-wing concept, including noise, could not be realized, and the weight and aeroelastic properties of the required large wing area resulted in an uncompetitive design, even though the supersonic aerodynamic efficiency was the best of all configurations.

This paper will now review the developments in technology, based on the targets selected, which occurred during the progress of the SST program, and the technological conflicts which developed as that design proceeded to completion. This technical data is presented only to illustrate the configuration integration and management tasks which resulted as this high-technology design was evolved.

TECHNOLOGY ASPECTS

From the very onset, the Boeing SST program aimed at achievement of maximum technical performance on the premise that attractive airline economics could only be achieved by that goal. It was soon realized that it would be more attractive to design a high-risk prototype first and to make changes in the production design based on prototype costs, flight test, and airline evaluation.

Although the purpose of this paper is to emphasize design integration and management, the advanced technical features dominated design decisions and so merit special consideration here.

Configuration Aerodynamics

The prime conflict in competing configurations throughout the program dealt with operational constraints and objectives in the low-speed mode with those at supersonic speed. The excellent supersonic and high-lift characteristics of the very low-aspect ratio, highly swept slender wing were well understood. Such a wing has low drag at supersonic speeds and can generate high lift at low speed by creating strong vortices at its sharp leading edge. It was clear, however, that lift generation with leading-edge vortices in the takeoff and landing modes would make it difficult to achieve low noise goals. Further, it had been assumed that sonic boom would make overland supersonic flight undesirable, and achievement of high subsonic aerodynamic efficiency then became an important design objective. This ruled out the very low aspect ratio of the slender wing and the search was launched for a different type of wing planform.

The highly swept arrow-wing enthusiasts supported the subsonic leading-edge configurations and variable sweep concepts to maximize supersonic performance, as well as subsonic performance. Those who favored a compromise to retain the known flight characteristics of contemporary military supersonic airplanes and those more nearly associated with the existing subsonic transports leaned toward the modified delta concepts; see figure 4.

Figure 5 shows the supersonic maximum L/D comparisons obtained for various wing geometries considered during this program. For the same wing area and thickness ratio, the highly swept configuration demonstrated (L/D) in the 9-10 range at the cost of increasing structural aspect ratio for a given span. At subsonic speed, the L/D increases with span for constant wing area as shown in figure 6 but when the sweep of the wing is high the cost of span in terms of wing weight is also high. Figure 7 shows a preliminary assessment made in 1961 of aircraft employing the competing concepts based on available data at that time. It showed that at a maximum payload/range requirement the operating field lengths of interest would not be met efficiently with the fixed-arrow-wing concepts, whereas the debate between the variable-geometry and the modified-delta-wing concept probably could be resolved only through complete design integration.

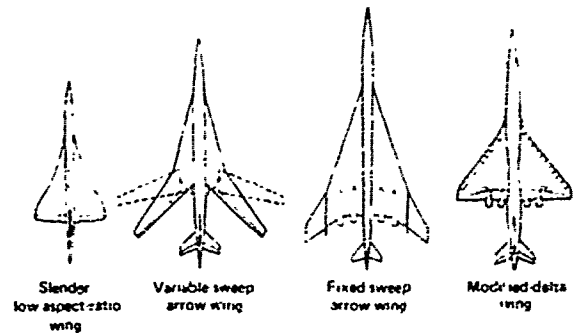


Figure 4.—SST Wing Planforms

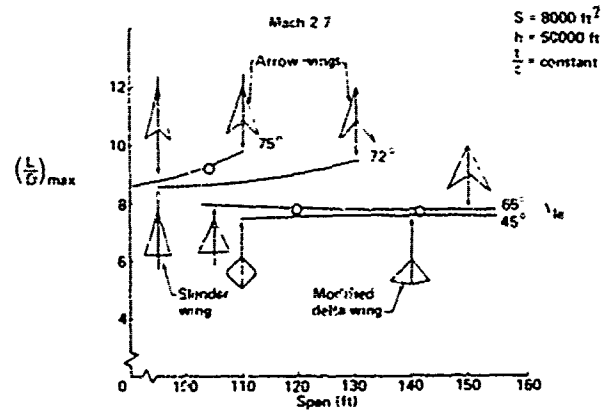


Figure 5.—Airplane Supersonic Efficiency

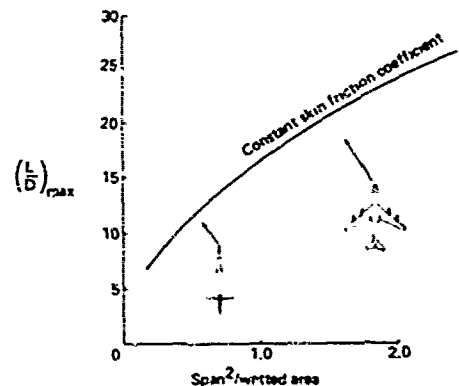


Figure 6.—Effect of Span on Aerodynamic Efficiency at Subsonic Speed

Once it was assumed that variable wing sweep would produce the necessary span for efficient subsonic flight and acceptable takeoff and landing performance, supersonic flight efficiency favored the highly swept-arrow wing. For a given thickness and cruise Mach number, the higher the sweep the lower the wave drag. Conversely, for a given cruise drag, the thickness of the wing could be greater, and it was hoped that this effect would offset the anticipated structural weight penalty and fuel volume limitations of variable sweep. Although thickness and sweep are important in the determination of wave drag, the distribution of thickness is also important, as shown in figure 8. As it turned out, the features necessary to accommodate variable sweep in detail had a tendency to influence this effect adversely. Furthermore, with the variable sweep pivot in the wing, an important loss of useful wing volume resulted so that additional thickness had to be provided to recover this volume for fuel. The highly swept arrow wing derives a major portion of its supersonic aerodynamic efficiency from its potential for low drag-due-to-lift factor, as shown in figure 9. When the sweep is high and the flow at the leading edge subsonic, it is possible to shape the wing so that the outboard portions are tailored to operate in the flowfield of the inboard portions of the wing. As shown in the figure, this tailoring is essential because in its absence a drag penalty occurs that more than cancels the potential benefits in drag due to lift of the highly swept-arrow wing. Factors that affect this problem strongly include aeroelasticity of the wing as well as the geometric constraints of variable sweep, and it became difficult to arrive at a satisfactory solution.

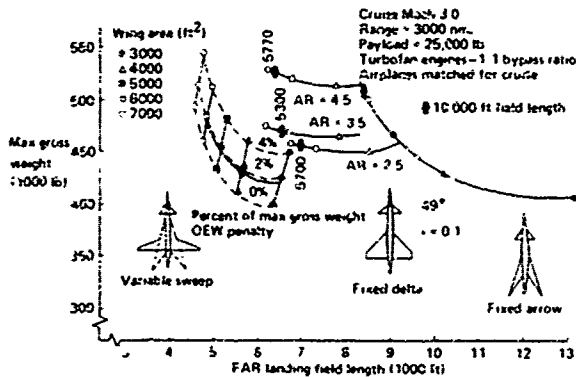


Figure 7.—Effect of Configuration on Low-Speed Performance

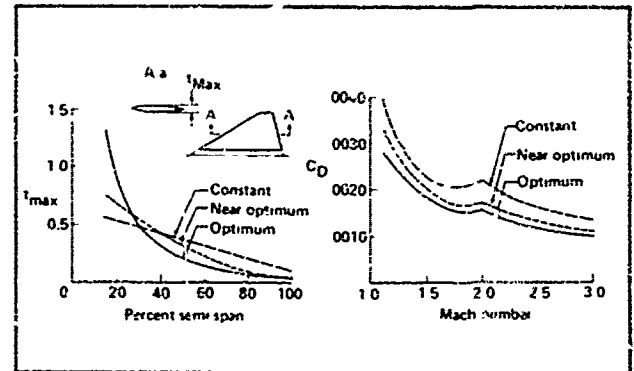


Figure 8.—Wing Thickness Development

From the point of view of supersonic drag, the ideal airplane should not have a separate trimming surface such as a horizontal tail or a forward canard. Trim at supersonic conditions is then achieved best when the center of gravity is near the aerodynamic center and when the lift distribution is such that what is best for low drag due to lift also satisfies the balance of longitudinal moments. In other words, it is necessary for the basic wing to have a positive pitching moment at zero lift. In the SST design it was found, however, that the design goals associated primarily with airport performance, noise, and subsonic operations in general made it desirable to add to the configuration an additional trimming and control surface. This became doubly necessary for the highly swept wings because these wings exhibit loss of stability due to aeroelastic effects at virtually all flight conditions so that the additional surface was needed to provide more stability. The greater the size of the additional trimming surface, the more delicate the requirements became for a totally optimized longitudinal lift distribution to ensure that the additional surface carried its share of the lift distribution. There was a limit to this, however, because the shape of the wing that provided the necessary values of pitching moment was poor from the point of view of zero-lift angle. Everything else being the same, such a wing required a longer landing gear for constant performance, so that the efficiency of the airplane tended to suffer rapidly if wing pitching moment requirements became excessive. Accordingly, the best tail size was the smallest practicable. This was found to be more likely to occur when the requirement that determined the size of the tail was one of only control power rather than one of stability. Critical flight control conditions for all large SST configurations occurred at low speeds and were compounded when wing flaps were used to improve low-speed performance. These requirements forced consideration of additional control surfaces on all configuration concepts—the size being larger generally on variable-sweep concepts because of aeroelasticity and generally smaller on delta wings because of the lack of a compelling stability requirement.

Nacelle placement was a key development early in the program. Both dual and isolated pods were considered. Through proper contouring of the external pod geometry, tailoring of the wing, and integration with proper consideration for area rule principles, it was found that the pod/wing combination could be designed so that the incremental drag of the pod was equal to the wetted area drag of the pod alone. This is illustrated in figures 10 and 11.

A nacelle location conducive to low drag is one near the trailing edge of the wing, producing favorable lift and drag interference and with the engine nozzles behind the wing trailing edge. Unfortunately, such an engine arrangement directs the airplane center of gravity toward the rear of the airplane, compounding the stability problems mentioned previously.

Having achieved desirable engine positions and tail size, the vertical location of the horizontal tail presented a basic problem. The tail had to be moved up to avoid jet blast, leading to serious longitudinal stability problems. At supersonic cruise, attitude and speed, the airplane jet plume tended to bend into the flight direction unlike that of subsonic aircraft. This was

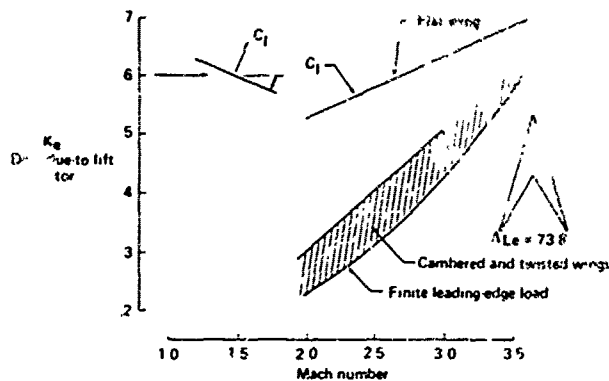


Figure 9.—Theoretical Drag-Due-To-Lift Factors of Highly Swept Wings

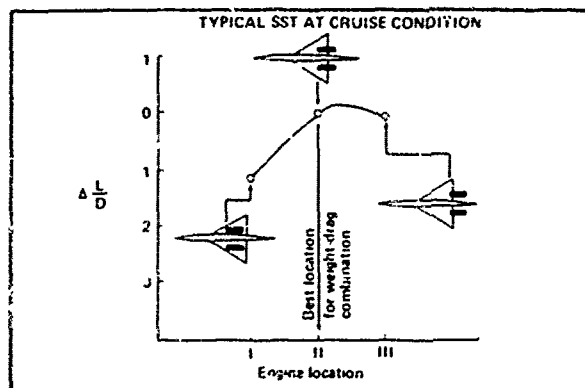


Figure 10.—Effect of Engine Location

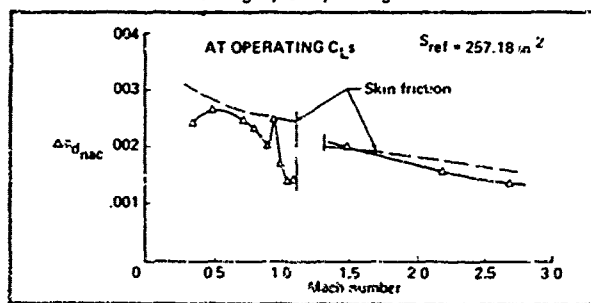


Figure 11.—Nacelle Drag Vs Mach Number

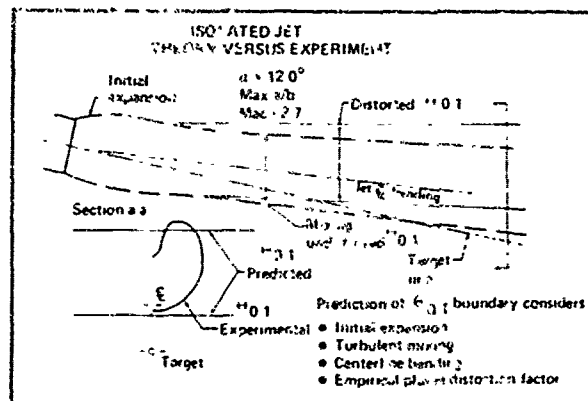


Figure 12.—Tail Exhaust Disturbance

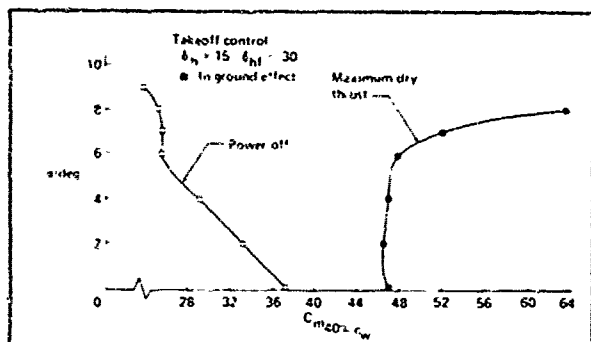


Figure 13.—Takeoff Thrust Effects on Longitudinal Stability

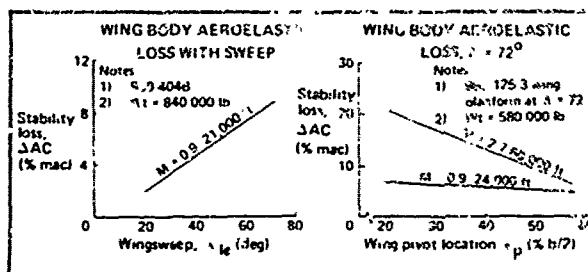


Figure 14.—Aeroelastic Shift in Aircraft

borne out in wind tunnel powered model tests. Thus when the aircraft flew at high angles of attack, the tail became engulfed in high-temperature exhaust gas, unless the tail was raised or the tail arm was reduced to avoid the problem. This problem existed on all variable sweep designs with inboard pivot and with wing-mounted engines. Figure 12 shows the thermal pattern on one variable-sweep concept at an angle of attack of 12°. Raising the tail required an increase in tail area to counteract the unfavorable high-angle-of-attack effects. The larger tail, however, was more susceptible to lockdown caused by the effect of the jet flow near the ground during the takeoff maneuver. Figure 13 shows the undesirable effect of power on the longitudinal stability of an inboard pivot low-tailed variable-sweep configuration at takeoff conditions. Variable-sweep configurations, with their requirements for larger tails, suffered particularly from these problems. Raising the tail even higher was not a practical solution because of high-speed drag and locked-in stall considerations. These problems were present on all configurations, except for those with tail-mounted engines. But the penalties attendant with compromise solutions were far less critical on the modified-delta concepts, where there was more room for spanwise engine movements and where the need for a large stabilizing horizontal tail was less acute.

Aeroelastic loss in stability constantly plagued all designs because of the large size of the aircraft. Figure 14 shows the forward shift in aerodynamic center with aeroelastic correction for 1g flight as a function of leading-edge sweep and wing pivot location for two variable-sweep concepts. Similar corrections were required of all configurations but to a lesser degree as the leading-edge sweep was reduced.

As the work progressed, it became apparent that constraints other than those of supersonic aerodynamic efficiency would dictate the final configuration. Most noticeable were those of noise, approach speed, landing field length, aeroelastic effects, allowable cg range, engine location, and basic stability and control requirements. These constraints affected the variable-sweep concepts to such an extent that all the benefits of high sweep at supersonic speeds and high span at subsonic speeds were compromised. Considering fixed-wing configurations, the need to retain high span made it necessary to limit the wing sweep to avoid excessively large structural aspect ratios (such as those obtained with arrow-wing configurations) with attendant structural and aeroelastic problems. The medium-sweep, moderate-aspect-ratio delta wing was thus finally chosen.

Propulsion

The integration of the engine in the SST configuration had a set of constraints which constantly pressed the technology limits. It was desirable to mount the engine pods under the wing at the trailing edge to obtain a maximum favorable interference effect between the pod and wing and to minimize angle-of-attack effects on the intakes. This led to considerations of wing boundary layer ingestion, intake flow field distortions, and interference between inboard and outboard intakes. To avoid having an inadvertent unstart in one intake trigger an unstart in an adjacent intake, interference between inboard and outboard intakes must be minimized.

Pod placement was constrained by the need to avoid excessive empennage heating due to inboard locations, flutter and flight control consequences for outboard locations, sensitive drag effects, thrust reverser effectiveness, impact on landing gear length, and interference between the pods and the landing gear. For configurations incorporating variable-sweep wings, keeping the pods inboard of the pivot magnified these problems.

Single pods with axisymmetric intakes and dual pods with two-dimensional intakes were both considered. Figure 15 shows each concept on a modified-delta platform. The single-pod, axisymmetric intake concept was selected after a careful consideration of the foregoing factors and for reasons of safety. With the pods below the wing, the intakes were subject to a complex flow field generated by the wing and body, which varied considerably with pitch and yaw in supersonic flight. Inboard pod locations were subject to large variations in apparent boundary layer depth, while outboard locations were subject to relatively large mach number and flow direction gradients. Figure 16 shows one of the earlier flow field measurements to illustrate the Mach number gradient which existed. Modifications to the wing lower surface contours were conducted (fig 17), resulting in the improvements shown in figure 16. Actual intake testing showed the revised flow field to be satisfactory.

Similarly, evaluation of intake unstart effects due to control malfunction, engine failures and seizures, and external disturbances were evaluated on models, illustrated in figure 18. These tests, in conjunction with boundary layer measurements, defined minimum allowable pod spacing, intake diverter heights, and the size of fences required between neighboring intakes to ensure a low probability of having one intake unstart trigger an unstart in an adjacent intake.

Throughout the early configuration developments, the relationship of the jet plume to the horizontal tail was under constant surveillance to ensure that no adverse interactions (thermal or aerodynamic) could occur during takeoff rotation or at high-alpha supersonic conditions. Figure 19 shows a hot jet survey model in the supersonic tunnel on an early configuration, and figure 20 shows thermal data obtained on this model at various maneuver load conditions.

It was also found that a major effort was required to integrate the engine, intake, and flight controls to meet operational and safety constraints. A prime example was the roll-yaw coupled maneuver that was induced whenever an intake unstart occurred. At high supersonic speeds, certain combinations of airplane attitude and engine failure/intake unstarts could result in an unsafe maneuver, if control integration was lacking. By employing intake unstart signals and nose-mounted accelerometers to initiate flight control responses, and by causing sympathetic restarts of opposite intakes in the case of severe engine failures (e.g., seizure), unsafe maneuvers could be avoided.

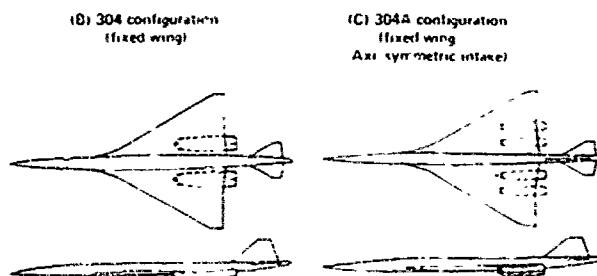


Figure 15.—Dual Versus Single Pods

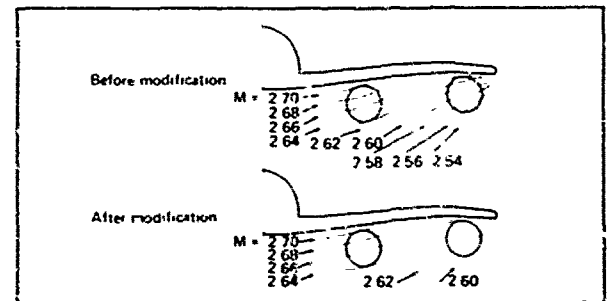


Figure 16.—Adjusted Flow Fields

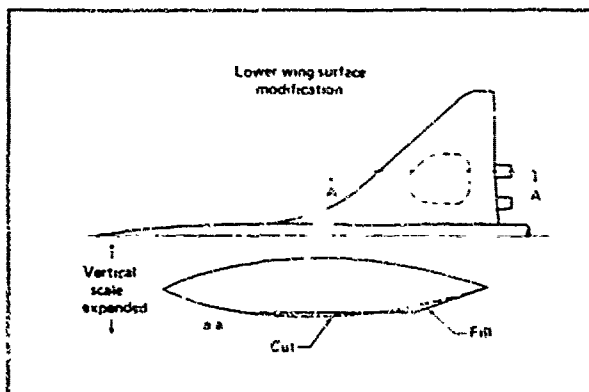


Figure 17.—Wing-Flow-Field Modification

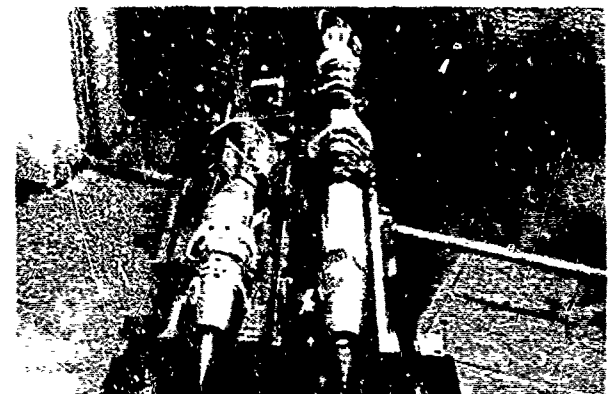


Figure 18.—Intake Interference Model



Figure 19.—Hot Model Test

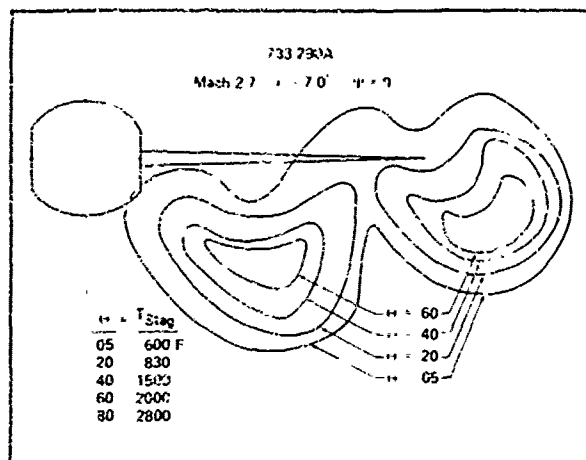


Figure 20.—Thermal Profile Near Tail

The mach 2.7 cruise speed dictated a mixed compression intake for efficient operation. Initially, there were a variety of engine offerings (dry and afterburning turbojets, and duct-heating turbofans) with differing flow characteristics. A variable-diameter centerbody intake concept (fig. 21) was initially chosen for its ability to match the airflow of each engine offering. After extensive design and testing, it was found that internal leakage and boundary layer bleed could not be reliably controlled, and a more conventional translating centerbody design was chosen for further development (fig. 22). By this time, the GE4/JSP afterburning turbojet had been selected as the project engine enabling an intake design to be selected.

The maximum demand airflow for the GE4/JSP engine is shown in figure 23, along with the airflow supply characteristics of the translating centerbody intake. It can be seen that translation of the centerbody alone would result in airflow starvation of the engine at transonic flight speeds due to mechanical constriction. Further complication to the intake, in the form of throat doors in the cowling, relieved this problem. However, the loads and weight of the intake increased substantially due to this added complexity.

The internal flow concept of the pod is illustrated in figure 24, which shows that a careful accounting of bleed, spillage, leakage, external by-pass, and internal by-pass and cooling flows was necessary to ensure accommodation of engine flow demands, accurate drag assessment, adequate environmental control system heat sink, engine cooling, and an opportunity for maximum nozzle performance at all operating conditions.

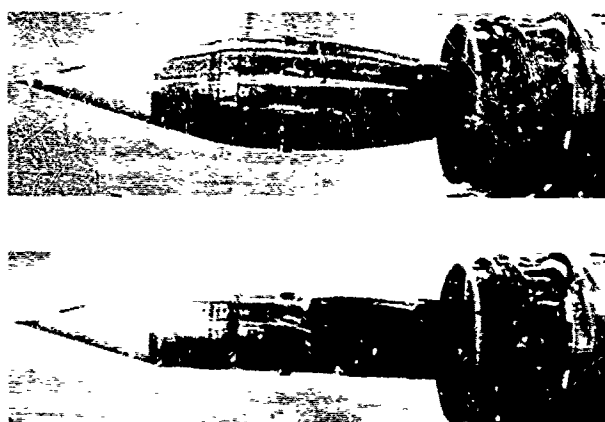


Figure 21.—Variable Diameter Centerbody Intake

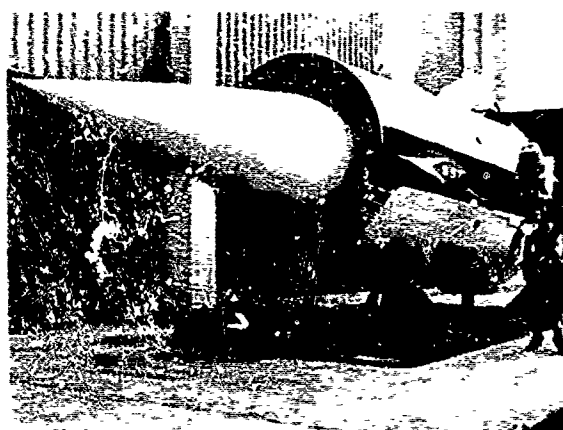


Figure 22.—Translating Centerbody Intake

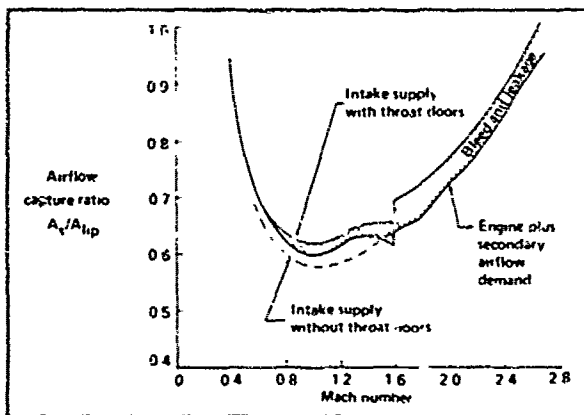


Figure 23.—Intake/Engine Matching

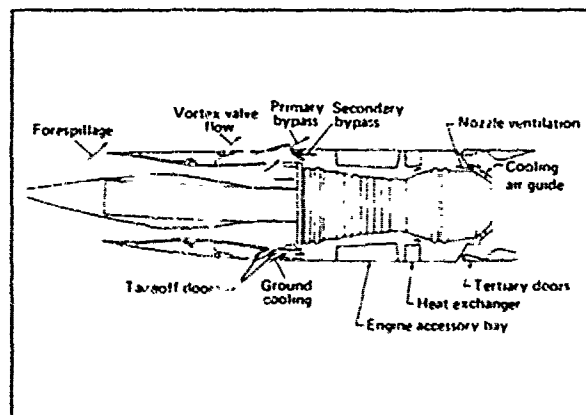


Figure 24.—Integrated Propulsion System

Initially, a convergent-divergent (C-D) ejector nozzle was employed. However, increasing concern regarding engine weight and noise led to the adoption of a two-stage ejector nozzle (TSEN). Figure 25 shows a model of this nozzle in the wind tunnel. The TSEN nozzle was lighter and more adaptable to a jet noise suppressor installation, although wind tunnel tests showed the low-speed performance of the C-D type was superior.

From the beginning, noise control strongly influenced the aircraft configuration development, as well as the pod design and integration. Noise considerations set the requirements for wing span, flaps, and the use of a horizontal tail. The intake and its controls were designed for choked operation at part power to reduce forward arc noise influence on the community. However, the design requirement for noise was an ever-changing target throughout the SST program. In these circumstances, control of the configuration development was most difficult to accomplish because even small changes in allowable noise levels would cause the configuration requirement to change.

Sideline noise at takeoff was a major problem, and much of the effort to control it was focused on suitable jet suppression and carefully selected operational procedures. Increases in engine takeoff airflow and the addition of spade suppressors resulted in some reduction, but these improvements did not keep up with the moving noise target. When the program was cancelled, Boeing was developing the high effectiveness multi-tube suppressor shown in figure 26. At that time, other nozzle/suppressor concepts, together with major revisions to the engine, were also under consideration for the production airplane.

Materials and Structural Concepts

A continual evolution in technological demands occurred throughout the SST program in the areas of materials and structural concepts. At the onset, superalloy and stainless steels were employed, and the major efforts were concerned with environmental behavior and long fatigue life at Mach 3.0. Skin-and-stringer construction were conventional while an insulation material was in development to provide the proper heat barrier. Commercial life of 50,000 hours for primary structure, adequate thermal insulation, and suitable fuel tank sealant were the elusive goals.

At the same time, interest in titanium alloys was developing to improve airplane weight ratio and reduce aircraft cost. Sheet titanium of the 8-1-1 category was evaluated initially. But the fracture toughness in a salt environment (see fig. 27) of such materials failed to fulfill the SST specification, and a change was made to 6-4 titanium as the primary structural material.

By 1964 Boeing decided to reduce the flight speed to Mach 2.7, principally to avoid the costly superalloys, and to reduce basic systems, insulation, and sealing costs. It was determined also that a structural weight reduction of at least 12% would be obtained through the use of titanium as the basic structural material. Figure 28 compares the incremental payload increase of the 2707-300 and several existing aircraft for a reduction in OEW. The SST benefits nearly twice as much as a conventional subsonic vehicle designed to the same range. Further, by reducing speed from Mach 3.0 to 2.7, the payload/gross weight ratio is improved as shown in figure 29, principally because the fuel burned is proportionally less at the lower speeds.

During the period from 1964 to 1967, when the heavy emphasis was on high wing-loading variable-sweep configurations, titanium 6-4 was used principally in conventional skin-stringer arrangements. Body sections also used integrally machined components where internal loads suggested a favorable weight trade. Noncritical structural elements were composed of limited use of diffusion-bonded honeycomb and polyimide-bonded honeycomb (with or without polyimide core). Such components as flaps, control surfaces, fairings, and tail cone employed these honeycomb concepts to reduce weight and cost. However, the basic loads in such components were quite low, and the main unknown at that time was their resistance to environmental conditions. Testing was initiated to determine the impact of environmental conditions on these materials.

The main structural design effort was concentrated on the heart of the variable-sweep structure, the multiple load path pivot. This effort was to minimize its weight, size, and volume to take full advantage of the cambered and twisted arrow-wing configuration. Figure 30 shows a full-scale pivot being tested in a dynamically loaded environment under thermal cycling.

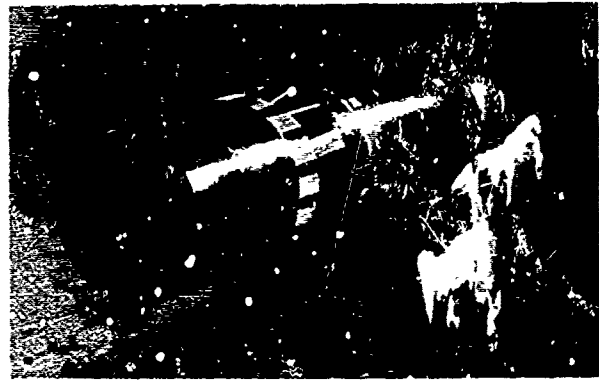


Figure 25.—SST Nozzle in Tunnel

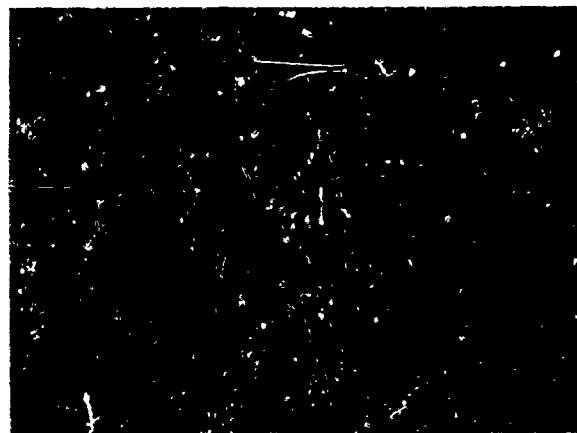


Figure 26.—Tubular Suppressor

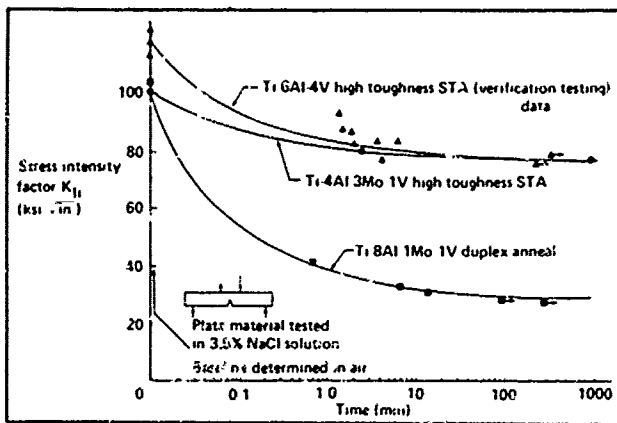


Figure 27.—Salt Solution Crack Growth Characteristics

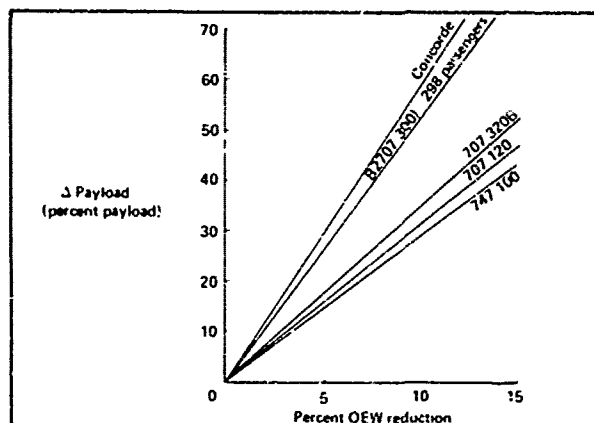


Figure 28.—Cost of Structure and Systems Weight for Speed

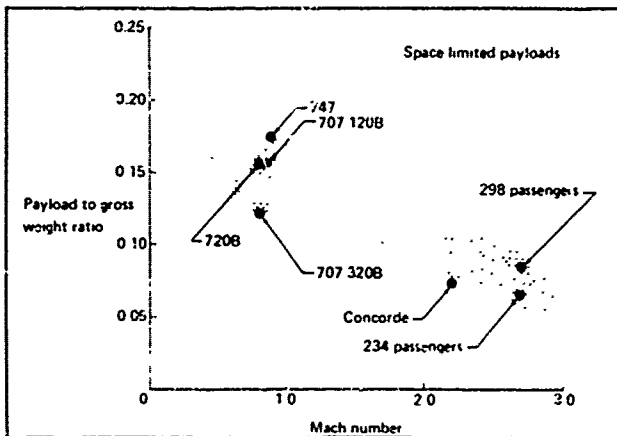


Figure 29.—Effect of Speed on Payload

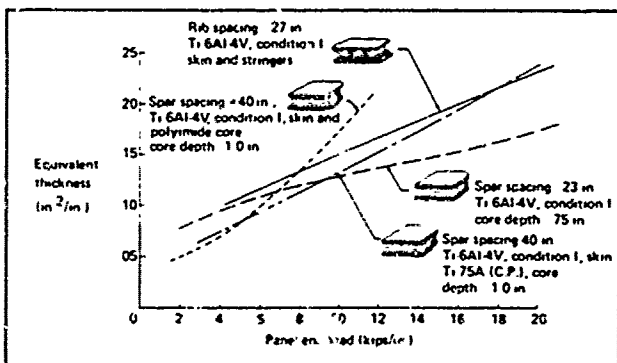


Figure 31.—Panel End Loads

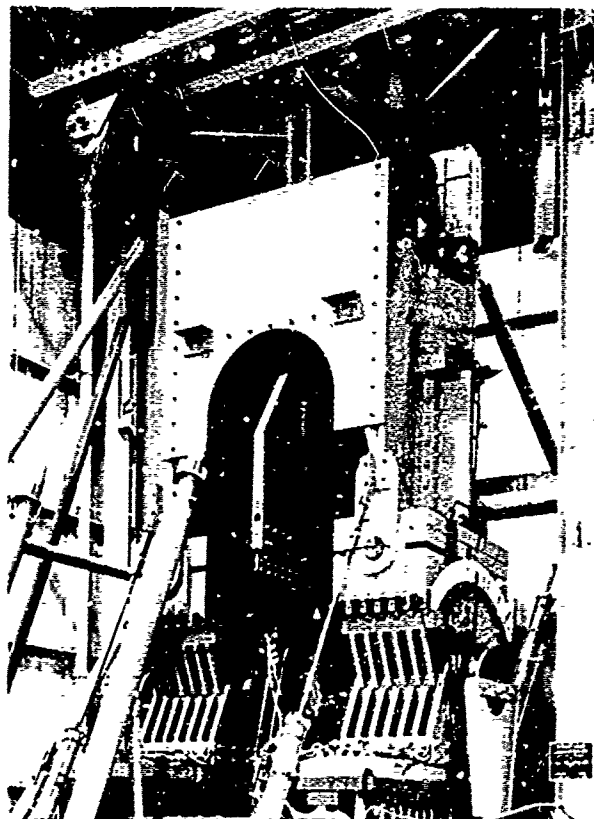


Figure 30.—Wing Pivot Test

When it was finally concluded that a fixed-wing configuration was a serious contender and that low wing-loading was required for low speed, the structural requirements became even more significant. The primary wing structure was no longer subjected to high end loads, and titanium honeycomb was used to make the wing more structurally efficient. To illustrate the increased use of honeycomb in primary structure, figure 31 shows a comparison of the equivalent thickness (measure of structural efficiency) versus panel end load of various structural concepts using titanium.

At first, attempts were made to use polyimide core bonded to titanium face sheet as much as possible on the wing primary structure, as well as for secondary structure. Wing tips and strake areas in particular were logical candidates. However, it was found from environmental tests (1-inch ice balls, for example, at Mach 1.2) that the polyimide bond and core failed to meet specifications. This choice of concept in primary structure gave way to a combination of manufactured and welded corrugated core and diffusion bonded honeycomb. Each of these concepts finally yielded to an aluminum-brazed honeycomb as primary for use on the prototype airplane. Many of these decisions were based on unsatisfactory structural component allowables from static and dynamic tests which occurred prior to final sizing of the structural elements. Figure 32 shows a sample of the aluminum-brazed honeycomb construction, and figure 32 illustrates a full-scale buildup element as planned for use on the airplane. Figure 34 is a schematic of the general structural system of the 2707-300 prototype.

It was planned to continue efforts on further extensive use of honeycomb, and to introduce diffusion-bonded honeycomb, with its better insulation characteristics, additionally, the specifications for this product were better understood.

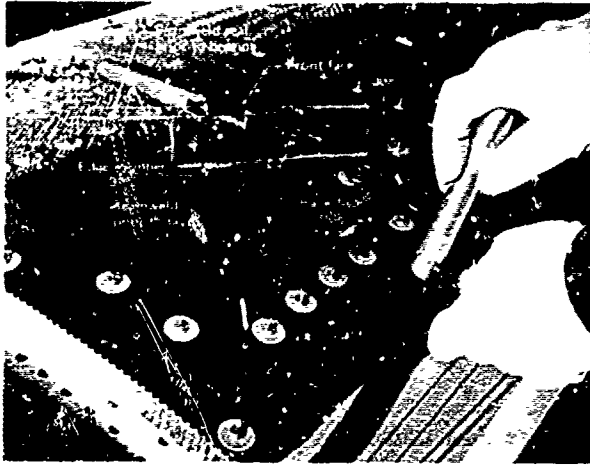


Figure 32.—Braze-welded Titanium Honeycomb

Low structural weight was paramount in this design. All-out attempts at development of honeycomb of varying core density and face sheet thickness throughout major structural members occurred in an effort to ensure that components were either load-, stiffness-, or thermal-, designed and that no excess weight was carried in a member unless a cost-effective decision could be made.

Cost-effectiveness was of paramount importance throughout design of the structural system. To accommodate such a study, the aircraft as it finally took shape in the 2707-300, was divided into sections. Those sections in the aft portion were given more credit for weight reduction than the forward end, because an SST balances tail-heavy, and ballast was required at all payload conditions. Hence, any weight reduction in the aft fuselage or empennage also reduced ballast requirements to meet the aft cg loading limits for safety of flight.

Control of the design integration on the SST was seriously contested as changes to structural materials and elements had to take place due to technology successes and failures in this area. These events not only affected panel weights and structural sizing for load design elements, but also strongly influenced the elastic properties of the aircraft, including flutter and its consequences.

Structural Dynamics, Loads, and Flutter The preliminary design portion of any airplane program is relatively short and does not allow time for a complete structural analysis. Consequently, for a high-performance airplane in which structural flexibility is a prime factor, the full implication of structural analysis at design go-ahead is not well understood.

To illustrate the impact of aeroelasticity on airplane design, ingredients that go into an integrated structural design analysis must first be understood. The fundamental forces that contribute to the aeroelastic solution are illustrated in figure 35. As depicted, the forces may combine in different ways, depending upon the particular airplane flight regime or maneuver that is being analyzed. It is fairly evident that these forces and the equations representing their interaction can be approximated in various degrees of sophistication, depending upon the allowable solution time and degree of airplane definition available. In the case of typical subsonic aircraft with relatively thick high-aspect ratio wings and fuselages with lower fineness ratios, the structural approximation can be represented simply by a beam type analysis without requiring detailed definition of internal structure (see fig. 36). The results of this simplified analysis will not differ greatly from final design analysis and will converge to an acceptable accuracy within three or four iterations.

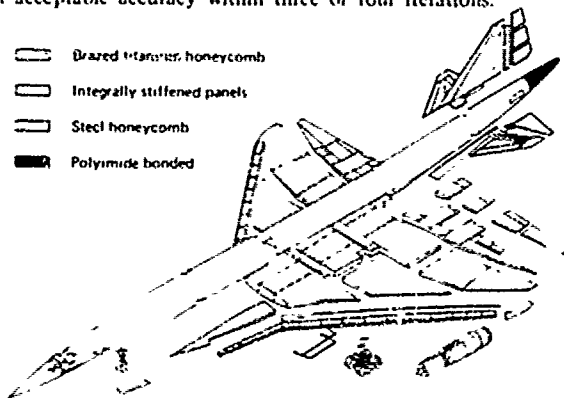


Figure 34.—Prototype Design Concepts, Sandwich and Integrally Stiffened Structure



Figure 33.—Full-Scale Wing Panel

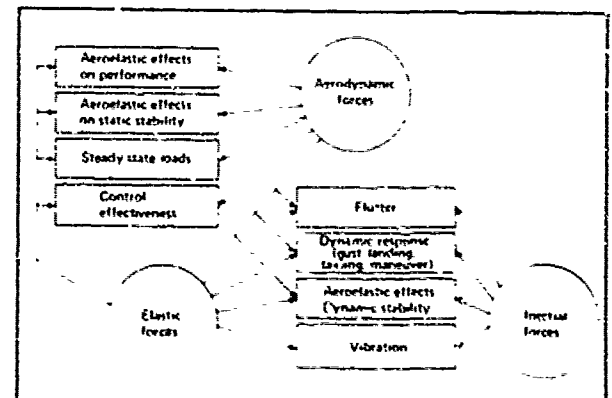


Figure 35.—Aeroelastic Triangle

During the design of the SST, the whole situation was found to be different. The low-aspect ratio wing with a 3% thickness/chord ratio was much more prone to aeroelastic effects than the subsonic transports. The wing aerodynamic force distribution throughout the supersonic transport flight regime moved over a much wider range, and yet the airplane center-of-gravity range had to be constrained within considerably tighter limits than on the subsonic airplanes. The net effect was that small changes in the aeroelastic solutions caused by the incorporation of improved definition or additional test data caused major perturbations in the structural design cycle. To accurately represent this interplay of the aeroelastic forces, a much finer grid system employing plate analysis techniques was used as illustrated in the right half of figure 36.

As shown in figure 37, the interdisciplinary constraints increase by an order of magnitude because the effects of hydraulic fluid compliance, control surface and actuator deflections, as well as the usual body, wing, and empennage deflections were required in the analytical model. Nonlinear characteristics, heretofore neglected or approximated by simple corrections, then became more important and required rigorous treatment. The interfacing of the various technical disciplines required a well-informed and highly integrated management organization.

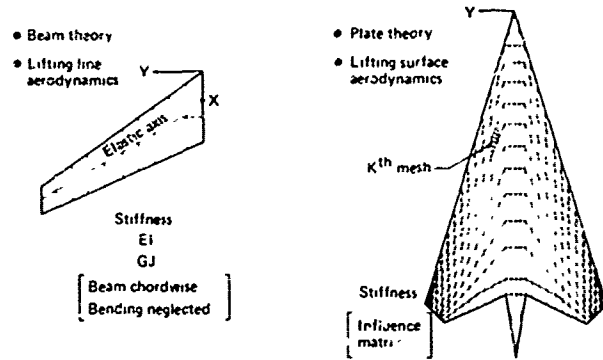


Figure 36.—Comparison of Subsonic/Supersonic Structure

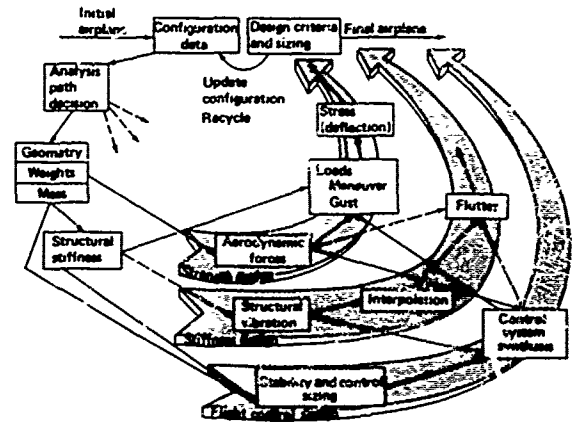


Figure 37.—Aeroelastic Cycle

The number of design conditions was increased by the addition of transonic and supersonic loading conditions to the usual low-speed, flaps-up climb and subsonic cruise conditions. The large number of fuel tanks with high surface area-to-volume ratio produced large variations in fuel mass distribution, further increasing the number of flight loading conditions which must be examined. It became apparent early in the preliminary design phase that proper decisions could be made only after each configuration under consideration had been tracked completely through its aeroelastic cycle until it converged into an integrated design, with consistent data for all subsystems. It was not the policy of the technical management to release configuration drawings until a reasonable convergence took place, and this could take as many as seven or more iterations because of data uncertainty at any given time. For example, on each configuration, experimental loads data were required from several wind tunnel models at subsonic, transonic, and supersonic speeds for final validation of basic aerodynamic loads, after all other elements of aeroelastic solution had converged. If these experimental loads data varied significantly from the theoretical loads, the structural design and analysis cycle had to be repeated to validate structural margins.

Figure 38 shows a typical 1-g transonic wind tunnel loads model under test. The proper timing of wind tunnel model construction and testing was a critical factor in keeping the SST program on schedule. Some of the effects of the mathematical modeling detail which compounded the understanding of the aeroelastic behavior of the SST are identified in figure 39. As indicated in this figure, local flexibility effects caused a significant change in deflections along the chord line. The inclusion of trailing-edge flexibility, for instance, had a dramatic effect on the airplane flutter speed. Figure 40 shows the magnitude of wing and body deflection relative to the jig shape as defined for the 2707-300 under the 1-g midcruise condition. These deflections must be accurately included when defining the twist and camber to give maximum cruise performance.

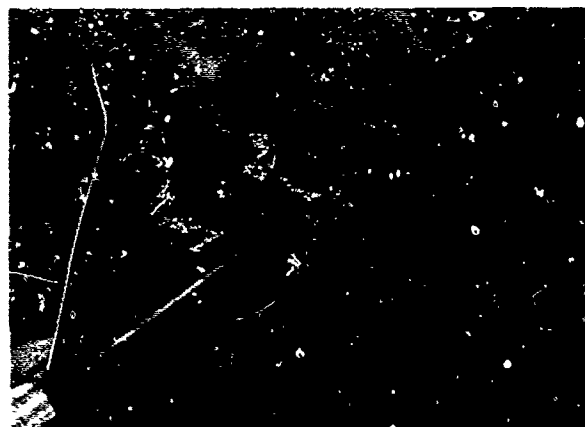


Figure 38.—Loads Model

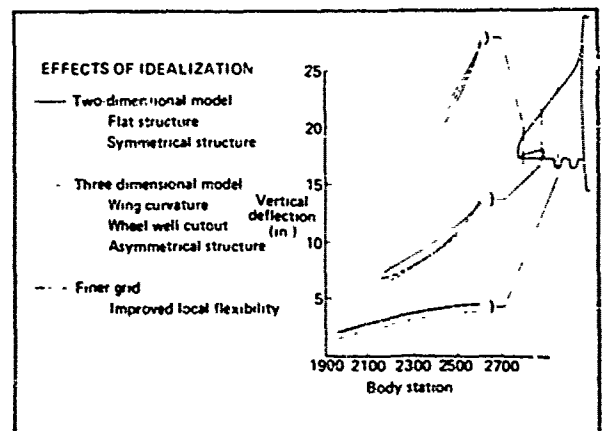


Figure 39.—Wing-Cruise Deflection

As was indicated earlier in figure 35, the inertial forces play an important part in the aeroelastic solution. When specifically applied to the calculation of body loads and deflection during a symmetrical pull-up maneuver, it causes the aerodynamic center of the airplane to shift aft on the order of 6% of the mean aerodynamic chord as shown in figure 41. When it is considered that the total useful range on the SST was only one-half that magnitude, it is clear that high standards of accuracy and very tight tolerance in the whole cycle are mandatory. The achievement of this accuracy took considerable time, analyses, and much testing.

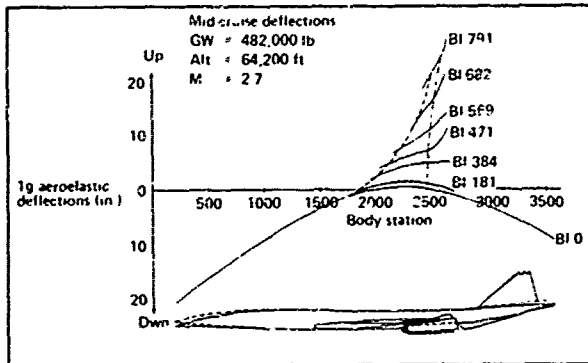


Figure 40.—Aeroelastic Deflections on B2707-300

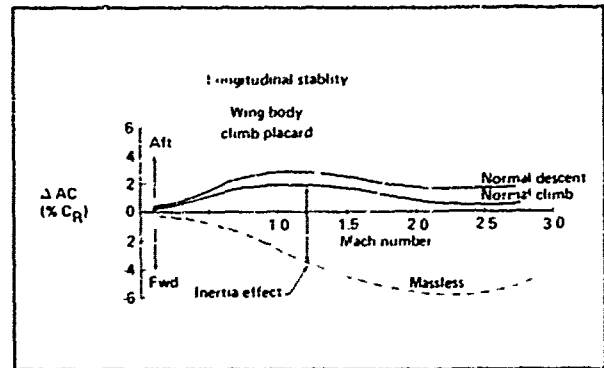


Figure 41.—Aerodynamic Center Shift Due to Aeroelasticity

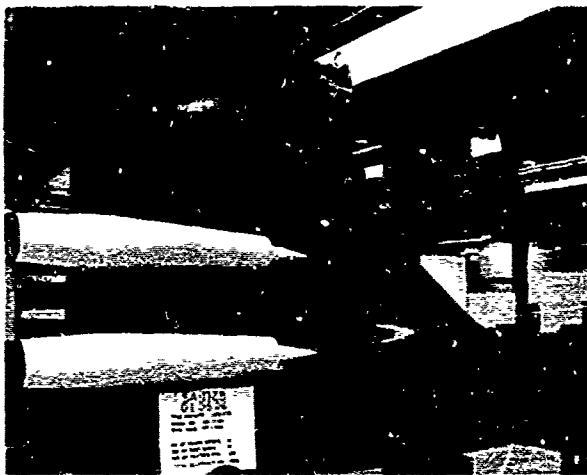


Figure 42.—Half-Span Transonic Flutter Model



Figure 43.—Full-Span Low-Speed Model

After a good representation of the structure, flight controls, and the desired aerodynamic cruise shape had been established, a flutter determination of each major configuration was essential. It was learned early during the SST program that a stiffness analysis of the entire aircraft, including wing, empennage, and all control surfaces, was necessary to assure that adequate flutter margins were achieved throughout the flight envelope. Precise representation of the engine pods mass and support beam stiffness was required because it was determined through the flutter analysis that the degree of coupling between the pod movement and wing flexure was critical in determining the airplane flutter speed. Flutter analysis methods for low-aspect ratio thin wings with large engine masses, located outboard and aft on the wing trailing edge, were still under development when the program was well under way. Excessive time required for flutter analysis and inefficient redesign methods for correcting flutter deficiencies were incompatible with the preliminary design schedule. To establish flutter boundaries in the transonic regime and to confirm the subsonic and supersonic analysis, it was necessary to build elaborate flutter models with scaled structural properties and with provisions for various fuel-loading distributions, surface controls, and actuator representations. Full-span, half-span, and empennage models were tested and correlated with mathematical representations of the modes. Figure 42 shows the details of a half-span transonic model; figure 43 shows a full-span low-speed model, and figure 44 shows the empennage models tested on the 2707-300 configuration. Similar models were evaluated on other competing configurations. Even with the wind tunnel test program, sufficient information was not available in time for management review which could have led to configuration changes to alleviate flutter weight penalties. These circumstances placed considerable emphasis on the future flight testing of the prototype under a restricted flight envelope and ground testing to establish firm requirements for added structural stiffness. In other words, it was essential to accept additional risk in the prototype by leaving conservatism in weight out of the airplane. This, of course, is the purpose of a prototype.

Solutions to structural problems related to aeroelastic effects are reflected as increases in strength and on stiffness which directly impact the airplane weight and its operating efficiency. Hence, much effort was expended in examining rib and spar arrangements, structural concepts, optimum distributions of structural material, fuel management, and flight control system frequency response to reduce the weight increment for stiffness and strength to a minimum. In retrospect, it is believed that a considerable further reduction in the weight increment for stiffness would be possible, based on flight test results, ground shake tests, and refined analysis methods over those defined of necessity during the design phase of the 2707-300 SST prototype program.

THE DESIGN INTEGRATION PROCESS

From the previously related technical experiences, it must be clear that the technical management in a high-risk technology design must be flexible, alert, and prepared for change. The management process had to be understood by all to be efficient. For example, a major decision of the technical management process in the SST program was to recommend a redirection of the entire program only 8 months after the final design process began. This decision was not made lightly! Another equally major, but less publicized decision of the SST program was to decouple the prototype and production programs to explore additional high-risk technology full scale.

The initial plan of the SST program was to proceed to design and manufacture an aircraft which would be identical in every way to the proposed production vehicle. One set of tools would serve for both vehicles. Such an approach would permit only minor modification during flight test, and such change could then be incorporated in production releases. The decision to design only a prototype using soft tooling, from which a largely redesigned vehicle would follow, permitted many cost-saving and increased-risk concepts to be considered as a part of the flight test program. For example, a folding ventral fin, incorporated in the prototype, could be removed piecemeal as high-speed directional stability and control were better understood from flight measurements. Similarly, wing tip extensions, strake additions, material substitutions in wing and body section, and dynamic ballast for flutter were all a part of the flight test plan for the prototype.



Figure 4A.—Empennage Model

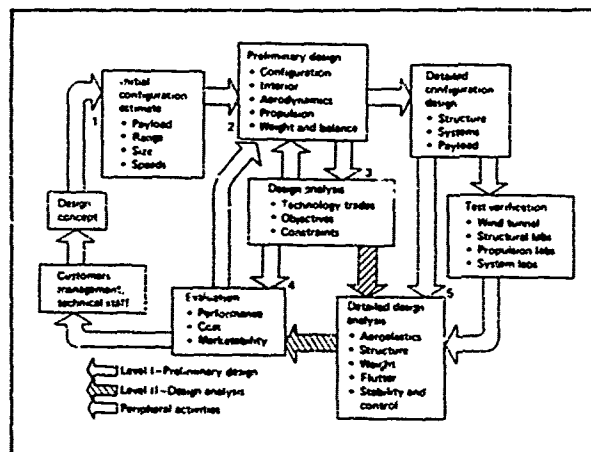


Figure 4B.—Preliminary Design Flow Chart

Preliminary Design

The preliminary design process usually takes place in three distinct phases. The initial phase in a modern aircraft preliminary design cycle is shown in figure 45. Competing concepts are proposed to and approved by a technical council of the program. A configuration manager, working with the chief technical and project engineers, establishes a configuration layout which is expected to meet design and operational requirements. After having established a layout of the interior payload system, fuel management system, landing gear, propulsion arrangement, and general configuration aerodynamic goals, a preliminary weight and balance check is made. Similarly, tail and control surface sizing are made consistent with static stability and control criteria. This aircraft is then evaluated, based on category I (parametric) weights, with performance, safety, cost, and economics estimated. Several iterations of the configuration take place until a fundamentally acceptable design results.

Progressively, the design is refined against more stringent and detailed design criteria and operational constraints. Technical and configuration trade studies, as well as detailed design features, are conducted to support the design evolution. To validate or update the safety, cost, and performance estimates of the configuration, detailed design analysis takes place in the areas of structure (loads, deflections, sizing, dynamics, flutter, etc.); flight controls (surface rates, hinge moments, hydraulic sizing, aeroelastic corrections, failure mode analysis, flight envelope analysis, etc.); propulsion (intake analysis, airflow management, thermal management, exhaust system, installation design, etc.); and category II (preliminary layouts) weight analysis.

Finally, an aeroelastic cycle analysis of the refined configuration, as illustrated earlier in figure 37, is made. For the SST, this process often took three or four iterations to converge on a reasonably acceptable solution for final configuration evaluation. Actual design drawing release took several more cycles to reach precise convergence. During this period, major validation testing of systems and components was carried out, and several preliminary design reviews on subsystem design occurred.

Somewhere in this latter cycle, the preliminary design process ends, and final design begins. Long lead-time material purchase and system supplier decisions occur when efforts on other competing designs is reduced to a small level or ceased. The precise time when preliminary design ends is never clear. Such a series of preliminary design steps took place in narrowing down the SST options to the 2707-300.

Design/Operational Requirements

Figure 46 shows certain representative SST economics design requirements imposed on all SST configurations to comply with airline desires and competitive economics goals. Figures 47 and 48 show representative SST operational and airport requirements as established within Boeing, based on past successful programs and government regulations.

It was found desirable and necessary to make such requirements as general as possible and to constantly consult configuration-unique trade-study data to be sure that no specific requirements would place a promising configuration in jeopardy unnecessarily. These requirements were reviewed on a continuous basis with potential airline customers and with government regulatory agencies.

Design Criteria

Criteria for aircraft design can simply be expressions of the facts of nature; they may be established unique to a given manufacturer's philosophy of design; or they may be a direct translation of government regulations or proposed regulations (FARs in the U.S., for example) in design commitments. No matter how the criteria are established, flight safety is paramount. Design life and design acceptability are other major factors in criteria selection.

During the 14-year SST program at Boeing, continuous internal technical committee action took place on the establishment of design criteria uniquely applicable to this vehicle. Criteria documents in such areas as structural integrity, stability and control, handling qualities, propulsion, safety, and aerodynamic cleanliness were developed to provide a meaningful set of constraints with respect to safety, life, and operational acceptability—without subjecting the design unnecessarily to constraints which would hamper its economic viability.

Because design criteria tend to be highly specialized and rarely read by higher program management, the basic constraints in each discipline were posted in the engineering workroom and color-coded against each given configuration for all to see. Figure 49 shows a posting for some of the objectionable flight characteristics. For example, if a given configuration evidenced locked-in pitchup at a corner of the operating envelope, under the most adverse conditions of weather and failure, it was revealed in bright red so that no one could miss it; yellow meant conditionally acceptable; and green indicated totally acceptable. The color was assigned by the discipline technical manager and no red marks were permitted to stand. It was found that unless some graphic means were available to notify management of such problems, they could readily be buried in the enormity of the program. The highlighting of problem areas in this manner proved to be most useful in the preliminary design management scheme.

In many instances, it was not easy to put down a concise statement of design criteria because insufficient knowledge existed. These areas were assigned to vigorous analysis, and tests were performed to establish tentative criteria until a better understanding could be achieved from subsequent prototype flight test.

OBJECTIVES
<ul style="list-style-type: none"> • Capture international passenger traffic in 1980 decade (500+ airplanes) • Provide airline operators with reasonable return on investment (15 to 20% ROI)
REQUIREMENTS
<ul style="list-style-type: none"> • DOC: 1 cent per ASM • Standard fares (no surcharge) • Utilization: 10-12 hours per day • Payload to-gross weight ratio: $\geq 5\%$ • Worldwide network of city pairs

Figure 46.—Economics

<ul style="list-style-type: none"> • Compatible with existing FARs • No sonic booms over inhabited land masses • Capability to go to destination following single-engine failure • Simple fuel management scheme/unrestricted seating • Low-speed operations equivalent to present-day aircraft • 30-minute turnaround • 50,000-hours design life • Overall design reliability $> 10^8$ hours before extremely remote failure (systems)

Figure 47.—Operational Objectives

<ul style="list-style-type: none"> • Operate from existing runways ($\leq 12,400$ ft) • Floatation as good as present aircraft • Turn radius compatible with existing airport taxiways • Meet subsonic community noise standards

Figure 48.—Airport Requirements



Figure 49.—Flight Characteristics Objectives

Engineering Workroom

Because the SST design embarked into so many different areas of advanced technology, it was essential that no discipline be allowed to isolate itself from the other elements. A centralized engineering control and workroom was established during the 1968 reevaluation period. All pertinent data were displayed on all competing configurations in one large room. Secondary control rooms were established throughout the program work areas for each major discipline. The system worked so well that it has been carried into several subsequent programs within Boeing.

Figure 50 shows a view of the actual SST workroom to illustrate the posting of schedules of analysis and design activity. Figure 51 is another view of the room to illustrate the number of visibility boards which existed on such items as aerodynamic performance, weights, propulsion, safety, and other technical data pertinent to an instantaneous view of program status with respect to competing configurations. The room, as shown, was photographed during the final design phase of the 2707-300 aircraft.

Figure 52 shows a typical 2707-300 display board of airplane performance. Changes to these boards were posted daily at the discretion of the individual task manager. Figure 53 shows a typical daily log of the technical concerns board, a technique used to identify the "top 10" most critical technical problem areas, where schedules of test and analysis were usually displayed together with key completion dates. Most often, the "top 10" directed the major technical program effort during final configuration firm-up before the design release process began.

The main purpose of the central engineering workroom was to give technical management all the major facts, as well as to provide total communications between major work elements of the program. Such a technique created a great sense of awareness as well as camaraderie throughout the huge (2000-3000) engineering work force employed on this project.

Engineering schedule maintenance was considered just as important as technical accomplishment because any schedule slide would mean increased program nonrecurring cost. A similar color code technique was used on all schedule information which was prime to holding design release dates. Care had to be exercised not to overcontrol schedules on secondary branches of the engineering program. The aeroelastic cycle analysis for the design provided a key schedule item. Such an analysis on this aircraft revolved around all major disciplines and systems and involved tedious data handling between the disciplines.



Figure 50.—Engineering Workroom—View 1



Figure 51.—Engineering Workroom—View 2

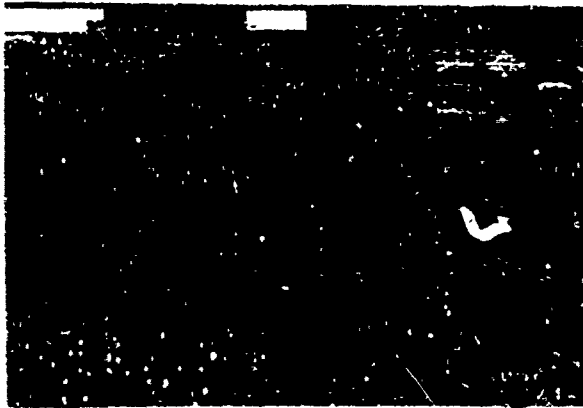


Figure 52.—Performance Comparison

Weight Control/Management

Once the preliminary design process has started, most deficiencies in any design creep in as weight increases. These increases are due to (1) better definition of design detail; (2) design load updates; (3) criteria conflicts; (4) material substitutions; (5) cost trades; (6) manufacturing limitations; and (7) as solutions to many other unforeseen problems. In the SST program, it became essential to organize weight control activity with weekly updates for management of the causes for all new weight changes. From such information, decisions could be made to offset weight increases through further trade studies and program modifications.

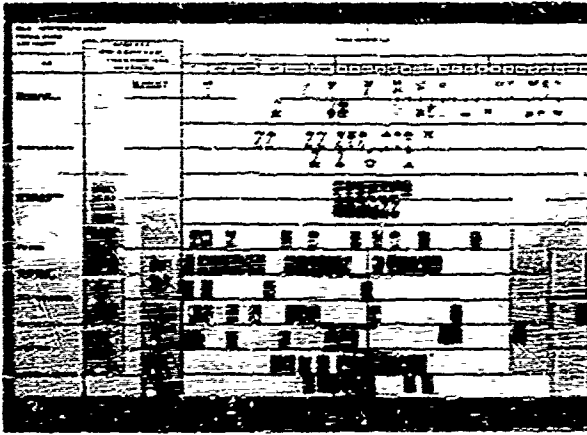


Figure 53.—Technical Concerns

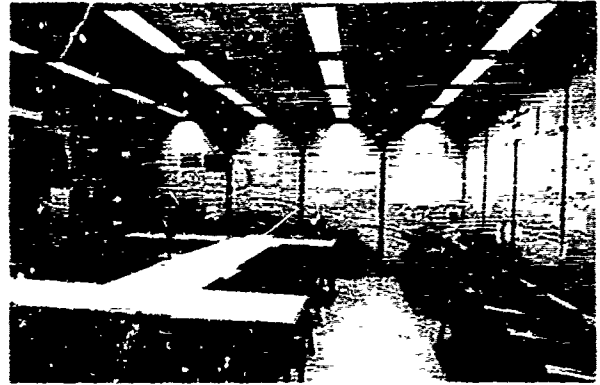


Figure 54.—Weight Control Room

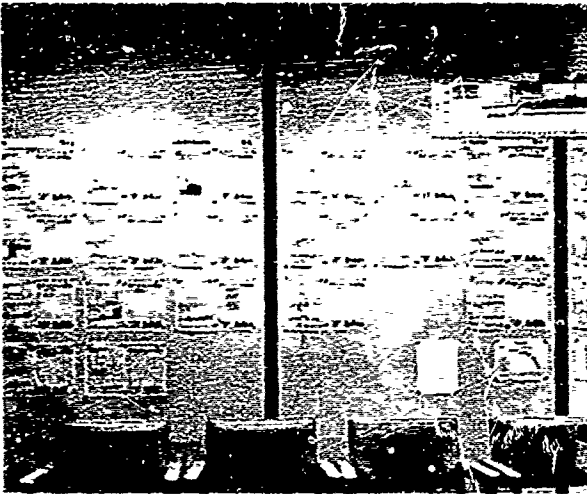


Figure 55.—Structures Weights

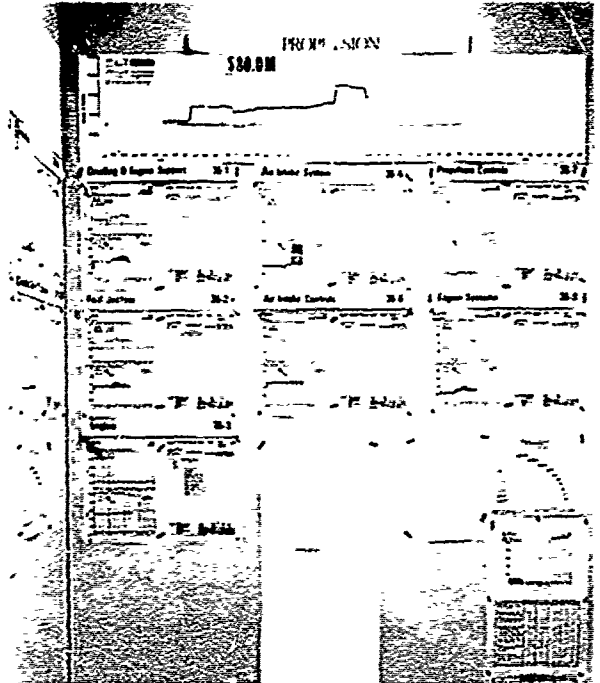


Figure 56.—Propulsion Weights

Figure 54 is a photo of the central weight control room, wherein each major subsystem was tracked separately, and the defining conditions for the subsystem weight were available for ready review. Actual weights against assigned target weights were shown, and a cost-effectiveness parameter for each subsystem as a function of material, structural concept, complexity factor, and ballast sensitivity was constantly updated. Status of weight trade study data applicable to the particular subsystem was also noted on the pertinent display board. Figures 55 and 56 illustrate typical weight display areas for the 2707-300 for the airframe structures and propulsion system. Figure 57 shows a typical summary weight tabulation against target weight for the 2707-300. The weights technology staff also managed forecast data on future weight trends on a weekly basis and a list of specific weight concern items, based on continual tracking of design detail throughout the engineering organization. Often this meant watching the detail drawings being made and debating with the designer on detail. In this manner, top management redirection could be given on a weekly basis before design details had proceeded too far.

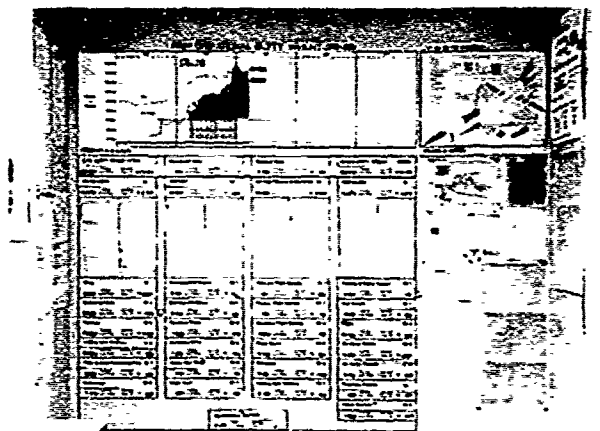


Figure 57.—Summary Weight Data

A weight reduction task force, composed of both manufacturing and engineering personnel, was initiated at the very beginning of the 2707-300 detailed design phase—just after preliminary design was complete. Suggested items for weight reduction were submitted to this team daily. Trade studies were initiated which involved materials substitution, design concept revision, manufacturability, and prototype cost. Such a task force was found to be a most valuable asset toward assurance of meeting nonrecurring cost control.

Design for Cost

In any preliminary design, it is essential to consider the ultimate cost of the production vehicle. The SST was no exception. However, the cost considerations in the definition of a preproduction prototype are quite different than for a prototype from which a production design would later take place.

It was necessary in the SST program to keep track of two designs: the production airplane expected from the prototype goals, and the actual hardware being constructed for test. Throughout the preliminary design phase, only the former, the production goal airplane, was considered. An analysis of the market capability of each concept was made, based on the economics evaluation of the configurations. In general, a production goal of 500 or more airplanes was considered reasonable. It was expected that the costs to design, develop, and fabricate the production airplane would be absorbed in the first 200 airplanes. Based on an estimate of the costs incurred throughout the program, up to the 200th airplane, it was determined that an average value in design of between \$200-\$300 per pound of weight could be expended usefully to maintain a proper balance in design for cost.

A value engineering analysis was made of each configuration after the concept was generally worked out. For example, figure 58 shows the value of structural weight improvement assigned to the 2707-300 design concept in relation to location in the aircraft. This figure shows that, although an average of \$200-\$300 per pound per aircraft was a fair price, the actual value was much higher in the empennage area, and much lower in the most forward areas of the cockpit. This gradient was caused by the fact that all SST designs tend to be tail-heavy and require large ballast to maintain proper cg spread. Such values were assigned to all structural subsections as a part of detailed cost/weight saving during detail design.

Similar cost/weight trends for mechanical, electrical, and electronic systems were developed as a means for cost-of-design control.

Other methods of value were established in terms of parts count and complexity as a means of general comparison of designs when all other economics viability indicators were constant. Such analysis had a strong effect on the final decision toward a relatively simple aircraft design concept, the 2707-300.

Technical Council

All major engineering decisions were conducted through a technical council composed of the director of engineering, the chief design engineer, and the chief technical engineer. The council convened weekly or more often as the program progressed. Prior to such decisions, the chief design engineer conducted preliminary design reviews (PDRs) on all major subsystems on a scheduled basis, while the chief technical engineer continually conducted detailed reviews on design criteria, subsystems specifications, and design requirements. The design had to converge on these targets, or revision had to be made. The technical council, meeting weekly to resolve the natural conflicts which occurred, led the way to a smooth process for keeping the preliminary design on course. All engineering personnel knew the decision process. This was found to be fundamental.

Management Council

Other basic program decisions which involve cost targets and actual costs, manpower spread between engineering, operations, finance, and program planning were reviewed weekly by a management council. Airline and government reviews of program status were conducted through this council. General activities between engineering, manufacturing, and outside procurement also were reviewed by this council, which had a significant outer-loop effect on the preliminary design as well as the final design process. Many other actions by this council were fundamental to the SST program but are outside the scope of this paper.

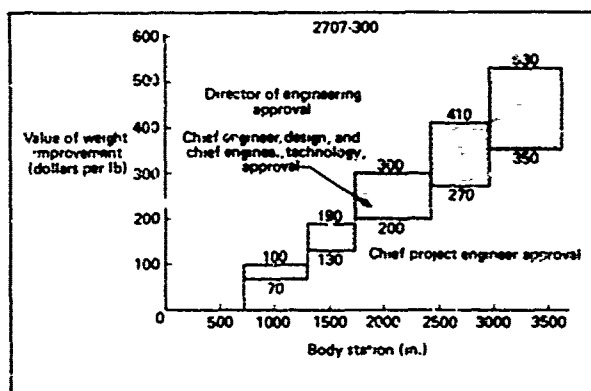


Figure 58.—Value of Weight Saved

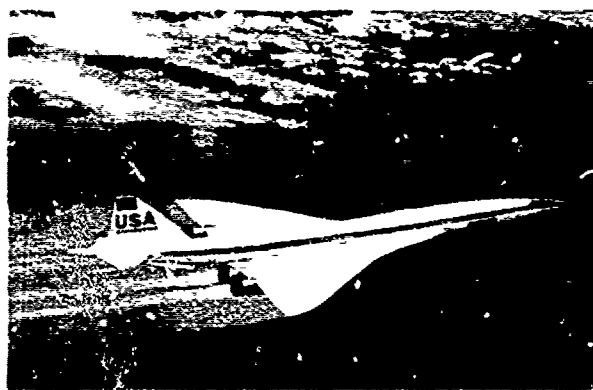


Figure 59.—B2707-300 Configuration

LESSONS LEARNED

The final design concept chosen for construction, the 2707-30C (fig 59), was the culmination of a series of conflicts between performance objectives and design constraints. At the onset of the program, an attempt was made to maximize both high- and low-speed aerodynamics. As design knowledge and constraints became better understood, the importance of dynamic loads and elasticity or stiffness of both primary and secondary structure began to have its effect. These consequences showed up as increased weight to offset the initial aerodynamic objectives.

Further, the attainment of peak aerodynamic performance to ensure economics feasibility of the airplane was limited by the effect of OEW cg range and payload/cg range, as constrained by stability and control criteria for safe flight.

Every attempt was made to reduce load paths and to minimize the basic balance problems to keep the developing configurations from constantly diverging.

The manufacturing cost of the airplane was found to be directly related to the same parameters which simplified the design and reduced load paths. Hence, the final configuration was a balance of cost, technical possibility, operational usefulness, and performance. No one was totally satisfied, but it was the best that could be accomplished within the constraints of technology, economics, and operational requirements.

Considering the size of this preliminary design effort, some positive lessons were learned, summarized here for consideration in high technology preliminary design planning:

- Establish total visibility of configuration alternatives as early as possible.
- Keep a constant update on the opportunities and constraints of each configuration.
- Post all important applicable constraints for review and modification.
- Continually conduct useful trades studies which influence costs, weight, and/or performance.
- Initiate formal weight reduction action as soon as a final configuration is selected.
- Let every member of the preliminary design team know where and how he can be heard.
- Keep a sense of urgency going on at all times to achieve greatest possible efficiency.

A second generation SST will evolve, perhaps, at some future time. If so, it is hoped that the experience gained in the Boeing/U.S. SST effort, as recited in this paper, will prove useful.

DESIGN EVOLUTION OF THE
BOEING 2707-300 SUPERSONIC TRANSPORT

Part II
Design Impact of Handling Qualities Criteria,
Flight Control System Concepts, and
Aeroelastic Effects on Stability and Control

by

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SUMMARY

The intensive efforts undertaken to develop an economically competitive SST stimulated new design approaches in the areas of airplane longitudinal control and stability and flight control systems design. Extensive research work was conducted to push the state of the art as hard as possible in the development of handling qualities criteria and in the design evolution of the stability augmentation system. The end product was a control-configured vehicle employing multiple redundant electronic stability augmentation systems to meet design requirements for both normal handling qualities and minimum-safe handling qualities. This design approach contributed substantial gains in range/payload capability over that attainable through the conventional approach that inhibits airplane design through the requirement to provide inherent aerodynamic stability. Throughout all of the design development work the effects of structural aeroelastics on aircraft stability and control played a major role in configuration design decisions. The complex engineering work involved in the aeroelastic analyses paced the configuration development design cycles and contributed substantially to the total engineering costs. The experience gained in these areas has identified the need for improved quality, automated aeroelastic analysis methods to speed the design development work and reduce the engineering costs and design risks.

INTRODUCTION

The preliminary design process for conventional subsonic commercial jet aircraft consists of closely integrated efforts in aerodynamics, propulsion, structures, and weights analysis. This design optimization seeks the most efficient configuration arrangement of wing, fuselage, propulsion system, and tail surfaces. The design of the flight control system is normally an important parallel effort, but not essential to configuration integration and optimization. The role of the flight controls design team changed radically from this past practice during the Boeing SST program with the adoption of the control-configured vehicle (CCV) design approach. The flight control system became an integral part of the main iterative design loop and a determining factor in configuration arrangement, structural weight, and aerodynamic efficiency.

The effects of structural flexibility on airplane stability and control were major considerations that strongly influenced the decision-making process, beginning with the trade studies phase of configuration evaluation. This attention continued through final configuration design development. The complexity of the aeroelastic analysis produced a costly, time-consuming engineering effort involving multiple interactions of all technical design disciplines. The engineering design cycle, as a result, was paced by that effort.

This paper will discuss the factors leading to the adoption of the CCV design approach and the benefits attained. The important role of structural aeroelastic analysis in the design decision-making process also will be described.

DESIGN IMPACT OF FLIGHT CONTROL SYSTEM CONCEPTS AND HANDLING QUALITIES CRITERIA

The design goal of the U.S. supersonic transport program was to develop and market an SST configuration sufficiently advanced in concept and design to be economically competitive with the best of today's long-range commercial jet transports. This formidable goal forced intensive efforts to reduce aircraft structural weight and to improve aerodynamic and propulsion system efficiency. In the area of flight control systems design, the efforts to push the state of the art as hard as possible were rewarded with impressive payoffs in improved aircraft performance. This work centered around the adoption of the control-configured vehicle (CCV) design approach, wherein the flight control system was employed to stabilize as well as control the airplane to effect a more efficient aerodynamic and structural configuration.

SST design studies showed that very substantial reductions in aerodynamic drag and structural weight could be achieved if the aft center-of-gravity (cg) limit could be shifted aft of the point required for inherent longitudinal stability. The configuration improvements resulted from the improved longitudinal balance situation permitted by shifting the operating cg range aft. The accruing benefits were reduced body length, gear length, and tail size, and the savings in weight and drag were reflected in substantial improvements in range/payload capabilities.

To attain these benefits, studies were launched to develop a stability augmentation system of sufficient capability and reliability to permit placement of the operating limit well aft of the stability neutral point, thus assigning full responsibility to the augmentation system for safe handling qualities. At the same time, research studies were oriented to determine what should realistically constitute minimum-safe design criteria, as well as criteria to govern the good handling qualities required for normal operation.

Range/Payload Benefits Achieved Through Hard SAS

The flight control system studies evolved a concept called Hard Stability Augmentation System (HSAS) because the system reliability approached that of hard structure. This was achieved primarily through design simplicity and system redundancy. The HSAS was an independent augmentation system employed to back up the basic Electronic Command and Stability Augmentation System (ECSS). Both HSAS and ECSS were quad-redundant. The ECSS was a sophisticated system tailored to achieve excellent handling qualities throughout the flight envelope, whereas the HSAS was required to provide only a conservative level of safe handling qualities in the remote event of inoperable ECSS. The total system reliability achieved with this approach permitted the SST to evolve as a CCV design, employing augmentation in the longitudinal axis to maintain safe flying qualities.

The incorporation of HSAS allowed an improved configuration arrangement from the standpoint of aerodynamic and structural efficiency. The integration of wing, nacelles, and fuselage was dictated by the consideration of supersonic cruise lift/drag ratio (L/D) and structural efficiency.

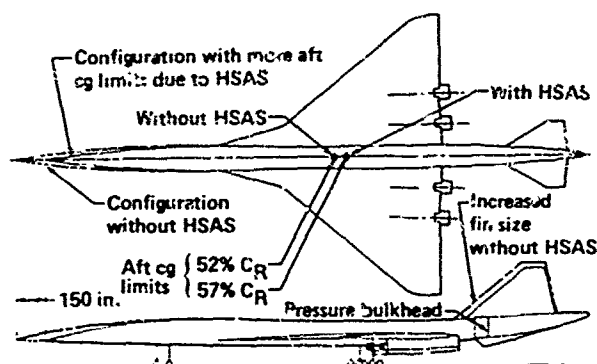


Figure 1.—Impact of Hard SAS on Configuration

The integration of wing, nacelles, and fuselage was dictated by the consideration of supersonic cruise lift/drag ratio (L/D) and structural efficiency. This configuration arrangement placed the engine nacelles far aft under the wing to achieve "favorable" interference. The longitudinal balance and instability problems created by the far aft location of the engine mass were alleviated by the HSAS, which permitted a shift in the operational cg range of 5% wing root chord (C_R) aft of the point required for inherent (OAS) stability. Figure 1 compares the HSAS SST configuration with a more conventionally balanced airplane having the aft cg limit at the static neutral point, 52% C_R . The 5% aft shift in the limit of 57% C_R , made possible with the incorporation of HSAS, placed the aft cg limit closer to the cg of the basic airplane/propulsion system combination (OEW cg), thus easing the balance problem. The payload cg could then be shifted aft, since forward payload location was no longer required to balance

a far aft OEW cg. This permitted an aft shift of the body on the wing and elimination of excess body length previously required for balance, resulting in a reduction in body length of 150 inches. The shortened fore body and the landing gear relocation aft coincident with the rebalance allowed a reduction in vertical tail size and a reduction in the landing gear length required for ground clearance. These configuration changes, plus some additional minor system revisions and a reduction in maneuver tail loads, culminated in a total weight saving of about 6000 lb. Significant drag reductions also resulted from the reduced surface area and improved trim drag situation. Substantial performance improvements accrued from the weight and drag reductions. The most significant gain resulting from adoption of HSAS was a range increase of 225 nautical miles. Figure 2 indicates the magnitude of the payload/range trades involved with this increased range capability. It is apparent that in order to achieve the same range as the HSAS airplane, the airplane without HSAS must reduce its payload about 30%. References 1 and 2 discuss the design evolution of the HSAS airplane and describe the flight control system and stability augmentation systems in some detail.

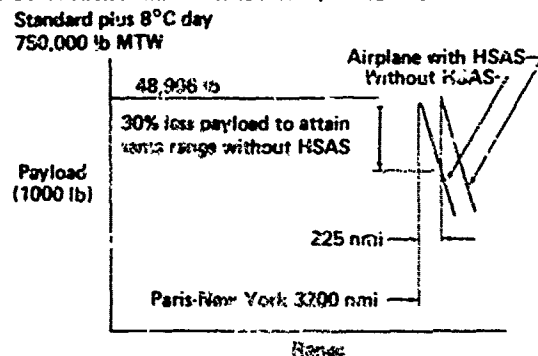


Figure 2.—Payload/Range Benefits of HSAS

Engineering Costs Associated With the CCV Design Approach

These impressive gains in range and payload were not achieved without an equally impressive increase in flight control system costs over those of current jet transports. For example, estimated total production costs of the SST flight control avionics systems were approximately double those of the Boeing 747, considering the 747 fully equipped with inertial navigation and Category III fail-operative autoland systems. This, however, is not an unreasonable cost increase in terms of percentage of total airplane cost; total flight control system avionics costs were estimated to approach 2.5% of the projected airplane selling price for the SST. This is about the same percentage of airplane price as the 747 systems.

The CCV approach to SST longitudinal balance and flying qualities had major impact on the aircraft design development engineering effort. In this new environment, the flight control staff became a major element in the configuration team, with a greatly expanded role in the design iteration process. It became necessary to include flight controls in the main iterative loop along with aerodynamics, structures, and propulsion staffs. The engineering burden was greatly increased in the flight controls area as a result of expanded analyses requirements and demands that paced the configuration development cycle. While conventional aircraft designs not relying on augmentation for safe flight can be synthesized with stability and control analyses of relatively few flight conditions, the supersonic CCV requires greatly expanded analysis. The number of flight conditions requiring detailed aerodynamic stability analysis increased over tenfold for the control-configured SST compared to conventional subsonic jet analysis. The depth of analysis also increased greatly due to the magnitude of the aeroelastic effects and the vital concern with aerodynamic and system nonlinearities with regard to assuring safe control of the unstable vehicle.

Data production and engineering analyses became highly computerized to keep pace with the engineering design cycles. Digital computer programs were designed to permit development of a complete synthesis of a proposed stability augmentation system on a 2-day turnaround basis. This effort involved the analyses of as many as 100 flight conditions. The role of flight simulation was also expanded greatly. The simulator became a primary design tool as well as an analysis or evaluation device, since the test results were crucial to configuration design decisions.

Engineering costs multiply rapidly in a number of areas when the CCV approach is adopted. Engineering manpower, computer resources, wind tunnel testing, and flight simulator testing all expand greatly over those required for more conventional design approaches. Scope of the increased engineering problem is illustrated by the increase of SST over 747 engineering design staffing requirements of nearly threefold in the critical areas of aerodynamic stability and control and electronic flight control systems.

Longitudinal Stability Achieved Through Hard SAS

High reliability was achieved for the HSAS through design simplicity, component quality, design installation quality, and the employment of multiple, redundant independent channels. To keep the system simple, minimum demands were placed on HSAS for

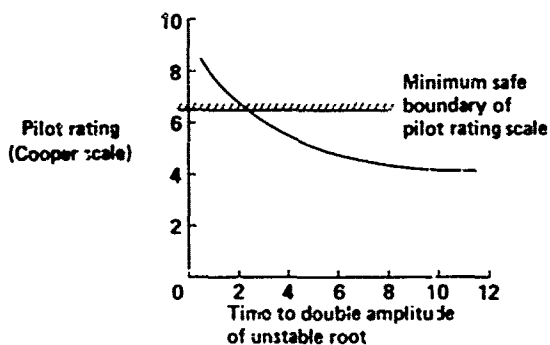


Figure 3.—Pilot Rating of Longitudinal Handling Qualities for Unstable SST Configuration

augmenting airplane handling qualities; the design requirements stressed the provision of a conservative level of minimum-safe flying qualities, rather than the development of good flying qualities, the latter being provided by the ECSS. Criteria for minimum-safe handling qualities were developed primarily through a program of extensive flight simulator testing. The criterion that guided the HSAS design concerned itself with the ability of the pilot to handle divergencies in pitch attitude or speed due to instabilities in the longitudinal axis. An example of the SST research to determine a minimum-safe criterion appears in figure 3, which shows the relationship between the pilot's evaluation of longitudinal handling qualities and the time-to-double amplitude of the unstable root in the three-degrees-of-freedom longitudinal equations of motion. The curved line on the graph summarizes the mean pilot rating as a function of time-to-double amplitude as obtained from various experiments on fixed-base simulators, moving-base simulators, and variable-stability aircraft. This work showed that pilots could safely control the airplane with time-to-double amplitude as low as three seconds. This is indicated on the graph by the pilot rating 6½ intersection with the curve. At pilot rating 6½, the aircraft handling qualities are objectionable, but the pilot can still retain safe control of the airplane. The HSAS design criterion, however, was selected as 6 seconds to double-amplitude to provide a comfortably safe margin for commercial aircraft design and operation.

To gain further perspective of the airplane longitudinal stability augmented by HSAS, figure 4 relates the aircraft stability levels with and without HSAS to the aft cg limits. The maneuver point defined in terms of neutrally stable short-period motion (with speed constant) is shown plotted versus mach number as a shaded band for the augmented airplane. The airplane stability defined at this manner is distinctly apparent to the pilot in terms of pitch rate response. For the most critical flight regime (landing approach), the effective maneuver point with HSAS operating is over 10% C_R aft of the aft cg limit. The more critical criterion, however, is the three degree-of-freedom criterion requiring the time-to-double amplitude of the unstable root to be not less than

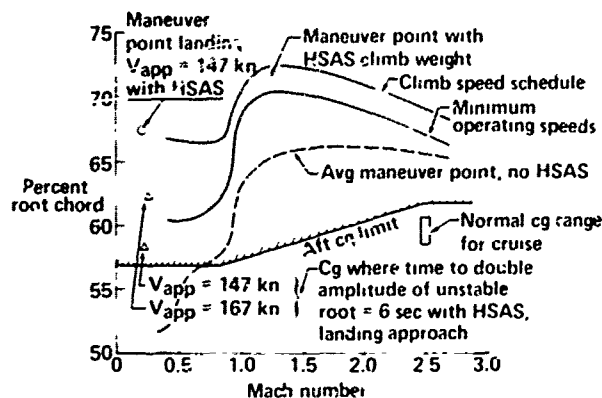


Figure 4.—Effect of HSAS on Longitudinal Stability

6 seconds. Figure 4 shows the cg at which time-to-double amplitude of the unstable mode equals the 6-second criterion for the critical landing approach task with HSAS operative. This cg is 58.5% C_R at the normal landing speed for maximum design landing weight (147 kn) and is 62% C_R when the landing approach speed is increased 20 kn to 167 kn. This shows an additional bonus provided by the HSAS: if necessary, the airplane can be operated safely with HSAS through the subsonic regime to landing without moving the cg forward of the mach 2.7 nominal cruise cg range.

The approximate maneuver points for the unaugmented airplane are also shown in figure 4 to provide an indication of the stability benefits achieved by the HSAS.

Determination of Aft CG Limits and Horizontal Tail Sizing

Development of the hardened stability augmentation system permitted a different approach to establishment of the cg limits and horizontal tail size. With inherent natural longitudinal stability no longer a requirement for establishing the aft cg limits for the unaugmented airplane, the paramount considerations became (1) the provision of adequate control capability at forward and aft cg limits and (2) the provision of acceptable handling qualities when operating in the HSAS mode as defined by the divergence criterion.

Evaluation of the design concept proposed for HSAS indicated that provision of acceptable flight characteristics in the HSAS mode would probably not be the determining factor for aft cg limit. Instead, longitudinal control was the apparent design constraint for both aft cg limit selection and for horizontal tail sizing. With the stabilizing benefits of the HSAS, it was possible to push the aft cg limit aft beyond the static longitudinal stability neutral point (into the unstable regime) to the point where nose-down control became critical at high angles of attack. This is illustrated in figure 5, which shows typical low-speed stability and control in terms of pitching moment versus angle of attack and also shows the selection of aft limit as the most aft cg at which full nose-down control will still provide adequate control for recovery from high attitudes or stall.

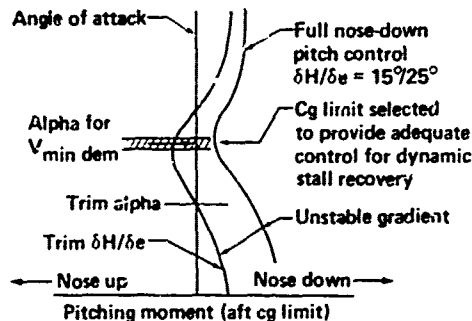


Figure 5.—Aft Cg Limit Selection, Subsonic

The establishment of horizontal tail size and cg limits evolved through an iteration process involving examination of high-altitude nose-down control requirements at aft cg limits and nose-up control requirements at forward cg limits. Horizontal tail size was selected to match equally the control power requirements at forward and aft cg limits. This is illustrated in the tail sizing diagram of figure 6. The most critical nose-down control requirement at the aft cg limit was met with the tail operating at its maximum lift (up-load); and the most critical nose-up control requirements at the forward cg limits were met with the tail operating at maximum attainable down-load. This design approach achieved absolute minimum tail size within the constraints imposed by the selected design criteria.

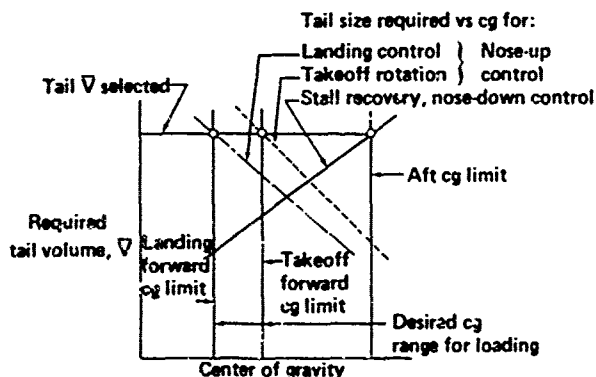


Figure 6.—Tail Sizing and Cg Limits Determination

The forward center-of-gravity limits for landing considered severe turbulence and wind shears in the final approach to landing. The landing flare criterion is shown in figure 7. Through a series of flight simulator experiments, a criterion was developed to relate pitch attitude acceleration requirements to the incremental lift or sink produced by the control input. The graph shows the acceptable boundary of pitch attitude acceleration versus the incremental normal acceleration (g) produced by the pitch control input. A negative value of g indicates the sink produced with control input. For example, short-coupled aircraft (such as tailless aircraft) generate relatively large sinking forces with pitch control input, relative to the nose-up control moments produced. The boundary then establishes a need for increasing pitch attitude acceleration for the more short-coupled aircraft. The relatively long tail arm afforded by the small aft tail control of the SST produced relatively large nose-up control moments with very little negative g sink, thus requiring rather minimal pitch acceleration capability.

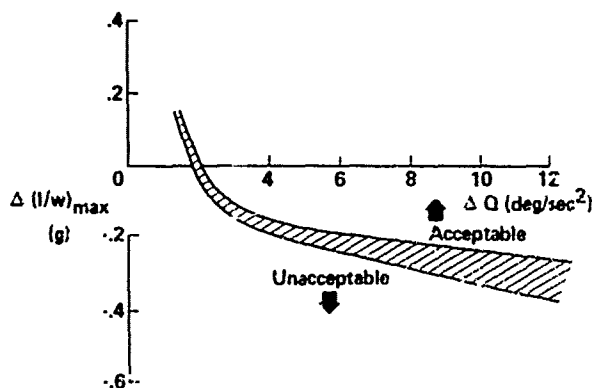


Figure 7.—Landing Longitudinal Control Requirements

augmentation system selected for the HSAS. The status of the engineering analysis at program termination indicated that the SST control system was closely approaching these design limitations. In the final analysis, the limitations on HSAS gain imposed by the structural modes would probably have approached closely the limitations imposed by the selected airplane controllability criteria previously illustrated in figure 5.

THE ROLE OF AEROELASTICITY IN CONFIGURATION DEVELOPMENT

The structural flexibility of the airframe as it impacts aerodynamic stability and controllability was a major design consideration for the supersonic transport. The long, slender fuselage and the thin highly swept wings that typify SST configurations are particularly sensitive with regard to the effects of aeroelastic deformations under load. The problem is further compounded by the relatively soft structural design (low structural load factor, 2.5 g limit) and the high mach number, high dynamic pressure operational environment. The severity of the aeroelastic effects on stability and control can become a governing factor in the configuration-design decision process. Unfortunately, the complexity of the aeroelastic problems with the multiple interaction of technical design disciplines does not lend itself to rapid solution, and the analysis can well become a pacing item in the airplane design cycle.

The discussions herein extend the material of Part I to cite some specific examples of aeroelastic effects on SST stability and control and emphasize the importance of providing high quality analysis compatible with rapid engineering design cycling.

Structural Flexibility Effects on Aircraft Stability

As previously discussed, the determination of the operating center-of-gravity range is a particularly critical design problem for SST configurations. The design considerations involve the provision of adequate longitudinal stability (whether by natural or artificial means), provision of the best possible trimmed L/D ratios over the normal operational envelope, and the provision of payload- and fuel-loading management that are compatible with airline operation. The problem of c_g range determination is compounded by the wide-ranging effects of mach number and structural flexibility on longitudinal stability and control effectiveness. The sensitivity of trimmed L/D and safe-control margins to c_g location makes it imperative that the longitudinal stability predictions be accurate and that c_g management be precise throughout the flight envelope. For example, the 300-foot long U.S. SST 2707-300 was designed with a c_g range of only 2 feet at takeoff and, through management of fuel usage, moved the c_g aft about 4 feet from takeoff to the beginning of mach 2.7 cruise and there maintained the c_g without ± 1 foot of the optimum c_g for cruise.

The effects of aeroelasticity on longitudinal stability are strongly dependent on wing planform. The upper half of figure 8 shows that for a variable-sweep arrow-wing configuration the aeroelastic effects completely dominate the mach effects on the aerodynamic center shift. The upper portion of the figure compares the maneuver neutral point variation with mach number for a rigid arrow-wing airframe with the flexible airplane maneuver points at V_{MO} (maximum operating speed) and V_D (maximum dive speed). Whereas, the rigid neutral points move sharply aft with mach number, the aeroelastic effects are destabilizing to such an extent that the high mach number stability is actually less than the subsonic stability. Thus, the airplane design would call for a fuel management system that provides for forward c_g movement as mach number is increased, as shown by the suggested aft c_g limit line. In contrast to this, the lower portion of the figure shows that the delta wing configuration is dominated by mach effects, the aeroelastic effects on stability being quite small and in this case slightly stabilizing. Thus, it is necessary to shift the c_g aft as mach number increases, in order to maintain optimum aerodynamic efficiency.

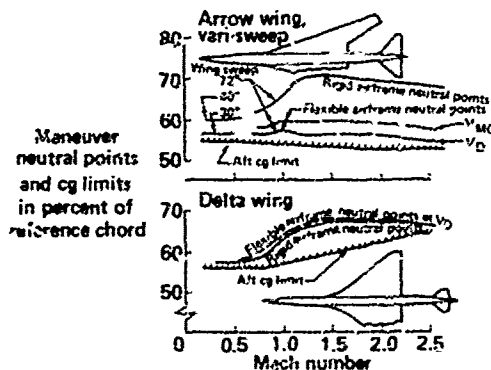


Figure 8—Effect of Aeroelasticity on Longitudinal Stability

An example of the nature and the severity of aeroelastic effects can be seen in a variable-sweep-wing configuration employing a canard surface to increase the longitudinal control and trim capabilities throughout the flight envelope. For this concept, the magnitude of the aeroelastic effects on longitudinal stability introduced by the canard proved to be a prohibitive risk from the standpoint of airplane stability and control.

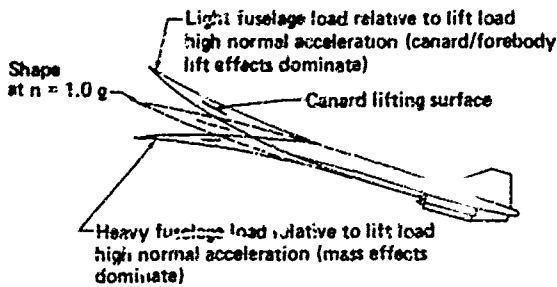


Figure 9.—Aeroelastic Effects of Canard in Maneuvering Flight

The shape of the elastic airplane is a function of the interaction of the structural flexibility, the mass, and the airloads. Figure 9 indicates the nature of the aeroelastic effects on longitudinal stability for the canard configuration. In maneuvering flight, the aerodynamic lifting forces acting on the canard, forebody, and forward inboard wing surface area tend to bend the forebody upwards. The mass effects of the loaded body (structural weight plus payload) produce a counter downward bending tendency. When the fuselage loading is light relative to the lift loading, the lift effects dominate, resulting in upward bending of the forebody. This is destabilizing since the upward bending induces additional angle of attack and additional lifting force, thus shifting the effective aerodynamic center forward. When the fuselage loading is heavy relative to the lift loads, the mass effects dominate, bending the forebody downward in a stabilizing direction.

Figure 10 shows the magnitude of the aeroelastic lift and mass effects for a long slender variable-sweep SST configuration employing a forward canard trim and control surface. This particular analysis is for maneuvering flight at mach 1.2 and shows the effects of aeroelasticity on the canard contribution to longitudinal stability. The incremental aeroelastic maneuver neutral-point shifts due to the canard are plotted versus flight dynamic pressure. For reference magnitude, the total center-of-gravity range is shown as 3-1/2% reference chord (root wing chord, 158 feet). The incremental aeroelastic lift effects are separated from the incremental mass or inertia relief effects to show clearly the relative magnitudes. The pure aeroelastic lift effect (massless airplane) is seen to be strongly destabilizing: at maximum dive speed, the neutral point is shifted forward 20% reference chord, or almost six times the entire cg range. Counter to this, the incremental inertia relief or mass effects are seen to be strongly stabilizing, shifting the maneuver point aft 27% C_R at a gross weight of 416,000 pounds and 21% C_R at a gross weight of 605,000 pounds. This variation with gross weight results from the fact that at lighter gross weights the mass effects are more dominant, since a given incremental change in angle of attack produces more load factor at light gross weights than at heavy gross weights. The net aeroelastic effects (combining the lift and mass effects) are found to be stabilizing. That is, compared to the rigid airframe with canard, the flexible airframe maneuver points are shifted aft due to the aeroelastic effects induced by the canard.

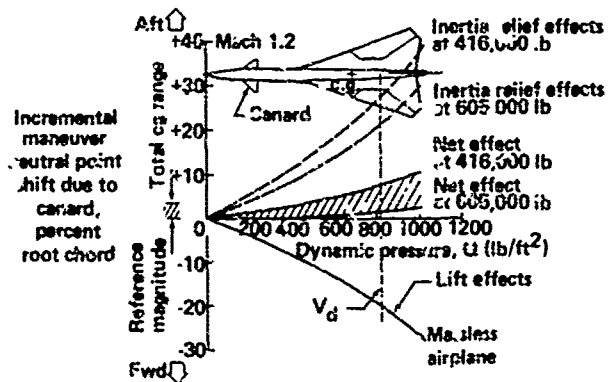


Figure 10.—Effect of Aeroelasticity on Canard Contribution to Longitudinal Stability in Maneuvering Flight

The risks associated with precise longitudinal balance of the flexible canard airplane are clearly evident here. First, the ability to balance the airplane precisely within a 3-1/2% C_R cg range is questionable when the canard aeroelastic effects alone are between five and eight times the magnitude of the entire design cg range. Second, the variation of the aeroelastic maneuver point shift due to the canard is a strong function of gross weight, for this condition, this variation exceeds the cg range by a factor of two for weights ranging from initial climb to final descent. Reference 11 details these SST design experiences with the canard configuration.

An additional area of concern is the provision of adequate directional stability. A well known characteristic of supersonic aircraft is the deterioration of directional stability with angle of attack at high mach numbers. This problem is compounded for long, slender, flexible aircraft by the aeroelastic losses in stability suffered from the bending under side loads of the aft body and vertical fin. Figure 11 shows typical SST directional stability and aeroelastic effects at the maximum operational speed, V_{MO} , at mach 2.7 for maneuvering flight at a load factor of 2 g. Here we see a 75% reduction in directional stability due to aeroelastic effects, the main source of the problem being the large loss in the stability input of the vertical fin. Stability problems of this magnitude impact the airplane design process severely in the areas of structures and flight control systems, where new design approaches are necessary to avoid undue penalties to airplane structural weight and aerodynamic drag.

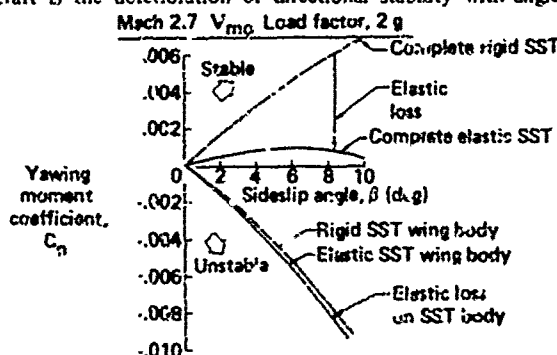


Figure 11.—Effect of Aeroelasticity on Directional Stability

Structural Flexibility Effects on Aircraft Control

The aircraft designer has long been plagued with the problem of loss of aerodynamic control effectiveness due to structural flexibility. This has been a dominant consideration on transonic military aircraft, and the current generation of subsonic cruise commercial jet aircraft have typically experienced a marginal situation with regard to providing sufficient high-speed effectiveness for control surfaces such as outboard wing ailerons. The problems are more acute for supersonic transports which operate over a much greater range of mach numbers and dynamic pressures.

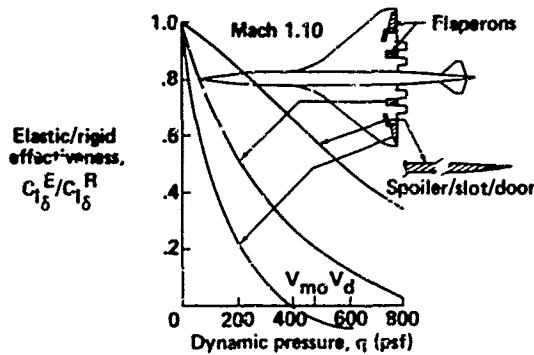


Figure 12.—Effects of Aeroelasticity on Lateral Control Effectiveness

Typically, trailing-edge controls outboard of mid-span are of little use as primary flight controls at other than low-speed conditions. Figure 12 shows the losses in lateral control effectiveness that are typical for a delta wing SST at transonic speeds. For this particular design case, the consideration of lateral control elastic effectiveness played a major role in configuring the wing trailing-edge flap and control systems and in the selection of the proper combination of controls to achieve good, coordinated lateral control response. The inboard flaperons maintain effectiveness to speeds beyond V_D ; however, at V_D they have lost 87% of their rigid effectiveness. The spoiler/slot/door controls, being located well forward of the trailing edge and relatively closer to the wing elastic axis, suffer low torsional losses and thus maintain nearly 50% effectiveness at V_D . Thus, a combination of inboard flaperons and spoiler/slot/door controls was selected for the high-speed roll control system. The outboard flaperons suffered extreme losses in effectiveness at speeds well below V_{MO} and thus were restricted to low-speed operation.

Similar data are shown in figure 13 comparing tail elevator aeroelastic effectiveness with an all-moving (slab tail) horizontal tail control. The elevator control is seen to lose about 80% of its effectiveness at V_D , whereas the slab tail maintains about 45% of its rigid effectiveness. When we further consider that the slab tail is several times as effective as the elevator when compared on a rigid basis, we are led to select the slab tail for high-speed control.

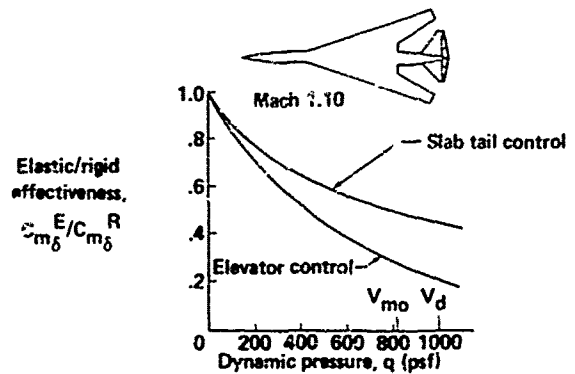


Figure 13.—Effects of Aeroelasticity on Longitudinal Control Effectiveness

A final example of the important role that structural flexibility plays in stability and control analyses is shown in figure 14. Here we see the pronounced effect of aeroelasticity on the design hinge moments for an all-moving horizontal tail control. At the transonic design condition, 1.5 load factor at maximum operating speed, the structural deformation of the slab tail resulted in a predicted 60% reduction in required control design hinge moment. This demonstrates the profound

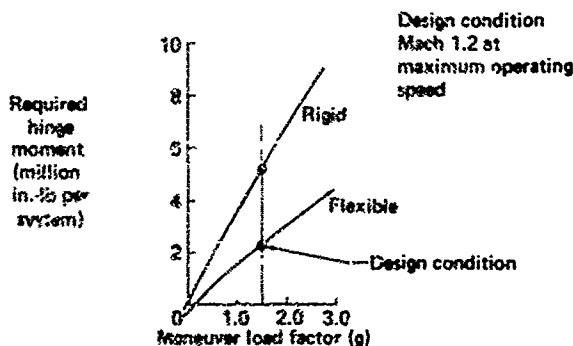


Figure 14.—Effect of Aeroelasticity on Horizontal Tail Hinge Moments

effect that aeroelasticity can have in a critical area of flight control system design, where the determination of hydraulic system power requirements is a vital decision impacting flight safety as well as aircraft weight and performance. The quality of the aeroelastic analysis itself thus becomes a dominant concern when specifying flight control system design requirements.

The Aeroelastic Design Cycle

The foregoing examples highlight a few of the stability and control design areas that are critical from the standpoint of structural flexibility effects for a supersonic transport. These are straightforward examples with application to either the CCV or conventional stable vehicle design approach. For the CCV design approach, flight controls are used to an unprecedented degree

for aircraft performance improvement, and the aeroelastic analyses assume further importance in the assurance that the flight control system is dynamically stable and compatible with the aircraft structural properties. The aeroelastic analysis, thus, takes on a vital new role in the design of a control-configured supersonic transport, becoming a key element in the configuration design cycle. Major configuration design decisions such as placement of the wing on the body, determination of the operational center-of-gravity range, the location, size and geometry of the tail surfaces, the type, size, location, and power requirements for flight control surfaces, and the type and weight of various structural areas are paced by the engineering efforts involved in aeroelasticity analysis.

In the U.S. SST program, the entire aircraft design development process became paced by the aeroelastic design cycle. This design cycle is a complex process involving the interaction and integration of all of the technical design disciplines. A proper discussion of this element of aircraft preliminary design would constitute sufficient material for a complete paper in itself and thus will not be attempted herein. However, a very brief discussion will provide some insight into the importance of this area in the design process.

Figure 15 presents a greatly condensed description of the aeroelastic design cycle, showing only the basic interactions between the structural analysis and flight controls/stability and control disciplines. The equally critical design cycle work of the other

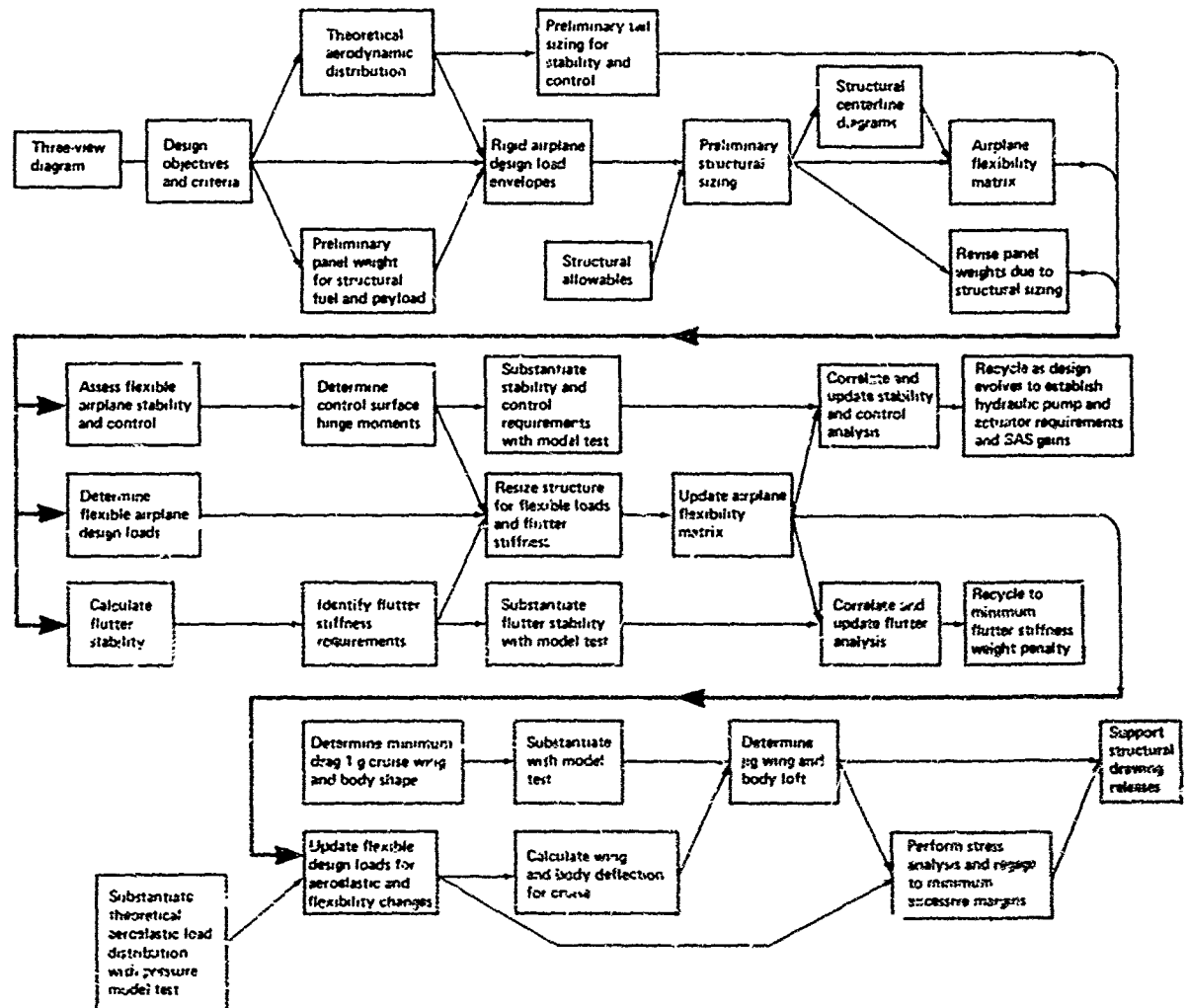


Figure 15.—Flight Control and Structures Staff Interactions in Aeroelastic Design Cycle

technical disciplines, such as aerodynamics, propulsion, weights, and systems, that interact with and support the structures and controls work, is not shown. The complete design cycle consists of the integrated workings of all of these disciplines interwoven into a complex supply and demand system of analysis, test, data base development, and decision making.

The initial activity consists of configuration definition and a well detailed development of the design objectives and design criteria. The configuration definition establishes the aircraft size, weight, and balance from the preliminary estimates of the aerodynamic, structural, and propulsion system configurations required to meet the payload/range, noise, field length, and economics goals. The stability and control analysis is a key part of this phase in establishing the airplane balance and configuring the tail surfaces. This activity locates the wing on the body and sizes the empennage surfaces, thus strongly influencing the areas of structures weights, and aerodynamics. The early estimates are based on theoretical analysis and hopefully some wind tunnel testing conducted during earlier research phases. Critical flight conditions and the degree of benefit from stability augmentation are assumed, and quasi-static aeroelastic corrections, including distributed mass effects, are estimated to establish the inherent stability and aerodynamic control levels required.

Having established the design criteria and the configuration base, the preliminary design work begins with estimation of design airload distributions, structural paneling, and other work to develop the airplane flexibility matrix and structure sizing that are required to support the initial assessment of stability and control, design loads, and flutter stability. The stability and control efforts during this phase refine the longitudinal balance estimates, configure the tails and all control surfaces, and develop the initial designs

of the stability augmentation and flight control systems, including identification of SAS gains, authorities, control laws, control surface deflection ranges, rates, and power requirements. The technical data base for this phase of the stability and control analysis is provided by wind tunnel testing and aerodynamic theory. The stability and control work is supported by the structural analyses that provide the effects of structural flexibility on the aerodynamic derivatives and develop the distributed mass aeroelastic effects for the range of airplane loadings calculated over the flight envelope. A detailed modal representation must also be included, if active controls are to be employed for stabilization of an otherwise unstable vehicle, or for structural load alleviation or ride qualities improvement.

If the preliminary design phase is successful in converging on a workable airplane configuration, the design cycle continues into a configuration validation phase. This is a complex design iteration process involving structure resizing, weight-and-balance adjustments, aerodynamic modifications, and adjustments to the configuration and systems to meet design requirements and optimize the overall configuration. This process is a continual exercise in adjustment and compromise as model testing and analyses provide further understanding of the design changes necessary to assure substantiation of aerodynamics, structural, and control system stability. Throughout this design iteration process, the flight control system design plays a crucial role in the configuration development. Since the control system is being employed to stabilize as well as control the vehicle, the limits of control system/structural dynamic stability must be established with accuracy. The final phases of the design cycle provide detailed configuration and systems definition and design validation required to proceed with confidence into the aircraft detailed design phase.

The entire design cycle described in the foregoing and shown in figure 15 will require anywhere from 12 to 24 months, depending on the complexity of the aircraft configuration, the assigned manpower, test facilities scheduling, and degree of computerized analysis in each technical discipline.

Current research efforts within industry and government are striving to speed the airplane design process through the development of computerized preliminary design systems. These programs can be extremely effective in streamlining the preliminary

design process and are already in use throughout the industry. There is, however, a very real problem that is not receiving sufficient research attention at this time; that is, the quality of the aeroelastic analysis is severely lacking in critical design areas, particularly for supersonic aircraft. The critical design areas for aircraft stability, controllability and structural loading usually occur at conditions of high load factor and involve controls deflected through large angles. For supersonic aircraft, the design conditions usually fall within the transonic area, where considerable uncertainty exists regarding aerodynamic flow characteristics. These flight conditions involve nonlinear aerodynamic effects such as shock-induced boundary layer separation, vortex flows, and airfoil pressure limiting. Under these conditions, our basic aerodynamic analysis tool, linearized theory, is not satisfactory. An example of this is seen in figure 16, which presents a simple comparison of measured pressure distribution with that predicted by linear theory for the rigid delta planform SST horizontal tail at 0.95 mach. For this case, the tail was down-loaded substantially with a tail leading-edge-down incidence of 10° and elevator trailing-edge-up deflection of 17.5° . Considerable difference is seen between the theoretical and experimental pressure distributions, particularly over the aft portions of the airfoil near the elevator hinge line (about 75% chord).

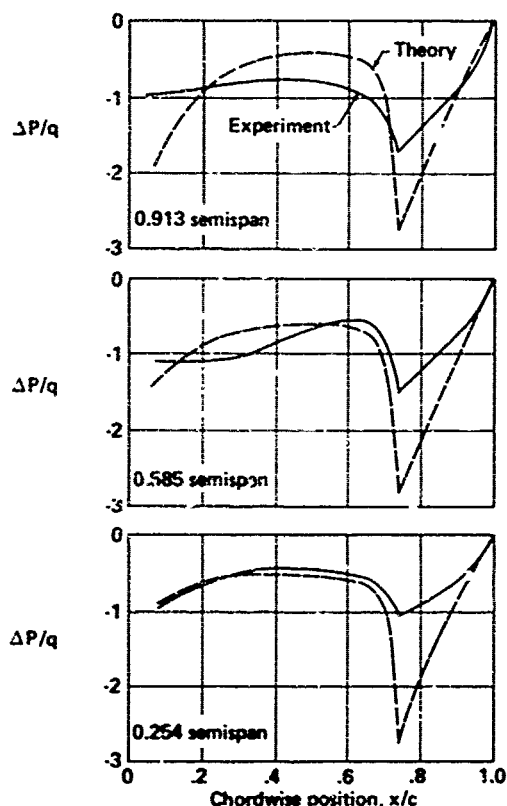


Figure 16.—Predicted versus Experimental Rigid Pressure Distributions for SST Horizontal Tail at Mach 0.95, With Tail Deflected 10° Leading Edge Down and Elevator 17.5° Trailing Edge Up

A program is needed to develop and verify methods of predicting stability and load distribution of elastic airplanes where the flight conditions involve transonic operation, high angles of attack, and large control deflections where nonlinear aerodynamic effects predominate. A worthwhile effort would involve a coordinated program of aeroelastic model testing and pressure model testing to develop empirical corrections to be applied to aerodynamic influence matrices developed with linearized theory.

An obvious conclusion of the Boeing SST development work is that any coming generation of supersonic cruise vehicles will rely on electronic flight control systems to an unprecedented degree for stabilization, load alleviation, and ride quality improvements. Research to improve the quality and speed the completion of the aeroelastic design cycle is sorely needed. This work

will assuredly be rewarded with substantial payoffs in reduced engineering costs and risks and in improved aircraft safety, performance, and economics.

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RECENT EXPERIENCE FROM B.A.C. AIRCRAFT FOR NATO

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SUMMARY

The paper draws on B.A.C. experience of national and international aircraft development programmes in an analysis of trends in total Cost of Ownership of combat aircraft. This Life Cycle Cost analysis is related to the Air Force budget, and some ways are considered by which Industry, Procurement Agency, and Air Force might alleviate the rising cost of Air Force operations.

Introduction

Figure 1 illustrates the extensive background of experience on which BAC can draw. This is a brief history of the aircraft built by the Military Aircraft Division of the Company. The bars indicate the timescales from 1st flight through the production history.

One of the symptoms of the operating cost problem is already apparent in fig. 1, namely the decline in the rate at which new projects have been initiated. Considering that in the days of Canberra there were many other new projects being started in other companies in the U.K., whilst in the 1970s the Jaguar and MRCA programmes are absorbing a much greater part of the front line aircraft budget, it is clear that the trend is even stronger than indicated by this diagram. TSR2 cancellation was proof, if any is needed, that budget constraints will override both military need and technical success.

Since Jaguar and MRCA experience contribute significantly to the content of the paper, it would be appropriate to identify briefly the stage which these projects have reached. Figure 2 shows a line-up of Jaguars: this aircraft is now going into full scale production, deliveries to the Armee de l'Air and Royal Air Force exceeding 30 at the time of writing.

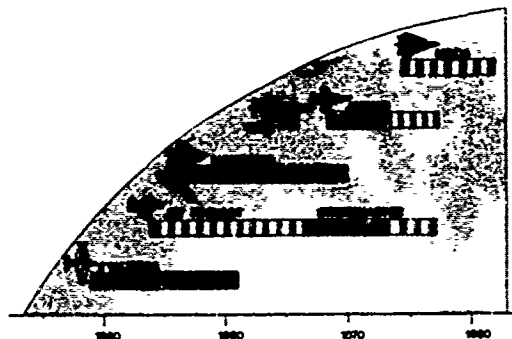
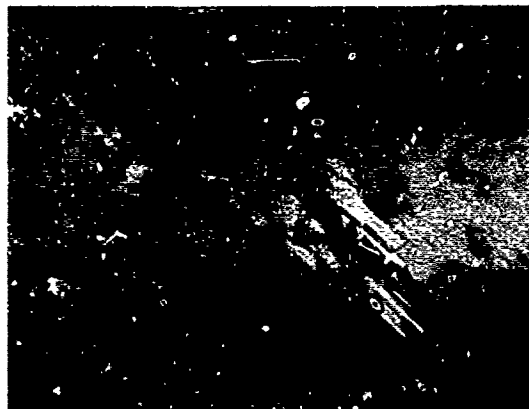
Fig.1 BRITISH AIRCRAFT CORPORATION
MILITARY AIRCRAFT DIVISION

Fig.2



Fig.3



The first MRCA Prototypes are in an advanced stage of construction. Fig. 3 is an artist's illustration of the aircraft, which is used to avoid unnecessary classification of the paper. The experience of international collaboration which we have gained on these two projects will be referred to later: the Jaguar with AMD-Breguet in France and the MRCA with MBB and Aeritalia in Germany and Italy respectively.

Airforce Budget Trends

Returning to the main theme, Figure 4 shows that the trend of lengthening life in service of combat aircraft is compounded by the evidence of declining fleet strength. Somewhat over a quarter of these totals in recent years have been front line combat aircraft, and the remainder of the discussion concentrates on this segment. By the 1980s Jaguar and MRCA will form a very dominant proportion of this combat inventory and will also constitute a significant part of several other European Air Forces.

Since lengthening service life and diminishing numbers are evidently due to budgetary constraints to a large extent, consider the trends which are evident in the Air Force budgets of several NATO nations, extracted from published data and condensed into Figure 5.

Despite differences and uncertainties of definition, the common dominant characteristic evident in all these budgets is the rise in the proportion necessary to cover manpower costs in operating the Air Force. Included in the manpower bracket are such items as pensions and support services which can be directly attributed to maintaining this manpower. The next block is the budget required for spares and other operating expenses, fuel and so on. These costs are, of course, determined by the characteristics of aircraft purchased in the 1950s and 60s, and at least until these are replaced the amount required for operations must remain roughly constant. Finally, as a residuo at the top is the funding available for new equipment including research and development. The extrapolation of this trend is of course a hazardous business, and in illustrating the top band diminishing towards zero in the late 1980s, it is assumed that the real level of total military budgets remains roughly constant. In essence, this trend reflects results already reported by other investigators, notably in the Mommer Report in Germany (ref. 1).

This depressing picture prompts the question:

How could the Air Force, Procurement Agency, and Industry be motivated to improve operating costs (including manpower requirements), for the benefit of "next generation" aircraft budgets? This motivation is inevitably difficult to perceive, since any improvement may well imply conflict with more immediate interests: for example in performance demands by the Air Force; in diverting R & D expenditure into mundane reliability problems; and in the implications of reduced levels of spares and overhaul business to industry.

This motivation problem is, perhaps, already partly answered by the implications of figure 5. Without significant changes, the Air Force would eventually have obsolete aircraft; the procuring agencies would have nothing to procure; and industry would have no new business. With the growing recognition of this problem in recent years it is tempting for each party to look for ways of blaming the others. Thus, for example, the Air Force could demand of industry "Provide better reliability and maintainability"; but tending to ignore the cost, risk, weight, performance penalties and to forget about who pays. Similarly, the procurement agencies can complain about rising costs of aeroplanes (but who specifies the performance and complexity standards?) or industry can point the accusing finger at the "extravagance" of Air Force flying operations without acknowledging that the proficiency of the Air Force in wartime might be at stake.

Each of these responses contains, in essence a reasonable question. Accusation does not provide answers, however. The objective of this paper is to suggest that fair and reasonable solutions are possible, through mutual analysis and evaluation of the problem by all the parties concerned. At the present time, the structure of the aircraft procurement process does not seem to encourage the mutual thrashing out and understanding of all the interacting interests which the situation demands.

Fig.4 RAF AIRCRAFT INVENTORY

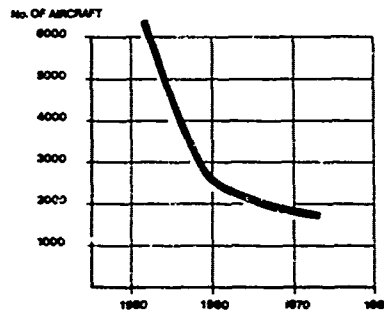
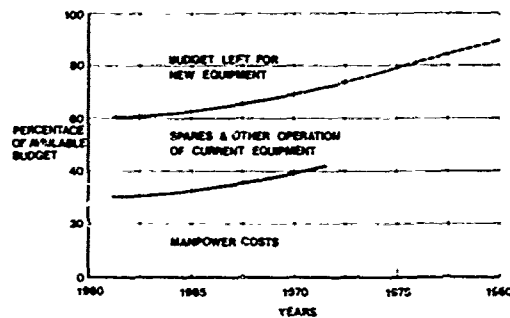


Fig.5 BUDGETARY TRENDS IN NATO AIR FORCES



The Procurement Process

Figure 6(a) illustrates the author's impressions of the functions performed by the three parties in generating a new combat aircraft project, and the major communication links. In essence, the only part of the Air Force function which gets through to industry (and that often via the procurement agency) is the specification, which in many ways presupposes a particular solution of the wartime task. Certainly some elements of the peacetime task are represented in the specification in the form of demands for reliability and maintainability, but the basic operating philosophy and the functional needs of the peacetime task are not really considered as matters of interest to industry.

Fig. 6a TASK BREAKDOWN (PRESENT)

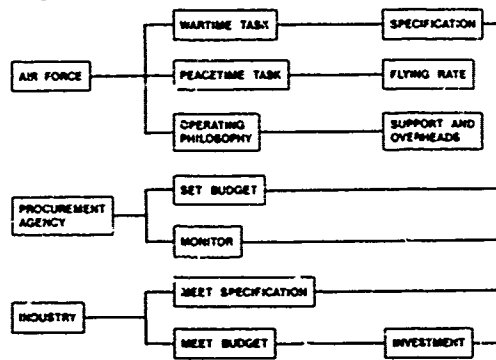
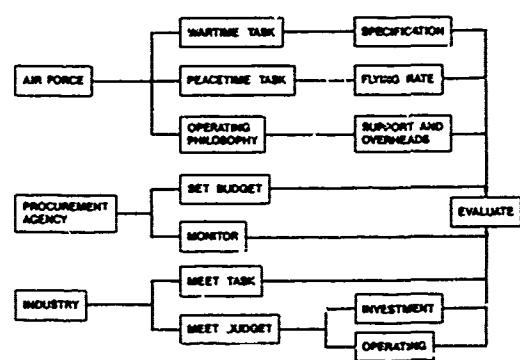


Fig. 6b TASK BREAKDOWN (PROPOSED)



The tendency for the compromises between specification and budget to be determined before industry has any real hand in the project is happily giving way to a process of earlier consultation. This however, still tends to be a one-way process, in which industry's contribution to the optimisation process is hampered by its minimal participation in official evaluation studies. On the cost side, there is of course very close monitoring by the procurement agency of industry's expenditure and future estimates, but this has tended to be limited almost exclusively to the investment phase, that is research, development and initial procurement. A slightly modified version of this diagram is illustrated in fig. 6(b), incorporating some remedies to the above criticism.

The key point is the inclusion of a joint evaluation process, in which all aspects of Air Force requirements can be examined in perspective with all aspects of the cost of ownership. Thus, industry would become more involved with meeting the total task, (in place of an aircraft specification) integrating engineering studies, cost and operational analysis into a single iterative process. The need for operating costs to be added in the monitoring process has already become self-evident. In the absence of a co-operative evaluation process, BAC have developed a preliminary life-cycle cost model: this study does not at present have the benefit of officially approved data in some areas, although there is encouraging evidence of widespread interest and cooperation at an informal level in the U.K., both in the procurement agency and the Air Force.

Life Cycle Costs

If life cycle cost analysis is to be convincing, it must be capable of matching and explaining the breakdown of past budget allocations. Recognition of the main driving parameters controlling constituent elements of life cost should lead directly to some indications of areas in which industry and the Air Force could influence these significant parameters to obtain maximum effectiveness within total budget constraints. At the same time, the functional basis for the extrapolation of diagrams like Fig. 5 should become apparent. For example, manpower costs are the product of manpower strength of the air force and wage rates. There is an inevitable tendency for manpower pay scale inflation rates to exceed national inflation indices, since the latter always to contain an element of material cost which is normally increasing at a lower rate. Furthermore the decline in military conscription over the years has tended to force a rise in pay scales at times considerably in excess of national average wage inflations. Skill levels required have increased, adding further to the need for competitive pay scales. Such changes may be transient, and therefore not amenable to extrapolation.

It is not proposed to develop this theme into a quantitative analysis in this paper, though it should be included in a serious evaluation.

The more fundamental question of manpower strength level must also be related to its functional requirements: consider the implications of fig. 7.

RAF manpower numbers have tended to decline with the inventory strength of the Air Force but not at the same rate. The rather rapid growth shown in Figure 7 in numbers of men per aircraft which took place in the 1950s was presumably due to increases in the performance and complexity of the aircraft. It is vital to know what features of aircraft complexity led to this growth. Did complexity stop increasing during the 60s, or was further increase offset by valiant efforts on the part of the Air Force to streamline operations and reduce overheads? Statistical extrapolation will not answer these questions.

It should be noted at this point that increasing complexity, and, if necessary, manpower and cost per aircraft is not necessarily a bad thing, if it is accompanied by a genuine increase in the effectiveness per aircraft. This has certainly been true in such features as payload delivery accuracy and survival characteristics of recent aircraft. However, intuition suggests that improvements in quality cannot indefinitely maintain the balance if numbers continue to decline at the rate they have in the recent past.

It is not the objective of this paper to examine the measurement and trends of aircraft quality: we have computer modelling techniques now available capable of fairly detailed evaluation of operational effectiveness in various military roles. Providing we assume that such methods are used to justify (or reject) any increase in technology standards or complexity of future aircraft projects, it is reasonable to study the trends of life-cycle costs in the context of an aircraft typical of present generation standards. It is not possible, in any case, within the scope of this paper, to consider the multitude of directions in which technology might develop.

For illustration purposes, and to avoid getting involved in sensitive commercial and possibly security hazards on actual aeroplanes, a hypothetical aircraft has been defined with the following characteristics and designated F'X':

HYPOTHETICAL F'X'

CLEAN T.O.W. 10,000 Kg
 MACH 1.5
 2 x 4,000 hp ENGINES (NEW)
 1 SEAT
 FIGHTER/ATTACK AVIONICS (EXISTING)
 IN SERVICE 1965
 15 YEAR PLANNED LIFE
 PRODUCTION 200

It was intended that this F'X' should have characteristics somewhere between, say, Hunter and Phantom, and it is not important to the argument exactly where this "typical" aircraft is placed in the spectrum. A full scale investigation would require analysis of each actual aircraft in the inventory.

Fig. 8 is a breakdown of the life-cycle costs of the F'X' on the left, using the BAC Life Cycle cost model, taken at 1970-71 economic conditions for wage rates and material costs. More detail of this breakdown will become apparent below. On the right, the same aircraft life-cost is compared, on a percentage basis, with the budget breakdown, using the same subdivisions as figure 5.

The equivalence of the two right hand columns is sufficiently close to conclude that the assumptions in the life cost model are reasonable. The budget breakdown, of course, covers all types of aircraft (combat transport, training, etc.) and mixes operating costs of 1960s aircraft with procurement of 1970s aircraft and R & D of

Fig.7 RAF MANPOWER/AIRCRAFT

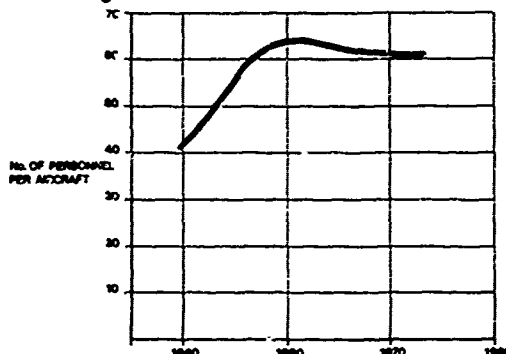
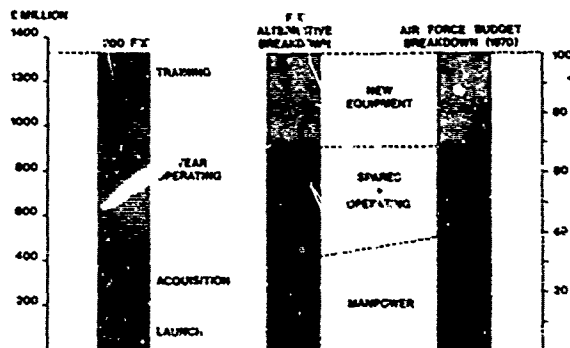


Fig.8 LIFE CYCLE COSTS



aircraft which may appear in the 1980s. Variations in assumptions about the quantity procured (200 in this case) and the life in service (assumed at 15 years) would also produce variations in the relative magnitudes of different elements of the cost build-up.

The main parameters associated with each block of the left-hand column are discussed below, and the kind of changes which might have appeared if this same aircraft, F'X', had been a product of the 1970s are illustrated.

Launch and Acquisition Costs

Fig. 9 indicates a typical parametric/statistical approach to cost estimation which has been the subject of many papers by Rand and others. BAC use this method increasingly in keeping track of costs during the early phase of new project studies. Sufficient real data has now been collected at BAC to provide a reasonable confidence level that the main controlling parameters are realistically represented. The numbers inserted on this and later block diagrams are those relating to the hypothetical F'X' aircraft.

This approach to costing avoids the tendency to optimism which afflicts most detailed task estimating methods; the records of what was actually spent in past programmes are modified by relative complexity factors which have themselves been validated by comparisons of past programmes. The danger of optimism in the inputs (particularly sizes, weights, equipments) must still be avoided. The central blocks of the diagram concern policy factors which may introduce significant departures from previous experience, and which may not always be predictable at the conceptual phase of design.

It would take too much space to go into details regarding the form of the inputs on the left of fig. 9; everyone in the business will have his own version anyway. Some aspects of the first two policy factors indicated, development philosophy and collaboration, are however worthy of comment.

Development Philosophy

This is a very large subject which could occupy a separate paper on its own. By way of definition, the management options considered under this heading are development batch programmes, prototype programmes (or mixes of these two), and development stretch of existing types.

It is not proposed to develop a formal analysis of the various implications of cost, risk, and timescale in this paper. Sufficient to note that each option has its place in some circumstances: development batch for minimum timescale on low-risk programmes; prototype programme for minimum risk (particularly with technology innovation); development stretch for minimum launch cost (but not necessarily best total life cost/effectiveness). The assessment of the balance between objectives of time and cost predictions, and the risk of exceeding these predictions, is not yet a science. BAC experience of many varieties of national and international programmes tends to indicate that some compromise between the development batch and prototype approaches may frequently be the best choice.

The basic F'X' costs of fig. 8 were based on such a compromise policy, and it is not considered that major changes would be introduced if this aircraft were being developed in the 70's. The opportunity for saving through the option of development stretch of the "existing" aircraft depends mainly on whether the basic performance limitations of a 1958 design (for 1965 in service) will be capable of survival in the changed defence scenario of the 1980's. This possibility is discarded for the purposes of the current analysis, though it might exist in some special cases.

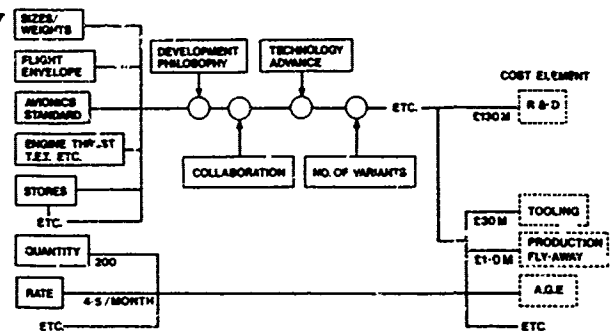
Collaboration

International collaboration agreements have so many possible varieties of political, administrative and commercial character that a neat classification is impossible. Certainly it would be almost inevitable that an F'X' of the 1970's would have a collaboration base: some attempt must therefore be made to define a framework for analysis of cost implications. The following very brief comments are concerned primarily with the "engineering" aspects:

(a) Design Authority

Collaboration on design authority, whereby several industrial and Government partners all have to agree at all decision levels, can lead to extremely heavy factors for duplication of studies in depth, possibly extending even to significant test programmes. However, it is hardly conceivable that the total cost, divided by the number of participants, would grow to the level a single

Fig. 9 ACQUISITION COST-STATISTICAL



nation effort, even with the most vigorous disagreements amongst the partners. The very real benefits sometimes obtained from melding a variety of viewpoints into the design should not be overlooked.

(b) Design and Development

If the design authority question is reasonably clear cut, the cost of sharing development work is small by comparison with the benefits of sharing the bill. However, there is inevitably a large amount of legwork involved in keeping the decision processes roughly in line with the development work, and work sharing agreements, for political or financial reasons, are liable to result in divisions which require extensive administration and some duplication.

(c) Component Manufacture

On the production side, the manufacture of parts can, of course, be split with the benefit of substantially increased quantities and with quite modest costs for liaison and transport once the process is properly under way.

(d) Final Assembly

Sharing final assembly in several different locations obviously requires duplication of assembly tooling and loss of the potential learning benefit in this area. Perhaps the need for separate national assembly lines will gradually disappear as integration within EEC becomes more firmly established.

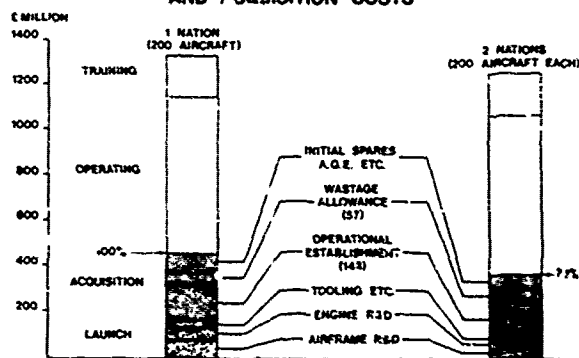
(e) Political factors

Finally, it must be noted that collaboration produces an element of stability. It is questionable whether Jaguar would have survived, as a national project, the radical changes in British and French plans for the roles of the aircraft - a major shift from training to strike roles on both sides. Incidentally, the tendency for collaboration to lead to several different variants of an aircraft, whilst adding inevitably to costs, may confer some benefits of flexibility in Air Force planning, and in export sales. Direct cost penalties may well be very small compared with separate projects for each role, provided that basic mission requirements are reasonably similar - e.g. strike, recon, and strike training. However the compromises required to achieve multi-role airframe compatibility might generate significant changes in the initial conceptual design parameters, with impact both on cost and effectiveness. All these factors can be realistically evaluated with the computer analysis tools now at our disposal.

Summarising, the total cost of R & D for a collaborative project may lie anywhere from 20% to over 50% above the single-nation cost, depending strongly on the extent of decision sharing and multi-role compromises. Sharing this higher total cost amongst the participating nations may yield savings of 25-40% against single nation costs. Production costs are reduced slightly by learning benefits, providing there are not too many variants.

Consider now the impact of a 2-nation collaboration arrangement on the R & D and initial procurement costs of the F1X1 aircraft in figure 10. The type of programme assumed here would remain the mixed prototype/development batch variety, and we have assumed that the more expensive pitfalls of collaboration have been avoided, and that the two nations have agreed upon the same configuration of the aircraft. The right hand column shows that some 20% of the total launch and acquisition costs would be saved. This is of course now fairly common practice in principle, though political compromise often still detracts from the ideal efficiency. The sub-divisions of the columns show how the various cost elements are affected. It is worth noting that the actual operational establishment aircraft in this analysis totals 143, the remaining 57 (making up the 200 production per nation) being an allowance for wastage over 15 years in service.

Fig.10 COLLABORATION EFFECTS ON LAUNCH AND ACQUISITION COSTS



It will be apparent that life-cycle cost savings through such collaboration may be only about 6-7%, emphasising the need for proper evaluation of role compromises.

Operating Costs

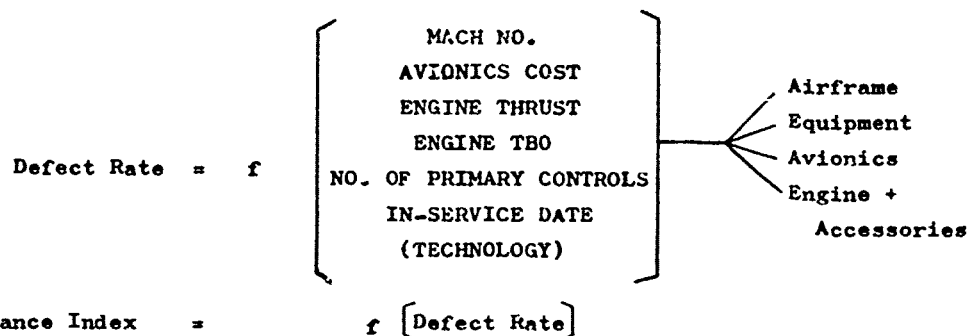
The main elements are noted in fig.11. Again some figures are noted for the hypothetical P'X' aircraft.

Key inputs having the greatest effect on the operating cost are system unit cost, reliability and maintainability, and peacetime flying task.

Since the aircraft is a "fixed quantity" in this analysis, its system costs are similarly fixed, as are the engine and aircrew. The peacetime flying task is considered further below: for the present, 300 flying hours per year per aircraft are assumed. Consider first the influence of reliability and maintainability. Not only engineering manpower (with its overhead of about four Air Force and civilian "support" men to every aircraft maintenance man), but also spares and overhaul costs are directly related to the defect rate. Spares and repairs of equipment and engines cost substantially more in the life of the aircraft than the initial procurement costs. In addition, the wartime sortie rate (and hence effectiveness) decreases as the repair workload increases, defect rate again dominating.

In view of its profound importance, such limited data as is available has been collected and scrutinised for some measures of a complexity index which might explain or determine trends in reliability and maintainability (R & M).

The main parameters which have emerged from this preliminary study are noted below:



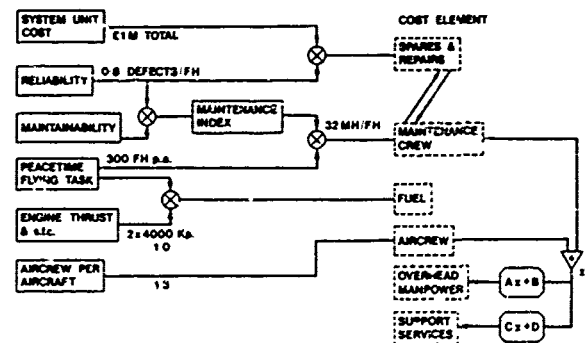
$$\text{Spares Cost per Flying Hour} = K \times \frac{\text{System Defect Rate} \times \text{System Cost}}{\# \text{ System Components}}$$

Clearly the random nature of the problems which arise in this area make it impossible to be precise at a detailed level. However, it has been possible to get a reasonable correlation with available data by using the broad system breakdown indicated. The appearance of parameters similar to those used in most statistical cost formulae is not surprising, as these are primarily indicators of "complexity" in the sense of numbers of parts, weight sensitivity etc., which also have adverse effects on reliability. Sheer size and weight, however, which are primary cost parameters, do not appear. The "avionics" label is attached to all instruments and systems which contain electronic or "delicate" electro-mechanical elements (but not main electrical power supplies). The inclusion of engine thrust is questionable: however, its influence is small, and qualitatively justified by the trend (with exceptions) of more stages, more complex cooling etc., with increasing size. Extra control systems (e.g. for a VTO aircraft) are allowed for on an estimated major parts count basis. The technology factor represented by in-service date appeared to make a fairly small contribution over the period 1950 to 1967, but the extrapolation of this factor requires evidence of more recent trends, discussed below.

As a first approximation, it has been possible to define maintenance index (that is, maintenance man-hours per flying hour), as a function simply of defect rate for the range of aircraft for which data was available. This relationship implies that broadly the same characteristics of complexity which influence reliability will also tend to imply greater congestion and difficulty of access for repair. This is obviously not entirely a functional relationship, and must be used with caution for extrapolation.

The relationship noted for spares and repairs costs is largely intuitive, and some constants have been chosen to fit with the rather inadequate available data. Some refinements are planned in this area.

Fig. 11 OPERATING COSTS



For the present, the justification for using these simple relationships is that they appear to work for a reasonable range of aircraft. Fig. 12 shows the empirical "complexity index" used as a predictive parameter for Maintenance Index.

Nine of the points on this diagram are taken from either recorded data or brochure estimates by manufacturers on aircraft which entered service between 1950 and 1967. They cover the range from basic trainers to advanced multi-role fighter/attack aircraft, with correspondingly wide variations in performance and avionic equipment. Several of these points represent data which was acquired after construction of the complexity factor formula, adding unexpected confidence. Adjustments have been made where necessary to all data points to unify definitions, to include all first and second line manpower, including armourers.

Two points are identified as forecasts for aircraft entering service in 1972. The fighter lying near the mean line is a development stretch of an existing earlier generation aircraft, and it is clear that this has inhibited any major change in the R & M characteristics. The attack aircraft point lying well below the average line is in fact the current prediction for Jaguar 'S' (the target is slightly lower at $10\frac{1}{2}$ man-hours per flying hour). In view of the good correlation of all the other data points, one would tend to regard this point with some degree of scepticism. Whilst it cannot be claimed that this prediction is confirmed at this early stage in the life of the aircraft (it depends on several thousand hours of learning experience in operational squadrons), some very powerful evidence is already accumulating that the trend will ultimately meet or improve on this target.

The achieved defect rates per flying hour of Jaguar and Lightning are compared in Fig. 13, and show that prototypes of the Jaguar 'S' version at under 1000 hours U.K. flying experience have already passed the Lightning service achievement at over 100,000 flying hours. The corresponding predictions from the statistical analysis would have led to the levels on the right-hand side: that is, closely similar defect rates at the end of the learning curve. There would have to be a very unexpected kink in the Jaguar learning curve if it were to finish above the prediction and target levels, so that confidence is rapidly increasing that this aircraft will demonstrate more than double the reliability of previous aircraft of similar complexity.

The fact that Jaguar prototype flying rates have been roughly double those achieved in Lightning appears to reinforce this confidence.

It should be noted that prototype flying of the French 'A' and 'E' versions (which have much less avionic content than the 'S') confirm their lower defect rates; this data is not illustrated owing to differences in defect recording rules. The UK data includes all unscheduled actions, whether defects were confirmed or not.

It would be useful to be able to indicate exactly how much it has cost to achieve this improvement in R & M technology. Unfortunately, it is extremely difficult to segregate those parts of costs of equipment development, design for access, and so on which contribute to this achievement; the same applies to weight and performance penalties. A significant part of the gain is attributable to equipment selection with reliability as a major factor in the choice. Where unproven equipment was unavoidable, suppliers' records in reliability were considered, and where possible complete systems were obtained from one supplier. Penalties were probably in weight more than cost. Direct costs of identifiable work on R & M amounted to about 7% of the airframe design and prototype build costs. (Ground and flight test work relating to R & M was not specifically recorded as such, and therefore cannot be ascertained). In future projects where existing equipment of proved reliability is not available, the cost penalty may become greater, and with it the need for more rigorous justification of expenditure. Possibly one of the urgent requirements for research investment is to undertake some pilot experiments to measure the cost of improved reliability.

Fig. 12 MAINTENANCE INDEX PREDICTION

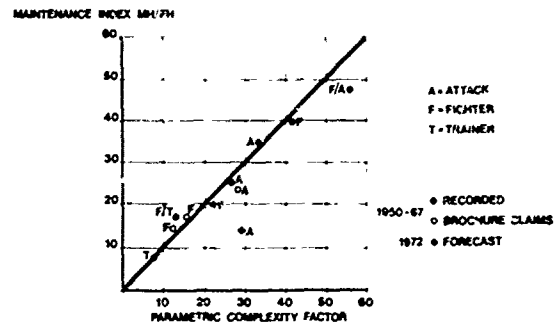
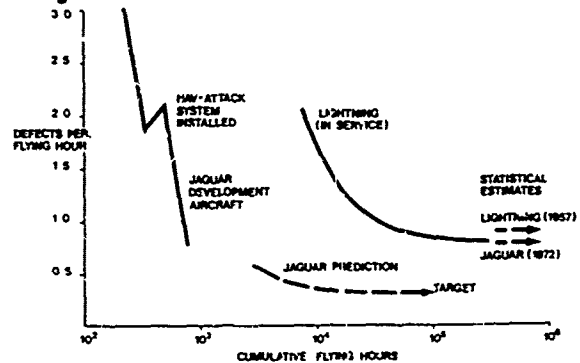
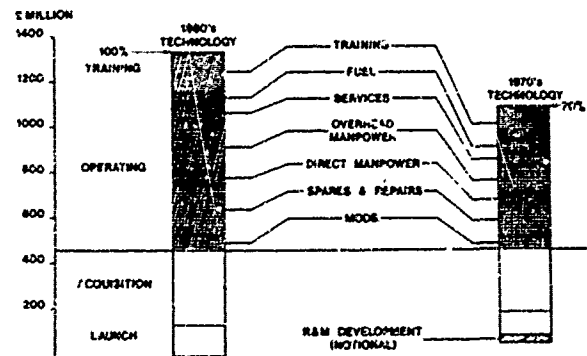


Fig. 13 TRENDS OF RELIABILITY IMPROVEMENT



If the improvements in reliability and maintenance index predicted for Jaguar are assumed to be achieved on the F15 in its 1970's version, implications for future operating costs are illustrated in fig.14.

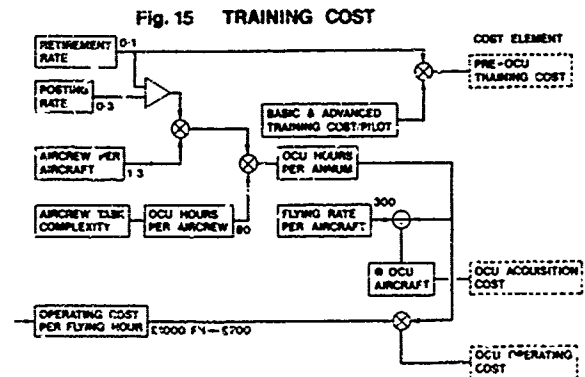
Fig.14 RELIABILITY AND MAINTAINABILITY IMPROVEMENT



The significant savings are, of course, in the manpower and spares costs. The direct manpower (which includes maintenance men and flying crew) occupies 15% of the manpower plus services budget at present levels of Air Force "overheads". The savings in spares, repairs and manpower reflect also in the pilot training costs, since the major element of these costs is the flying cost of aircraft used for operational conversion training. An overall saving of about 30% on the total operating and training costs is indicated. A small and entirely notional allowance is shown at the bottom of the left hand column as the cost of improvement in reliability and maintainability. As illustrated, it represents over 30% addition to the total research and development costs of engines and airframes. This would seem to be generous by any standards as an allowance for a task which has probably already been achieved on Jaguar. Nevertheless, this expenditure is trivial when compared with the savings.

Training Costs

Although included in the assessment of operating cost improvements, there are other aspects of pilot training having significant cost implications. The costs of pilot training occupy a share of the budget approximately equal to the total fly-away procurement costs of the aircraft fleet. This pilot training cost is dominated by two parameters: the replacement rate of pilots moving out to other jobs, and the operating cost per flying hour of the training aircraft (fig. 15). It is assumed here that the number of hours of flying training required is fixed by the task and aircraft complexity, in this case at 80 hours per pilot. This O.C.U. element of the training process accounts for about 85% of total training costs. The basic and advanced training is included for good measure but not discussed in detail here.

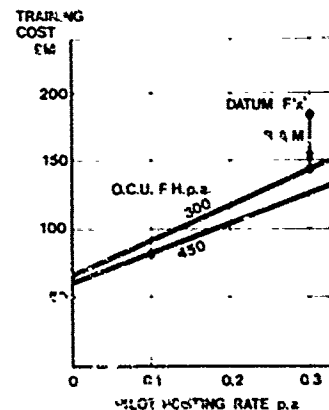


The total "production task" of the training process is essentially registered by the number of operational conversion hours per annum required. The ratio of this to annual flying rate per aircraft determines the number of O.C.U. aircraft required and hence their acquisition cost. The total flying hours are charged at the appropriate operating cost per flying hour, as derived previously for the operational aircraft, leading to the total operating cost of the O.C.U.

Assuming that the improvements in R & M shown in fig. 14 are achieved, it would seem possible to visualise increasing the flying rate per O.C.U. aircraft substantially above the 300 hours per annum which are typical today, say to 450. This would reduce the number of O.C.U. aircraft required in the inverse ratio.

Even greater economies could be visualised if the Air Force were prepared to decrease the rate at which pilots are posted. There is already some evidence of a trend in this direction, and the introduction of more multi-role aircraft will also decrease the training required on second tours of duty. For the sake of illustration, training costs for a posting rate of 10% per year have been evaluated against the 30% used in this initial study. The effects of these changes are illustrated in fig. 16.

Fig.16 EFFECTS OF AIRFORCE TRAINING POLICY

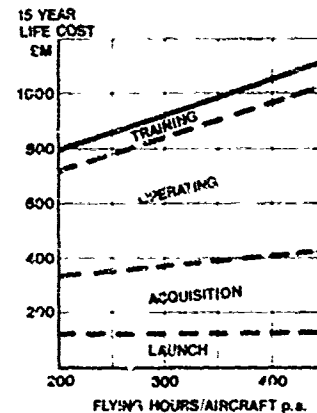


The original training cost is shown as being reduced firstly by reliability and maintainability, secondly by increased flying rate on to the lower line, and thirdly by the reduction in pilot posting rate. The latter two features yield savings of about 40% in training costs.

Peacetime Flying Rate.

Another question concerning Air Force operating policy relates to the flying rate demanded of operational squadrons in peacetime. Some of these will have specific defence roles to accomplish which dictate flying requirements. However, for flying which is essentially done to "keep the pilot's hand in" at his operational task, one could postulate that in exchange for longer tours of service in one aircraft type, pilots might be expected to remain adequately qualified at a lower flying rate, say at 16 hours per month per pilot in place of the current 20. In terms of aircraft flying hours, this means a reduction from 300 hours a year to about 250. Savings in manpower, spares and fuel are achieved, whilst wartime achievable sortie rates should be maintained well above the capabilities of current aircraft (despite a reduction in maintenance manpower) by virtue of the expected improved R & M characteristics. In addition, the reduction in operating costs achieved could be devoted to a larger fleet procurement. Fig. 17 shows the total cost variation with flying rate over a wider spectrum, assuming the 1970's standard of costing for the F1X aircraft.

Fig.17 EFFECTS OF PEACETIME OPERATING POLICY

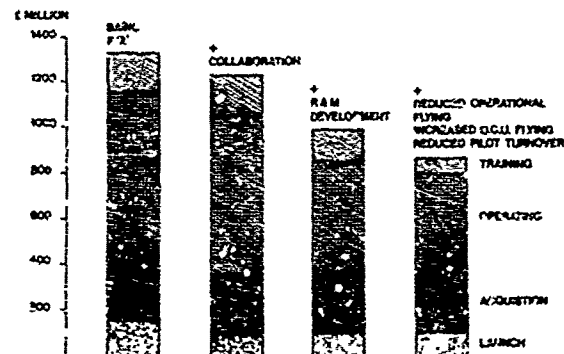


Conclusions

Fig. 18 collects together all the areas which have been suggested for economy: collaboration, R & M, leading to the feasibility of Air Force operational changes. Some of these gains should become visible, rather than theoretical, as Air Force experience in operating Jaguar builds up. It would of course be unreasonable to expect such gains to be possible over the whole spectrum of the Air Force inventory (transports, basic trainers, guided weapons etc.). However, over the expensive section of combat aircraft, the kind of savings illustrated should be entirely realistic.

On many of these issues, particularly on questions of Air Force operating philosophy, industry can only make the suggestions not the decisions. It is suggested that industry could make a valuable contribution to a more comprehensive evaluation process, both in the context of integrated engineering studies and with its substantial techniques for cost and effectiveness analysis.

Fig.18 SUMMARY OF POSSIBLE ECONOMIES



If all the cost savings illustrated in fig. 18 were realised across the whole combat aircraft sector, and devoted to increasing the fleet strength, the combat aircraft inventory could be increased by at least 50% in effective strength, either by growth in numbers at current standards or by increase in effectiveness reflecting technology growth. Such technology changes would be incorporated only where they provide greater fleet effectiveness than could be achieved at the same cost by larger numbers of aircraft.

Alternatively, if justified by technology changes, and/or by changes in enemy defence posture, the fleet replacement interval could be significantly reduced within the same budget constraints. Clearly, it is unlikely that replacement intervals would be cut below, say, 10 years, but some compromise between shorter replacement times and increasing numbers could well be considered as the best way of using funds freed by lower operating costs.

Since most of the savings indicated can only be achieved in the next generation of aircraft, little benefit can be expected in the budget allocations of the next few years. The achievement of benefits even 10-15 years hence depends on recognition and evaluation of the trade-offs for projects now in the pipeline.

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Notes

The views expressed in the paper are those of the author and do not necessarily represent those of the British Aircraft Corporation.

Acknowledgments

The author is indebted to Mr. R.F. Creasey and Mr. I.R. Yates for their advice and criticism in the preparation of this paper.

F-15 DESIGN CONSIDERATIONS

by

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1. SUMMARY

The major design considerations of the F-15 air superiority aircraft are traced from the initial requirements, through the design, and into the flight testing. Selection of the overall configuration is discussed with particular emphasis on the wing, inlet, and secondary power design. The ground and flight test programs are briefly reviewed.

2. LIST OF SYMBOLS

A	Aspect ratio of span to mean aerodynamic chord
AMAD	Airframe Mounted Accessory Drives
APU	Auxiliary Power Unit
b	Span
BPR	Engine by-pass ratio of fan air flow to compressor air flow
C	Airfoil chord
C_D	Coefficient of drag
\bar{c}	Centerline
C_L	Coefficient of lift
fps ²	Feet per second squared
$g's$	Load factor—the normal acceleration expressed as number of gravities (32.174 fps^2) experienced in maneuver
JFS	Jet Fuel Starter
l	Distance
L/D	Ratio of lift to drag
M	Mach number
psf	Pressure, pounds per square foot
t/c	Thickness to chord ratio
T/W or T/TOGW	Ratio of engine thrust to airplane weight at, for example, takeoff
TR	Taper ratio of tip chord to root chord
W/S	Wing loading, ratio of airplane weight and wing area
Λ	Leading edge sweep
τ	Twist

3. PREFACE

From an arbitrary genesis in mid-1966, tracing the evolution of the F-15 airplane design continues to current flight test research and development. By 1966, Air Force planners had summarized tactical fundamentals from the Korean War and early Vietnam conflicts: (1) That, regardless of the initial speed and altitude of engagement, the air battles gravitate to low speeds and low altitudes as the adversaries strive for maneuvering advantage. (2) That the firing ranges of air-to-air missiles and guns have changed attack and disengagement tactics. (3) That superior pilot skill and tactics cannot completely compensate for an

inferior aircraft. In effect, these operational experiences stressed that airframe and engine design criteria should emphasize maneuverability rate-of-change characteristics as parameters equal in importance to those maximizing altitude, speed and range. This paper describes the rationale and trades performed by the designers to define the proper balance between dynamic maneuverability and classic steady state performance.

The corollary to emphasis on maneuverability is emphasis on stringent weight control. History, in effect, had to be reversed from a somewhat passive exercise in weight monitoring to intensive configuration control. In World War II, the Air Force fighter wing loading (weight to wing area ratio, W/S) ranged in the 30-55 pounds per square foot (psf) band. The W/S band was overlapped in the Korean War from 50-90 psf. Through 1965, the W/S range expanded from 80-150 psf as the fighter mission complexity included weapons delivery on ground attack missions and area intercept mission capability by multi-purpose variable sweep fighters.

From the days of F-X conceptual design analyses through F-15 production decision, all subsystems comprising the F-15 have matured in an environment of weight control more stringent than any other Air Force weapon system.

4. DISCUSSION

The performance specification for the F-X (early F-15 designation) still quoted the classic criteria of maximum speeds, a primary mission profile combining cruise radius with added dash distance and nonstandard day takeoff and landing capability. However, the design criteria stressed as equal in importance to 1-g performance, a capability to maneuver at high load factors (g's) and at subsonic, transonic and supersonic speeds at low, middle and high altitudes — literally design criteria within the predictable air combat arena. In addition, transonic buffet free maneuverability at medium load factor was identified to assure the pilot's capability to function while maneuvering.

McDonnell Aircraft Company, the St. Louis, Missouri Division of McDonnell Douglas Corporation, proposed in June 1969 to build the airplane today configured as shown in Figure 1.

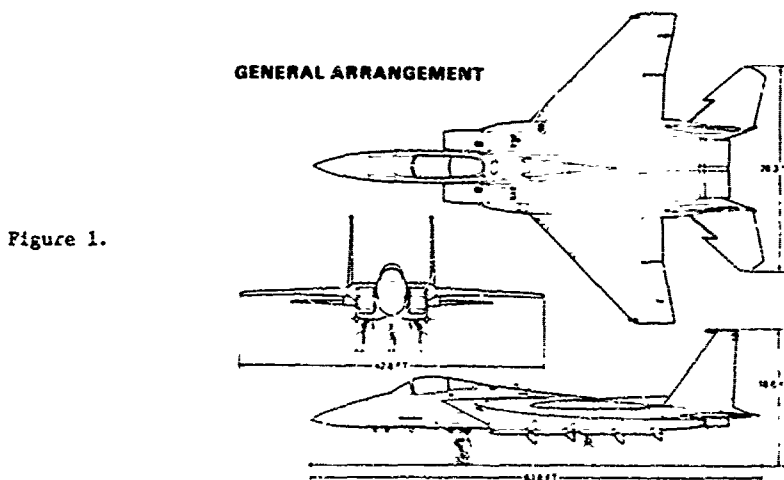


Figure 1.

The proposal configuration was the consequence of analyzing by computer aided design and evaluating by parametric techniques over 480 designs and 7500 iterations. McDonnell Aircraft Company applied these analyses to nearly 70 "hard point" design studies which included actual configuration layouts, weights estimates, systems installations, wetted and cross section areas and performance. Of these analyses to mid-1968, five specific designs were studied to verify such selections as a single crewman, fixed rather than a variable sweep wing and two fuselage mounted turbofan engines rather than a single engine or twin podded turbojet engines. The genealogy of these specific F-X designs leading to the proposed F-15 configuration appears in Figure 2.

4.1 Crew. The crew station as shown in Figure 3 is located for maximum air-to-air combat visibility; however, the concept of an air superiority design mission did not inherently specify the crew size. The analysis of this important design parameter investigated both subjective pilot surveys and a human engineering task study of the air-to-air combat mission from briefing to debriefing, including the crucial dogfight segment of the mission.

Pilot opinion and experience complemented the task analysis in determining the practicality of a single crewman. Summarized, the pilot's proficiency is dependent upon:

- An integrated weapons and fire control system
- Head-up displays
- Good visibility outside the cockpit

Figure 2.

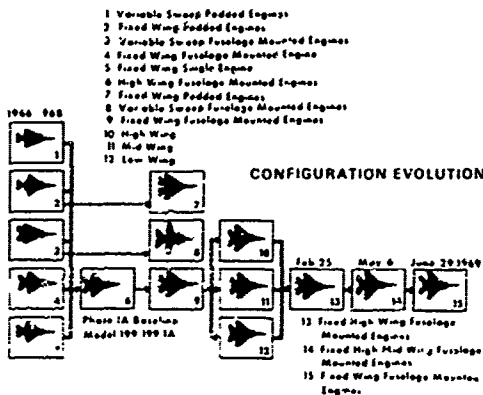
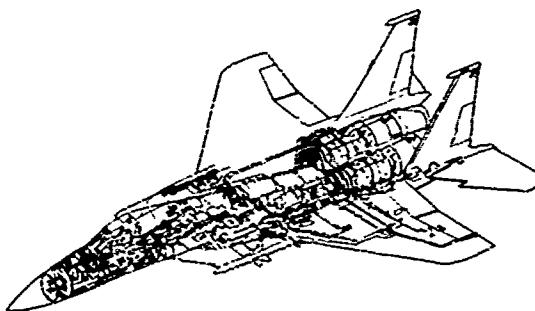


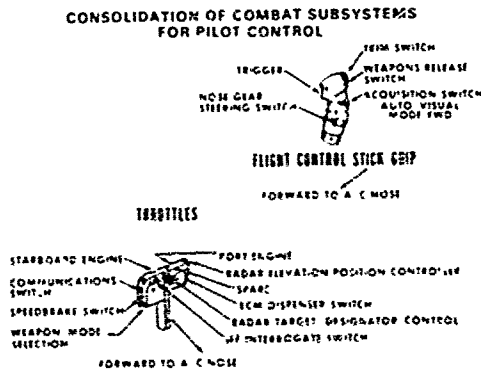
Figure 3.

F-15 INTERIOR ARRANGEMENT



In Figure 4 the diagram of the throttles and flight control stick indicates the consolidation of control functions for the air-to-air attack modes. Thus, the pilot can control not only his engines and airplane maneuvering but, also, combat related subsystems without removing his hands from primary controls. The crew station "green house" canopy was designed to optimize the pilot's visibility as can be judged in Figure 1.

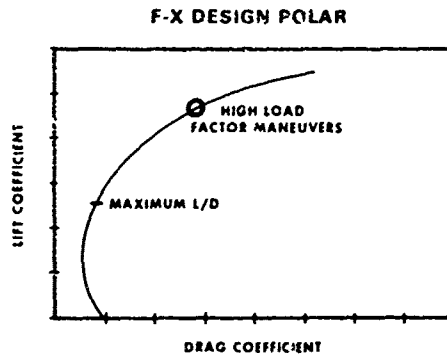
Figure 4.



"Hard point" configuration design layouts and mission analyses confirmed the capability to meet the human engineering requirements. To perform the same combat mission radius with approximately the same maneuver capability, the two-man aircraft would increase in size approximately 15 percent and in takeoff weight by nearly 5500 pounds over the single man aircraft.

4.2 Wing Selection. The concept of a fighter combining supersonic speeds and subsonic range with transonic maneuver ability under high load factors (g's) afforded the designer the rare opportunity to reinvestigate wing technology. To derive the F-X wing required the determination of planform features - area, span, taper and sweep - and airfoil section - twist, camber and thickness - as well as the location of the wing on the fuselage - high, mid-wing or low. This new look at wing design evolved from an era emphasizing 1-g flight at lift coefficients (C_L) less than required for maximum lift/drag ratio, low aspect ratios, high sweep, low thickness ratios and high wing loading. The past era of fighters performed inefficiently at high C_L 's. For the F-X, however, at middle altitudes the high transonic load factor maneuverability became the critical design point. The C_L at this condition, as shown on Figure 5, is considerably higher than the C_L for maximum lift/drag (L/D). The drag polar at this point is no longer parabolic so that the fundamental characteristics of the flow had to be investigated.

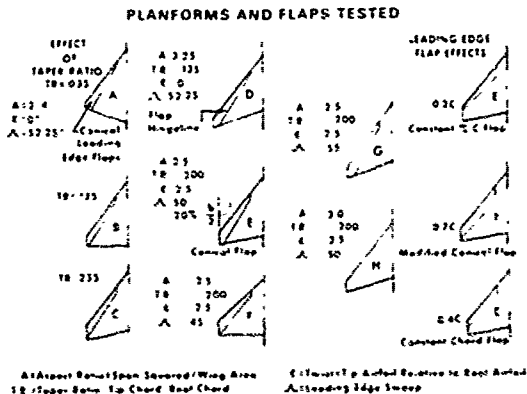
Figure 5.



Wind tunnel testing complemented theoretical work centering around Multhopp's Extended Lifting Surface Theory and the Woodward Linear Theory extended to subsonic speeds and around Whitcomb supercritical wing theory as applied to high aspect ratio (span/chord) variable sweep wings. Extensive wind tunnel testing afforded data for a trade study to compare wing parameters at the transonic high lift load factor and supersonic lift conditions with the other performance lift conditions. The lift coefficients at which drag "breaks" from the parabolic and the degree of drag increases, were found to be functions of thickness ratio, sweep and camber.

The drag-lift advantages of high camber wing section at subsonic speeds were compared with penalties at supersonic speeds. The consequent investigation of 13 planforms compared leading edge flaps and cambered wings effects due to wing taper, aspect ratios and leading and trailing edge sweep. Representative planforms and characteristics evaluated are shown on Figure 6.

Figure 6.



Most have conical leading edge flaps, the hingeline being formed by a line through the wing apex and at a constant 80 percent of local semi-span. Transonic high-lift drag was improved dramatically but high lift devices added both weight and complexity to a weight sensitive fighter. Increasing wing camber shifts the center of pressure aft, causing a shift in pitching moment. Consequently, trim drag increases. However, a combination of twist (ϵ) and camber concentrated outboard produced little change in pitching moment. These trades of twist and camber effects are shown on Figure 7 with the selected wing planform.

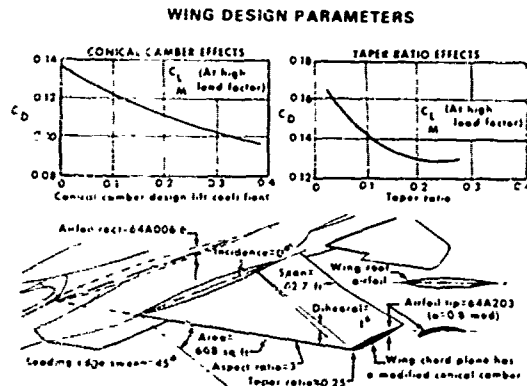


Figure 7.

The basic choice between variable sweep and fixed wing geometry was determined after the "hard point" design configuration analyses. To fly the same air superiority mission radius, the fixed wing configuration was 4700 pounds lighter, most of this in structure. The fixed wing fighter permitted about an eight percent less engine thrust at sea level. At the critical transonic high load factor condition, the fixed wing fighter could maneuver better, primarily because the variable sweep aircraft was structurally heavier. On the other hand, the higher aspect ratio and greater basic fuel load of the variable sweep fighter increased its ferry range approximately five percent over the smaller fighter. The criterion for a basic air superiority mission favored the selection of the fixed wing fighter, primarily because of its generally better maneuver capability at combat weight.

Early in the F-X design studies, the contractor's designers favored locating the wing high on the fuselage. The continuity of wing/fuselage upper surface improved aerodynamic lift relative to a design where the fuselage interrupted wing lift. This high wing location also contributed to structural efficiency by simplifying structurally redundant load paths at the wing and fuselage junction by direct transfer of wing loads to the carry-through structure. The resulting wing box over the engine ducts was lighter weight than a mid-or low-wing design and also reduced fuselage structure complexity. The high wing arrangement facilitated carrying external stores, particularly the tangential installation of air-to-air missiles on the corners of the fuselage. These considerations were judged to outweigh the design advantages of a low wing which permitted locating a simple, shorter, wide tread landing gear and taking advantage of aerodynamic ground effect during takeoff and landing.

4.3 Empennage Design

4.3.1 Horizontal Tail. Fundamentally, the precise flying qualities of the maneuvering fighter depend upon the tail surfaces to modify the wing's contributions to these qualities. The high wing plane had early been established so that positioning the horizontal tail relatively low became desirable to improve longitudinal stability at high angles of attack. Furthermore, a trade study confirmed that the optimum tail size for minimum drag is often larger than the tail size for minimum weight because of the higher induced drag created by smaller horizontals. A wind tunnel investigation of numerous horizontal tail sizes and locations was performed to establish a basis for aerodynamic effectiveness. This information was then coordinated with the basic weight and balance variations that occur with variations in horizontal tail size and location (see Figure 8). A low tail also provided higher load factor trim characteristics and aided in reducing neutral point shifts with Mach number. The design studies included consideration of exhaust gas impingement and effects of engine vibration on the tail structure. Further studies on horizontal tail configurations were conducted to improve efficiency by establishing the leading edge sweep for an optimum balance of structural and aerodynamic effectiveness. This positioned the spindle minimum hinge moments at a chordwise location near maximum section depth.

4.3.2 Vertical Tails. The selection between a single, larger vertical tail and smaller, twin vertical tails was resolved initially in favor of twin verticals plus ventrals. In a later drag reduction program, the ventrals were deleted but the verticals were increased in area to maintain satisfactory levels of directional stability at critical design points. Other factors were decisive rather than the aerodynamic considerations regarding selection of a single vertical or twin verticals of the same effective area. Aerodynamically, the end plating by the twin vertical tails provided for more effective horizontal tail control capability. The twin verticals provided a more linear variation of yawing moment with sideslip. Also, twin verticals provided high directional stability for improved dutch roll dynamic stability characteristics and provided rudder control redundancy for combat survivability. Located on engine booms, the verticals increased IR shielding of engine exhaust in side profile for better survivability.

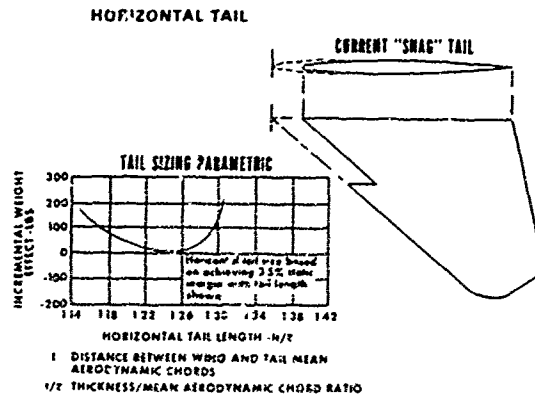


Figure 8.

Only two boom structures were required instead of a third "spine boom" for a single vertical tail. The shorter span of twin vertical tails reduced torque on the fuselage. The greater distance of the twin vertical tails from the engine center line, provided a lower noise and temperature environment and permitted lighter weight structure than did a single vertical and its supporting structure.

The vertical tail sizing was influenced, first, aerodynamically by the fuselage and, second, by positioning twin tails, one surface each on the "tail booms" that straddle the engines (Figure 1). The dominant design criteria to provide adequate directional stability occurred at high angle of attack and transonic and supersonic speeds.

4.4 Power Plant. Combined with the airframe design trade offs, concurrent engine cycle analyses, engine/inlet compatibility studies and engine nozzle trade offs created a matrix of configurations from which to select a single airplane design to achieve F-X maneuverability and performance objectives. The propulsion studies included analyses of podded and fuselage engine installation and of dual powerplants compared with large single engine installations in hard point airplane designs.

4.4.1 Engine Cycle Analysis. The initial engine studies iterated the thrust requirements to meet maneuverability and mission objectives of the F-X and thereby size the engine. The principal objective of this analysis was to determine the by-pass ratio (BPR) and thereby size the engine for the F-X air superiority fighter. For an engine sizing criterion, ranging from zero (pure turbojet engine) to 2.2 (large turbofan), the designers created a thrust parametric derived from complex BPR variation with engine airflow and pressure ratio. The ratio of sea level static thrust to take off gross weight ($T/TOGW$ = thrust loading) was then calculated as a sizing parameter to compare maneuverability at the required altitude, velocity and load factor. For this study, overall engine pressure ratio was varied from 12 to 29. At intermediate thrust (military power) the $T/TOGW$ increased with increasing BPR at the design condition. At maximum thrust (afterburner power) the design $T/TOGW$ decreased with increasing BPR at supersonic speeds; in the cruise subsonic region, however, practically no variation of $T/TOGW$ occurred with BPR. All other parameters being equal, $T/TOGW$ increased with increasing load factor. Again, the design critical condition was the transonic high lift load factor requirement. At this point, the parametric engine studies indicated a best F-X minimum size engine BPR to be within the range from 0.5 to 1.0. Even so, the trends suggested the pure turbojet appeared competitive, consistent with all parameters.

The subsequent hardpoint engine-airframe optimized for best combination of TOGW and maneuver capability indicated the aircraft TOGW for the turbojet (BPR=0) configuration was nearly 10,000 pounds heavier than the optimized turbofan configuration. The hardpoint engine-airframe design optimized for a turbofan at mid-BPR range cited above.

4.4.2 Engine Installation. To demonstrate relatively the same maneuverability and performance, the fixed wing airframe with pod-mounted engines weighed 2700 pounds more than the fuselage-mounted engine configuration. Also, the sea level static thrust of the pod-mounted engines was sized to be nearly 10 percent greater. Single engine out effects on directional control were greater considering podded engine configurations. After this hardpoint design optimization study in 1968, twin turbofan engines with a BPR somewhat less than unity and buried side-by-side in the aft fuselage, became the selected conceptual propulsion unit.

The hardpoint design study of single high thrust engine/airframe configuration revealed that two lower thrust engines buried in the airframe design were more practical for other than reasons survivability. The single engine fighter was optimized to accommodate an engine nearly 95 percent larger than one engine in the optimized twin-engine fighter. The twin-engine TOGW and fuel load were more than the single engine fighter design by approximately two percent, yet both aircraft had the same range capability. However, the twin engine fighter could maneuver somewhat better. The hardpoint design studies also indicated that the critical excess thrust to weight parameter (total thrust less airplane drag to weight ratio) of the single engine fighter optimized at a lower maneuver capability relative to the twin engine design.

Figure 9 presents the Pratt & Whitney Aircraft F100-PW-100 engine and characteristics.

F100-PW-100



Figure 9.

THRUST CLASS	20 000 LB
THRUST TO WEIGHT RATIO CLASS	9.1
BYPASS RATIO	0.7
OVERALL PRESSURE RATIO	23
AIRFLOW	224 LB/SEC

4.5 Inlet/Engine Compatibility. To achieve inlet/engine compatibility, the airflow must be supplied to the engine compressor face with sufficiently low distortion to preclude engine stalls everywhere within the airplane operational envelope. Inlet trade off studies were conducted for both fuselage mounted and podded engine configurations. These studies were concurrent and integrated with the engine studies already mentioned. Side-by-side engine inlets were studied, but this arrangement, located either above or below the fuselage, is subject to undesirable flow interactions between inlets at supersonic speeds, regardless of the cause of single inlet instability. Pod-mounted inlet designs permitted a desirably straight, relatively short subsonic diffuser but, as noted in the engine trades, the airplane drag, total weight and engine-out performance of the airplane militated against this configuration. Consequently, the fuselage mounted inlets were extensively studied; some of the configurations are shown on Figure 10.

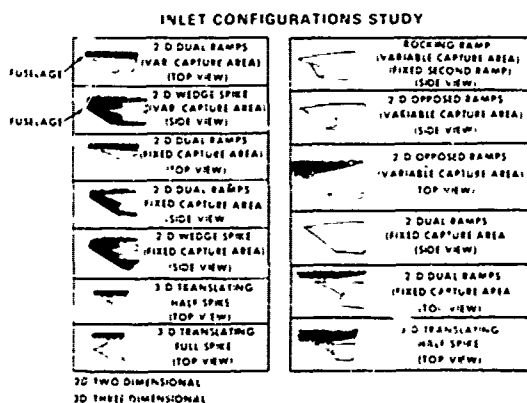


Figure 10.

All inlets were evaluated with respect to inlet drag, weight, recovery and distortion characteristics at high supersonic speeds and at angle of attack. External, rather than mixed, compression flow was selected for all inlets because the added complexities of fabricating and controlling a mixed compression inlet outweigh any achievable performance in the F-X air-to-air design maneuverability speed regime. Asymmetric inlets were eliminated from later consideration because of high distortion at high angle of attack, and high angle of attack maneuvering is an F-X inherent requirement. Consequently, various configurations of 2-D (two dimensional) ramp inlet designs remained as the most desired candidates. The F-15 inlet is a two-dimensional, horizontal ramp design with a variable capture area.

Figure 11 depicts the present inlet variable geometry and the associated inlet control system parameters.

Figure 11.

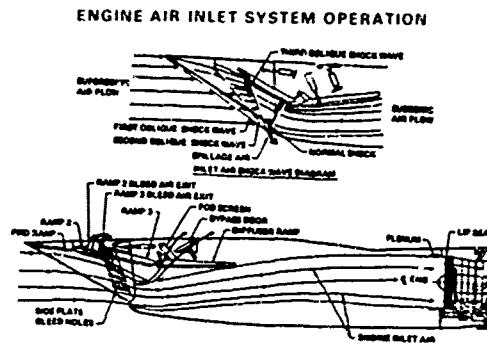
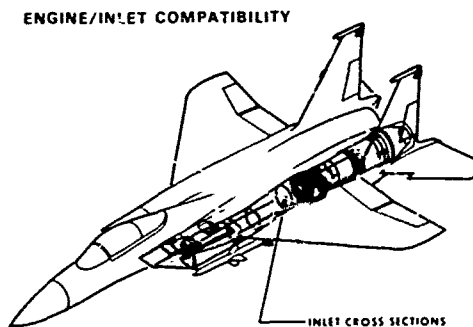


Figure 12 relates the propulsion system with the complete airframe.

Figure 12.

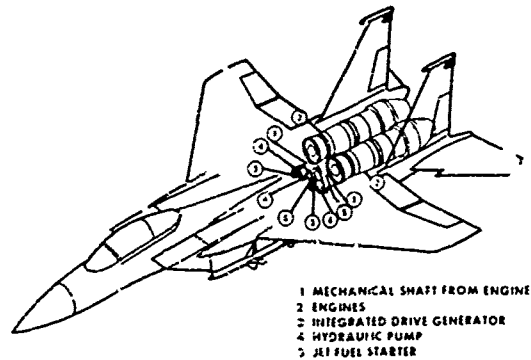


The most unique feature of the inlet is its variable capture area. Its variation is accomplished by rotating the entire forward ramp system as a unit about a point near the lower cowl lip. The included angle of the inlet leading edge is fixed so that changing the capture area also changes the first ramp deflection angle. The capture area is varied as a function of Mach number, angle of attack and free stream total temperature. The third ramp is linked to the diffuser ramp, and is scheduled as a function of Mach number. To enhance matching at hot day or low altitude conditions where engine corrected airflow diminishes, a temperature bias increases the third ramp angle. A similar amount of supersonic flow turning, before the airflow passes the bottom lip, is maintained under such conditions by closing down the capture area and thereby reducing the effective first ramp deflection angle. By virtue of this temperature biasing, closing down the capture area at supersonic speeds below middle altitudes also relieves structural loads. For purposes of control system mechanization, the third ramp is defined relative to an inlet reference line which moves with the capture area. Thus, the third ramp is really scheduled relative to the first ramp. The schedules are defined so as to produce the desired total deflection of the flow relative to free stream. The by-pass system is an aerodynamically closed loop such that the door is positioned to provide a desired throat Mach number. This desired throat Mach number is, in turn, a function of free stream Mach number.

4.6 Secondary Power Subsystem. For a discussion of the F-15, the Secondary Power Subsystem must be defined as two-engine power takeoff shafts, two airframe mounted accessory drives (AMAD's), one central gear box and one jet fuel starter (JFS). The schematic in Figure 13 is a pictorial definition of the subsystem. The early trade studies investigated the location for mounting accessories, whether to be engine mounted or to be remotely mounted. Remotely mounting the accessories reduced airplane cross section because the engine cross section was reduced by approximately one square foot without accessories attached. Furthermore, a complete engine change could be expedited because engine-airframe connections were eventually reduced to a minimum of 11 connections for each engine—one fuel, one environmental control, two electrical connectors, one power takeoff shaft, one throttle, one nozzle fairing strap, and four engine mounts.

F-15 SECONDARY POWER SUBSYSTEM

Figure 13.



The trade offs also studied the function of engine starting to some detail, comparing remote mounted accessories relative to the engine to assess the arrangement relative to engine starting systems. Self sufficiency versus ground cart assist was included. Further, the arrangements were considered to provide energy sources for ground maintenance. These studies favored the remotely mounted Shaft Power Unit (auxiliary power unit - APU), considering its weight, reliability and maintainability characteristics relative to those of the jet fuel starter (JFS), the cartridge starter, the hydraulic starter plus gas turbine power unit, or the gas turbine compressor plus air turbine starter. Later, as the result of a weight and cost saving study, the APU was replaced by the lower weight, lower cost JFS. The JFS has been designed to transmit power to either engine in any sequence, through the left or right AMAD gear box for starting. For current F-15's, the JFS mounted on the central gear box is configured for 30 minutes continuous duty for self-sufficient ammunition loading.

4.7 Testing.

4.7.1 Wind Tunnel. In the process of refining their F-15 design, McAir had accumulated 12,772 wind tunnel occupancy hours (WTOH) before signing the contract. This test time was three times the McAir tunnel time up to first flight of the F-4. Furthermore, since contract, McAir has occupied National Aeronautics and Space Administration (NASA) facilities at Ames and Langley Research Centers and those at the Air Force Arnold Engineering Development Center (AEDC) tunnels as well as their own facilities at the St Louis, Missouri plant and at Douglas Astrophysics Laboratory on the West Coast. In the four years since contract award, McAir has accumulated more than the pre-contract tunnel occupancy hours. The detail to which the testing has investigated the F-15 characteristics is summarized by Figure 14a and 14b.

Figure 14a.

F 15 WIND TUNNEL TEST PROGRAM OCCUPANCY HOURS	
AERODYNAMIC TESTS	
4 7% MODEL IN THREE FACILITIES	3493 HOURS
7 5% MODEL IN ONE FACILITY	198 HOURS
13% MODEL IN ONE FACILITY	190 HOURS
STABILITY & DYNAMICS TESTS	
4 7% MODEL IN FOUR FACILITIES	816 HOURS
7 5% MODEL IN ONE FACILITY	388 HOURS
10% MODEL IN ONE FACILITY	120 HOURS
1 20 SCALE MODEL IN ONE FACILITY	80 HOURS
75% MODEL IN ONE FACILITY	72 HOURS
13% MODEL IN TWO FACILITIES	388 HOURS
13% HELICOPTER DROP MODEL	140 DROPS
STORE SEPARATION TESTS	
4 7% MODEL IN TWO FACILITIES	404 HOURS

F 15 WIND TUNNEL TEST PROGRAM OCCUPANCY HOURS (Cont d)	
PROPULSION TESTS	
7.5" INLET FLOW FIELD MODEL AT ONE FACILITY	145 HOURS
1/6 SCALE INLET MODEL AT ONE FACILITY	813 HOURS
4.7% JET EFFECTS MODEL AT ONE FACILITY	405 HOURS
FULL SCALE INLET TEST AT ONE FACILITY	861 HOURS
1/6 SCALE BLOCKAGE AT ONE FACILITY	82 HOURS
LOADS TESTS	
4.7% MODEL IN ONE FACILITY	267 HOURS
7.5% MODEL IN ONE FACILITY	497 HOURS
FLUTTER TESTS	
10% TRANSONIC EMPENNAGE MODEL AT ONE FACILITY	39 HOURS
12% TRANSONIC FULL AIRPLANE MODEL AT ONE FACILITY	1204 HOURS
12% EMPENNAGE MODEL AT ONE FACILITY	864 HOURS
12% WING MODEL AT ONE FACILITY	711 HOURS
12% COMPLETE AIRPLANE MODEL AT ONE FACILITY	594 HOURS
12% COMPLETE AIRPLANE & COMPONENT MODEL AT ONE FACILITY	180 HOURS
MISCELLANEOUS	
25 MM SEC 4.7% MODEL AT ONE FACILITY	47 HOURS
SPEEDBRAKE BUFFET 12% MODEL AT ONE FACILITY	91 HOURS
TOTAL OCCUPANCY	12645 HOURS

Figure 14b.

Of these tests, two areas are unique, the Propulsion Test and the Stability and Control Tests. For inlet/engine compatibility analyses before contract, McAir devoted over 5000 WTOH to inlet and nozzle design, engine spacing and airplane configurations with engines podded and buried in the fuselage. Subsequent to contract award, McAir has devoted nearly 2000 WTOH more to investigating the characteristics of the final inlet/engine design (Figure 14b). The scope of the detailed testing as included:

- Optimize boundary layer bleed system
- Optimize schedule of ramps
- Verify adequacy of inlet offset distance from the fuselage
- Define effects of control system errors
- Evaluate lower cowl lip shapes
- Evaluate modifications to sideplate geometry
- Investigate changes to throat slot geometry
- Evaluate alternate third ramp configurations
- Investigate alternate by-pass control sensor locations

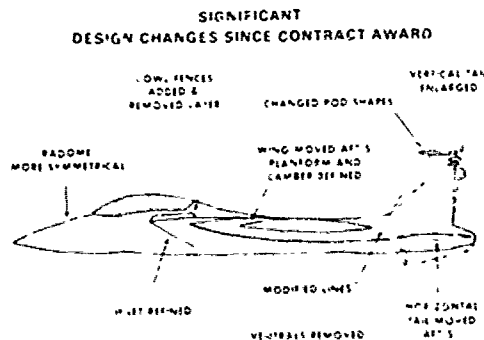
Over 600 WTOH have been devoted to full-scale inlet-engine tests at AEDC facilities. As a result of this meticulous investigation of this technology and hardware, flight tests to date have produced no evidence of inlet-induced engine stall.

The second area of model tunnel testing unique to the F-15 is the Spin Program to investigate the fighter's high angle of attack and spin characteristics before first flight in July 1972. These investigations of pre- and post-stall characteristics at AEDC and at NASA Langley Research Center include effects of center of gravity, control surface positions as well as incremental changes to basic fuselage nose shapes, wing and empennage relationships and inputs from the control augmentation system.

To complement the wind tunnel tests, a helicopter F-15 model drop program has been coordinated with NASA Langley Research Center. Since the first drop in March, 1972, NASA Langley Research Center has completed over 40 drops of this 13 percent scale model.

The lessons learned in the wind tunnel were continually applied to the maturing F-15 design before first flight, as shown on Figure 15.

Figure 15.



These changes, for example, reduced drag, improved balance characteristics and wing structural efficiency, inlet airflow and consequent engine stability margin and maintained airplane lateral-directional stability, while increasing the flutter margin.

The wind tunnel program is nearly completed now. Any changes in WTOH will essentially be in areas where flight testing has indicated need for further improvement.

4.7.2 Flight Test. The flight test program designated specific aircraft to investigate technical areas. Fighter F-1 was assigned to flutter tests, for example. The "snag" horizontal tail (see Figure 8) is a specific example of design improvement from applying lessons learned from the wind tunnel to flight testing. As a design tool, the wind tunnel indicated the horizontal tail to be more flutter prone than did earlier analyses and, after pressure distribution tests, indicated that a tail with a "snag" leading edge met flutter standards. At this time, Fighter F-1 was nearing first flight; its "clean" horizontal tail was ballasted to meet flutter requirements during early handling qualities testing. Subsequently, flutter tests with the newly installed "snag tail" demonstrated this tail configuration change to be quite satisfactory for installation on all aircraft.

The F-15 is well into the flight test program and is undergoing the comprehensive tests analyses common to all systems.

**System Analysis for a Battle-Field Air Superiority
Fighter Project with Respect to Minimum Cost**

by

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Summary

The battle-field air superiority fighter is an air defense aircraft with the function to gain and to hold air superiority over the combat area for limited time and limited operational area.

For given budget the fleet effectiveness can be optimized by reducing the price of the aircraft thus allowing for a higher number of aircraft. Mainly a low cost design can be achieved by minimizing the take-off weight. Design philosophy is to adjust range, payload and equipment to absolute necessary requirements thus arriving to a "simple" design, but not to compromise air combat capability, i. e. maneuverability.

For the evaluation of different solutions a method is discussed which shows the role of aircraft characteristics with respect to air combat and which can be used in the preliminary design phases. In the present paper, the influence of mission and design parameters, i. e. range, combat time, wing loading and thrust/weight ratio on take off weight, system cost and air combat effectiveness for the battle-field air superiority fighter is shown. Different versions of a battle-field air superiority fighter are presented and comparative results shown.

1. Introduction

The studies for a battle field fighter project start from a consideration of the threat situation in Central Europe which results from the strong numerical superiority of the Warsaw Pact states in tanks and aircraft. In order to deter a credible defense potential has to be set against this threat. Concerning the air forces the defense potential must allow apart from the defense of tank attacks especially an effective air defense. For this, ground-based defense systems which cover the whole range of altitudes have to be considered. But these defense systems have to be completed by aircraft (see figure 1). The aircraft serve to secure the airspace at the flanks and at weak spots, to fight against penetrated aircraft and to undertake the air defense, where the ground-based air defense has collapsed.

With regard to the situation of the Federal Republic of Germany (FRG), a modern battle-field air superiority fighter could be of interest, which is capable to gain and to hold air superiority over the combat area for limited time and limited operational area. This task includes also the interception of intruding fighters and combat aircraft. The experience from different conflicts of the past has shown that the fight, that is the defense against enemy aircraft, leads frequently to the air combat in the dogfight mode; therefore, pronounced air combat capability is required.

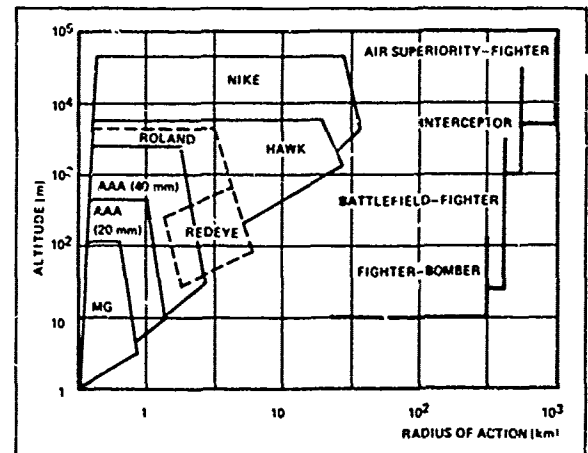


Figure 1: Air Defense Systems

The following missions are considered for the battle-field air superiority fighter:

- | | | |
|--|-----------------------|---|
| <ul style="list-style-type: none"> - Air combat and air patrol missions - Interception, but extreme altitude excluded - Escort missions for own CAS-aircraft - Limited capability for close air support - Battle-field reconnaissance | }
}
}
}
} | primary missions

secondary roles |
|--|-----------------------|---|

2. Analysis of Requirements

In analysing requirements of a battle-field air superiority fighter, it is not intended to discuss design criteria of fighter aircraft, since this was done already in the past by several authors [1, 2]. This paper concentrates mainly on cost considerations during the formulation of design requirements.

2.1 Design Philosophy

The fundamental level of the costs for a weapon system is already determined by the design requirements. Therefore it is necessary to consider the viewpoints of cost already during the preparation of such requirements. This is done by obtaining a survey about the influence of the different requirements on the cost.

The design philosophy in the case of the battle-field air superiority fighter is not to compromise air combat capability, because as air combat simulation shows - already a relative slight inferiority leads to the result that the fight against the opponent will be lost in most cases. A reduction in cost will be reached especially by minimizing the take-off weight. Therefore, radius of action, payload and equipment are confined to absolute necessary values and combat time is chosen in order to maximize fleet effectiveness. Furthermore, design and construction are kept as simple as possible and complexity and use of not proven technology is tried to be avoided.

For the battle-field air superiority fighter a single seater is proposed. The aircraft should be fitted with two engines in spite of the higher aircraft unit cost, for safety reasons especially in peacetime operation, where flights over the densely populated European country must be considered.

With limited budget it is not possible to procure for each task a special optimum aircraft or weapon system resp. In the contrary, it is necessary to assign one weapon system to several tasks. However, one has to take care that only such tasks are combined which can be accomplished by the weapon system without considerable additional costs. As figure 2 shows, the battle-field air superiority fighter could cover the principal and secondary missions as mentioned.

MISSION	REQUIREMENTS											
	RANGE	PAYLOAD	MISSION TIME	LOW LEVEL PENETRATION CAP	MAX. SPEED	HIGH ALTITUDE SURVIVABILITY	MANEUVERABILITY	ACCELERATION	ALL WEATHER CAP	NAV. ATT. CAP		
BATTLEFIELD AIR COMBAT	1) WITH EXTERNAL FUEL TANKS 2) WITH ADDITIONAL LOW COST AVIONICS
AIR PATROL	
ESCORT OF CAS AIRCRAFT	PRIMARY MISSIONS
AIR SUPERIORITY AT EXTREMELY HIGH ALTITUDES	
INTERCEPTION LOW MEDIUM ALTITUDE	SYMBOLS + SUFFICIENT - NOT SUFFICIENT
INTERCEPTION HIGH ALTITUDE	
CLOSE AIR SUPPORT	.	(+)	(-2)	SECONDARY MISSIONS
BATTLEFIELD INTERDICTION	.	(+)	(-2)	
AIR INTERDICTION	-	(+)	-	-	
DEEP INTERDICTION	-	(+)	-	-	
STRIKE	-	(+)	-	-	
BATTLEFIELD RECCE	
TACTICAL RECCE	

Figure 2: Multi-Mission Capability

2.2 Definition of Cost

The considerations with regard to cost cover the so-called system cost as shown in figure 3. The life time over 15 years is considered as peacetime operation. The system unit cost are the research and development cost and the procurement cost per aircraft.

The estimation of cost in the early design phases is not without problems concerning accuracy. However, the main purpose of the cost considerations is to achieve a reasonable level of cost by analysing the influence of requirements on cost. Therefore not the absolute cost number seems to be of most importance but the relative number. For this purpose, the statistical methods used seem to be sufficient. Moreover, it can be assured that the values given are relatively close to absolute cost numbers on the basis of 1973.

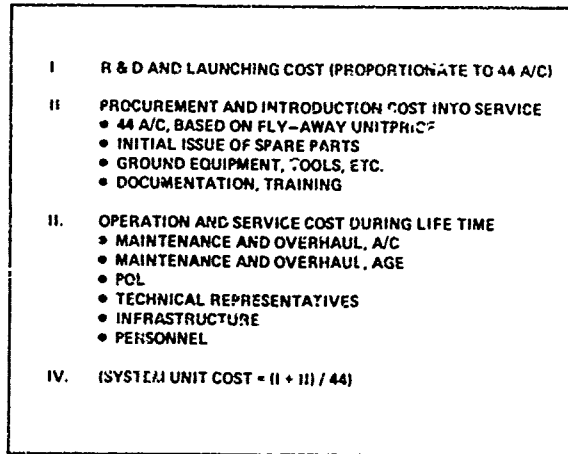


Figure 3: System Cost Definition

Fig. 4 shows the quantity of the system cost and the flyaway unit cost to be expected. The operation of a squadron of 44 battle-field fighters is expected to cost appr. 50 Mio DM per year. An amount of appr. 10.5 Mio DM for the flyaway unit price should be achievable.

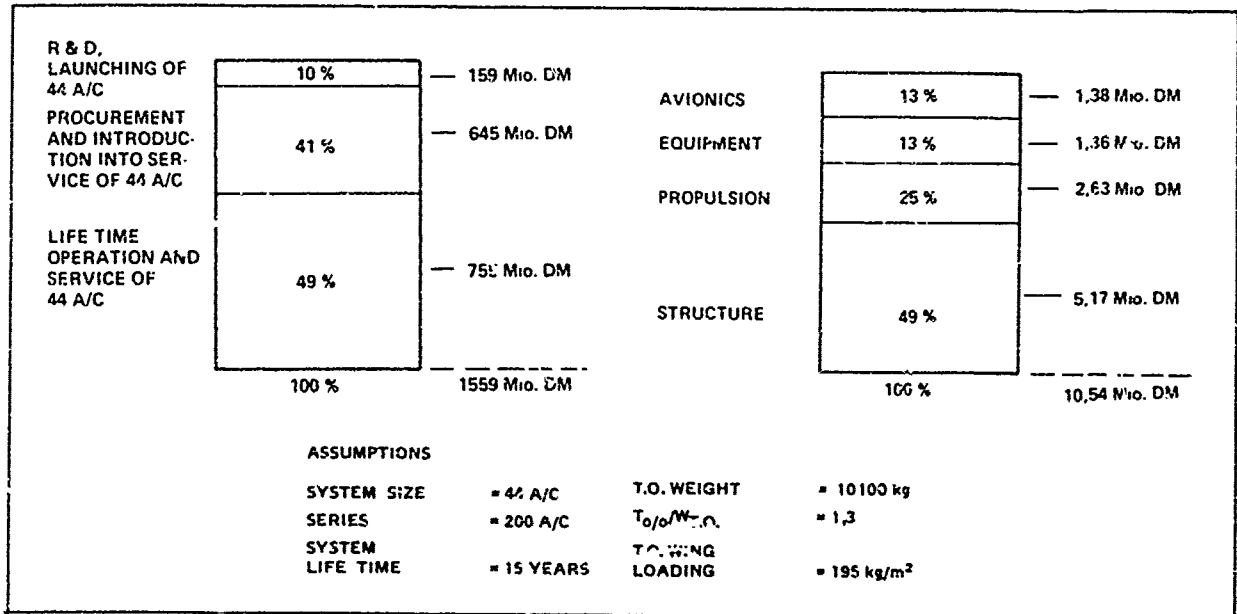


Figure 4: System Cost and Fly-away Unit Cost

The requirements with regard to the avionics can influence the cost in a very decisive way. Four alternate avionics packages with increasing performance were defined and studied with regard to the cost effects. It is necessary to consider not only the net price of the avionics but also the implications on weight and volume. As figure 5 shows, the cost increases very sensitive with the increase of effectiveness. The avionics package, which would meet all expectations, costs more than twice the proposed solution but has only an estimated improvement in effectiveness of appr. 40 %.

The evaluation of the effectiveness of avionics is difficult because of the problem to quantify clearly the influence of avionics on mission success and because of the variety of tasks to be performed by the avionics. Here a method was used which relies on engineering judgement and is similar to the method which is described later in chapter 3.

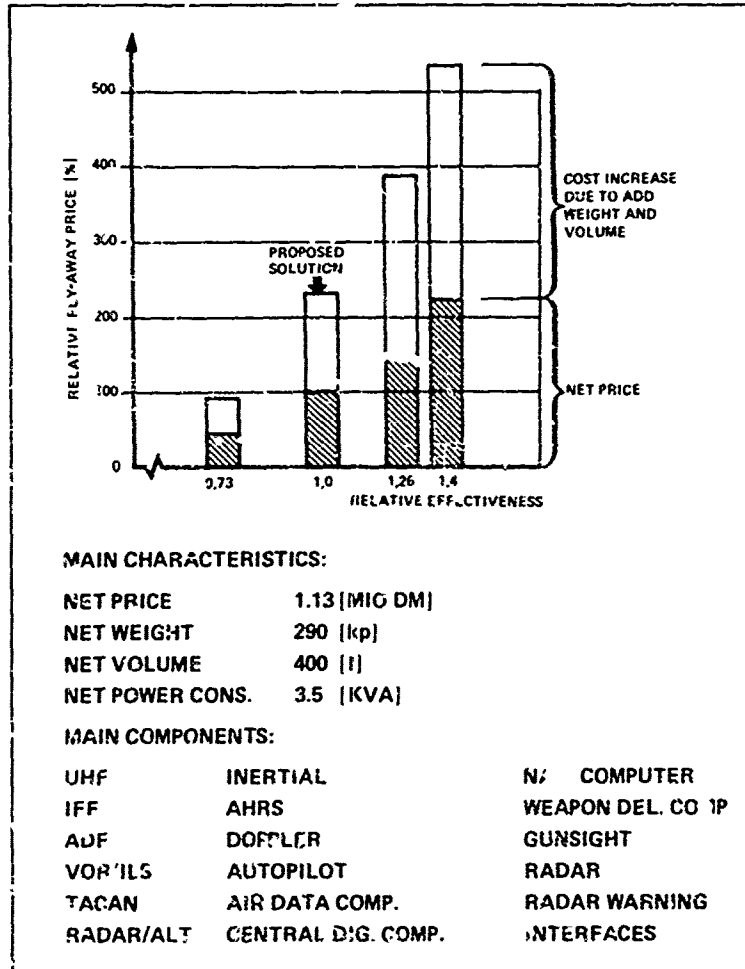


Figure 5: Cost of Avionics

In figure 6 the system unit cost is shown as function of take-off gross weight. The system unit cost increases by one Million DM, if the gross weight increases by one metric ton. The weight of 10 metric tons results in system unit cost of appr. 18 Million DM.

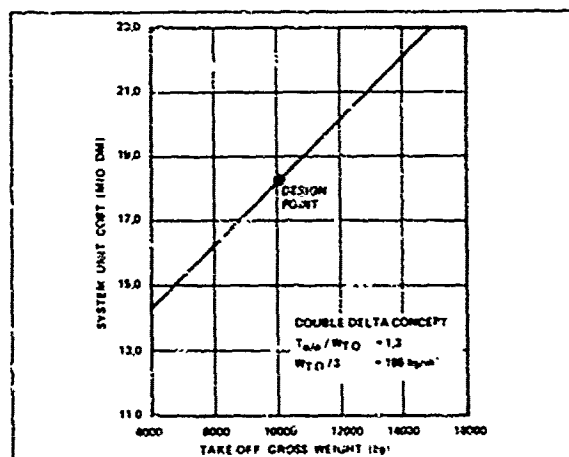


Figure 6 System Unit Cost vs. Take-off Weight

2.3 Discussion of Requirements

The following considerations are based on results for a more detailed conventional design and on parametric studies derived from this design. The results are valid for a mission with a radius of action of 150 nm with outbound and inbound flight at 30000 ft altitude. The combat time is 2.5 min. in 15000 ft altitude at 0.8 Mach number. 10 min. loiter time is added in advance to the combat phase. Fuel reserve is 5 % of total fuel consumed.

Among excellent flying qualities and a perfect cockpit visibility, the combat performance that is sustained turn rate and 1 g-specific excess power (SEP) are the decisive parameters for air combat and interception [1, 2]. For this reason, these parameters could not be compromised and cost reasons cannot determine the choice of turn rate and specific power, but one has to fulfill the requirement to achieve comparable or even better performance values than the potential future opponent aircraft.

Figure 7 shows estimated SEP and turn rate values of some realized aircraft and projected aircraft. The altitude of 15000 ft and a Mach number of 0.8 are considered as representative with regard to the flight regime of air combat. The minimum requirements for the battle field fighter are a SEP-value of 150 m/s and a turn rate of 10 °/s. But some recent fighter projects shows a jump to values of SEP > 250 m/s and turn rates of appr. 15 °/s. It is reasonable to assume that an opponent will develop comparable aircraft.

These high performance values are obtained with high thrust/weight ratios (T/W) and low wing loadings (W/S), as figure 8 illustrates. The performance values mentioned correspond to thrust/weight ratios of 1.2 and wing loadings of 300 kg/m² in the case of a conventional design.

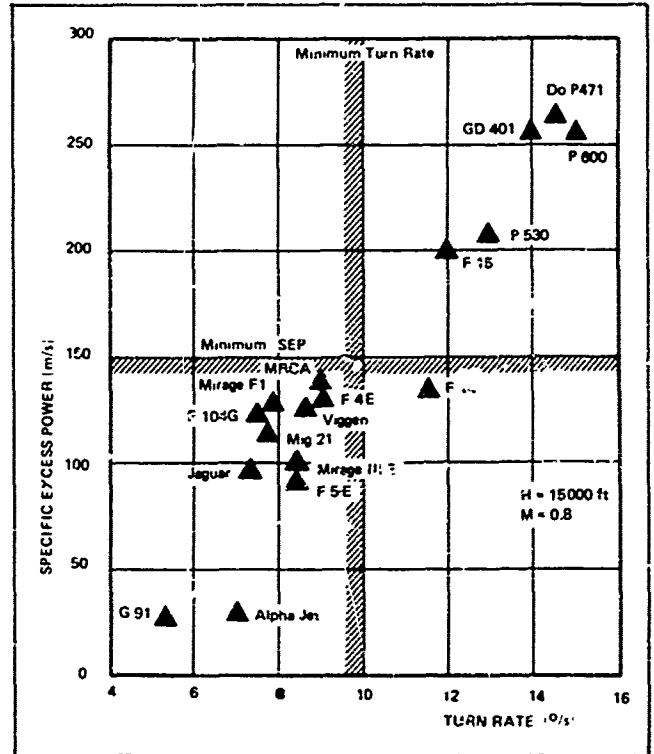


Figure 7: Minimum Combat Performance

What are the limits of SEP and turn rate? The human tolerance concerning g-loads can be assumed in the range between $n = 7 - 7.5$. This tolerance restricts the turn rate to appr. 16 °/s. Another limitation can be found by considering the "buffering limit" with "moderate buffering" roughly at a lift coefficient of $C_L = 0.7$. This limit yields SEP values of appr. 290 m/s. Figure 8 shows further, that the costs rise continuously with the increase in SEP and turn rate. The influence of SEP and turn rate on the system unit cost is shown more detailed in figure 9. Doubling the specific excess power from 150 m/s to 300 m/s increases the system unit cost from 14 Million DM to 19 Million DM. The additional improvement of the turn rate from 13 °/s to 15 °/s costs a further 1 Million DM.

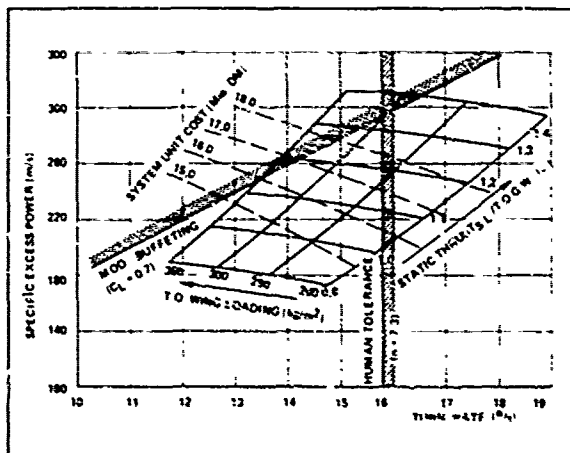


Figure 8: SEP and Turnrate

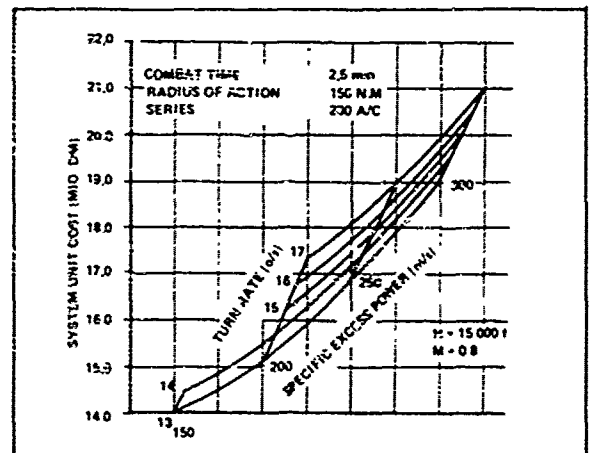


Figure 9: System Unit Cost and Combat performance

The thrust/weight ratios and wing loadings discussed here allow to fulfill easily requirements concerning short take-off distances and maximum speed (see figure 10). For the escort of own CAS aircraft and for interception and pursuit of enemy aircraft near ground, a sufficient speed capability is necessary. Therefore a maximum Mach number of 1.2 at sea level is chosen. The service ceilings is not considered as a determining requirement because ceilings of 65000 ft are easily achieved.

In order to achieve high turn rates, the buffet limitations have to be considered as mentioned already. As figure 11 shows, high turn rates can be achieved only with relative low wing loadings of appr. 300 kg/m² assuming the C_L-range for "moderate buffeting" as shown [3]. These wing loadings would be also valid, if advanced technology allowed for much higher C_L-values for buffeting and for higher g-tolerances due to advanced cockpit design. Considering the altitude and speed range of air combats, figure 11 shows also that for 30000 ft altitude and 0.9 Mach even smaller wing loadings would be of advantage with respect to buffeting avoidance.

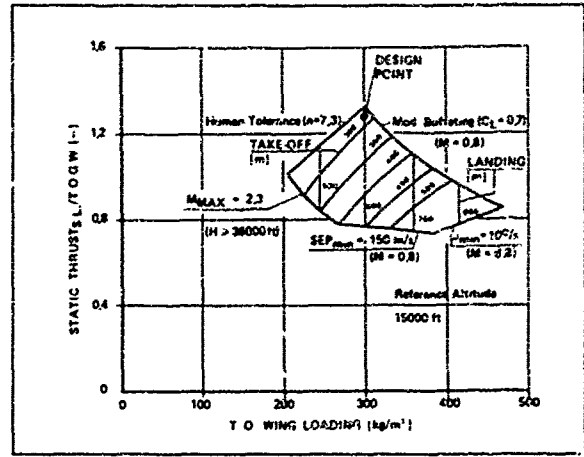


Figure 10: Thrust/Weight-ratio and Wingloading

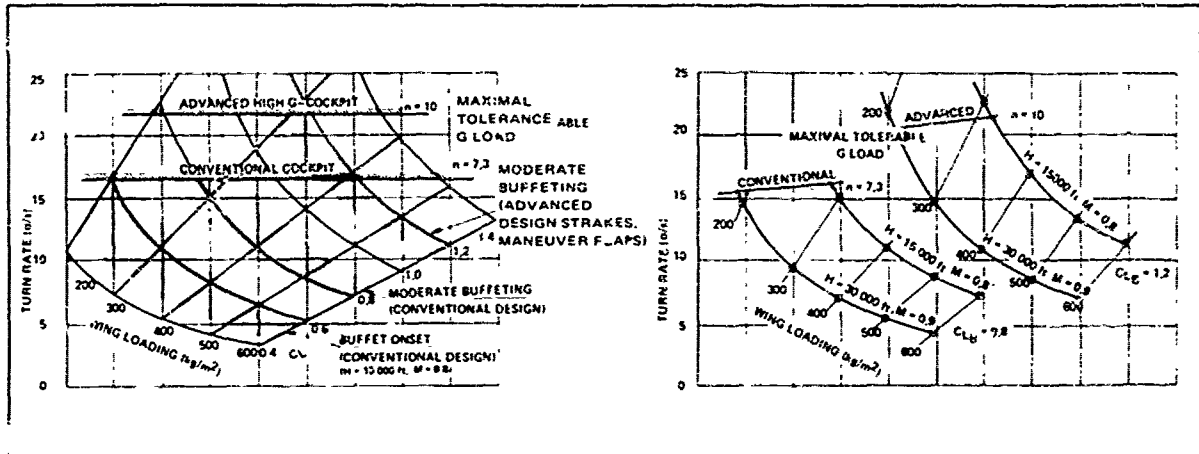


Figure 11: Turnrate and Buffeting Limitations

Combat time and radius of action influence weight and cost in a similar significant way as SEP and turn rate (see figure 12). An increase in the radius of action from 150 nm to 200 nm raises the cost by appr. 1 Mill. DM, doubling the combat time from 2.5 min. to 5 min. raises the cost by appr. 1.5 Mill. DM.

Combat time and radius of action, however, do not play such a decisive role concerning air combat performance. Therefore, these parameters can be chosen considering cost.

In case of an air defense fighter, a radius of action of 150 nm is sufficient, if the operation in the FRG is considered. Concerning the combat time, a low value is aimed at. In order to determine the length of combat time, a maximization of the fleet effectiveness for given budget can be attempted.

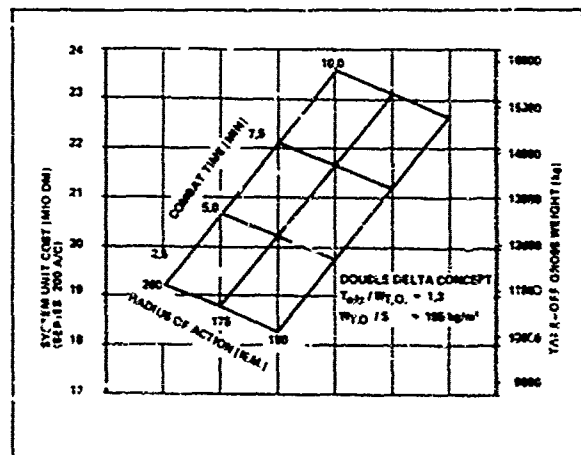


Figure 12: System Unit Cost vs Combat Time and Radius of Action

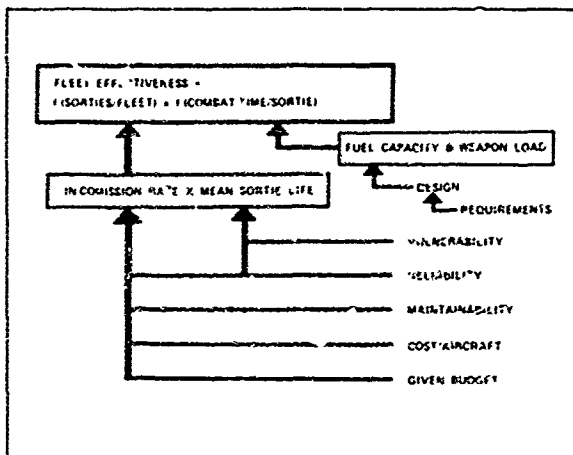


Figure 13: Fleet-Effectiveness for Battlefield-Air Superiority Fighter

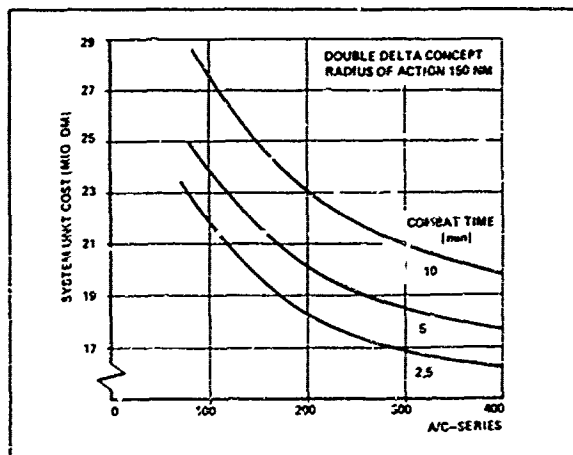


Figure 14: System Unit Cost vs Aircraft Series

The fleet effectiveness depends in a certain way on the number of possible sorties of the fleet and on the combat time per sortie. A take-off weight and combat time respectively are sought, where for a given budget and constant SEP and turn rate the fleet effectiveness reaches a maximum value. When evaluating the number of sorties per fleet, the parameters listed on the right in figure 13 have to be considered. As figure 14 shows, the system unit cost depends strongly on the aircraft production number. The number of aircraft which can be procured with a given budget is proportional to the reciprocal value of the unit price.

Increase in combat time raises the take-off weight (see figure 12). The corresponding increase of the dimensions of the aircraft leads to a higher vulnerable area, as figure 15 shows. The figure shows also the reduction in the mean sortie life as effected by the increase of the vulnerable area.

In figure 16, the influence of combat time on the maintenance index, the maintenance cost and the availability is shown.

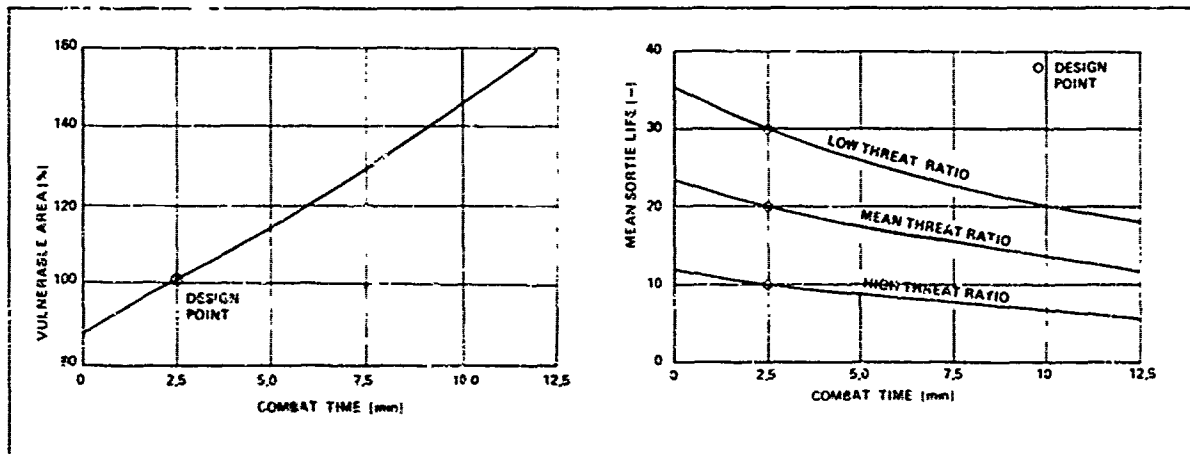


Figure 15: Vulnerable Area and Mean Sortie Life vs Combat Time

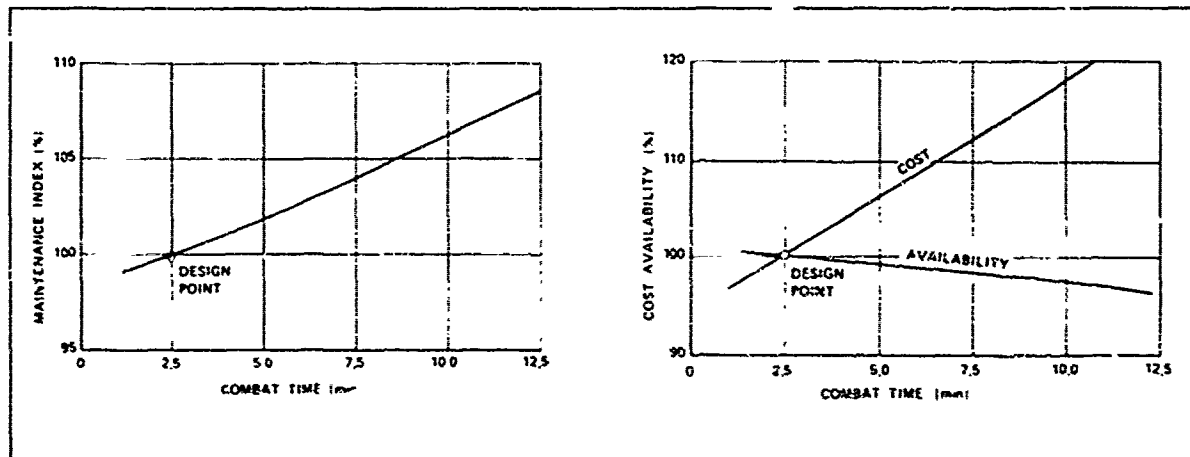


Figure 16: Maintenance Index, Cost and Availability vs Combat Time

The collection of all mentioned influences on the fleet effectiveness leads to the results shown in figure 17. As combat efficiency, the cumulative distribution of the duration of air combat, as it is known for example from air combat simulation could be taken. It is assumed that 60% of all air combats do not last longer than 2.5 min. and 95% of all air combats do not last longer than 4.5 min. The number of sorties per fleet as function of combat time depends especially on the budget given and on the threat level.

The essential statement of this diagram is that, for small budget and/or high threat, smaller aircraft with limited combat time are more favorable, whereas for higher budget and/or low threat, larger aircraft seem to be of advantage.

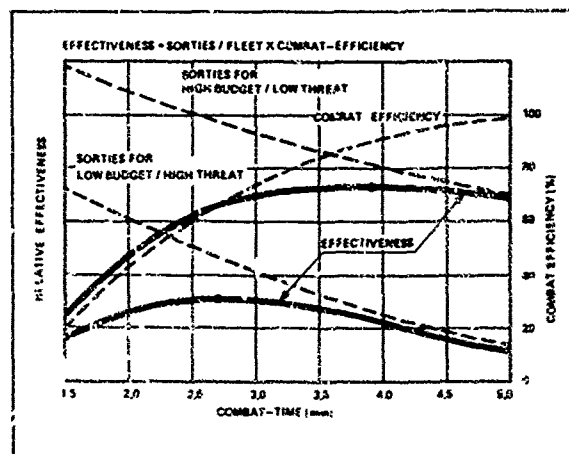


Figure 17. Fleet Effectiveness for a given Budget

For the small radius of action discussed here and for the assumed predominance of missions with air combat, a design with internal fuel only is less expensive that means that the design mission is flown without the use of external fuel tanks. Figure 18 shows the effect on cost, if internal fuel for a given mission is exchanged against external fuel. While the weight and the cost of the clean aircraft decrease with the percentage of external fuel, the increasing cost of the external tanks and the additional installation has to be added.

Naturally, by the use of external fuel tanks, the mission flexibility is increased. The outbound flight and the loiter time during air patrol can be flown with external fuel. The fuel pods would be dropped, only if an engagement with the enemy occurs, thus the air combat is started with full internal fuel tanks. The use of external fuel tanks allows increase in combat time, loiter time and radius of action.

In the case of the battle-field air superiority fighter, the use of external fuel tanks would result in a cost increase of appr. 0.5% taking the design mission as a basis

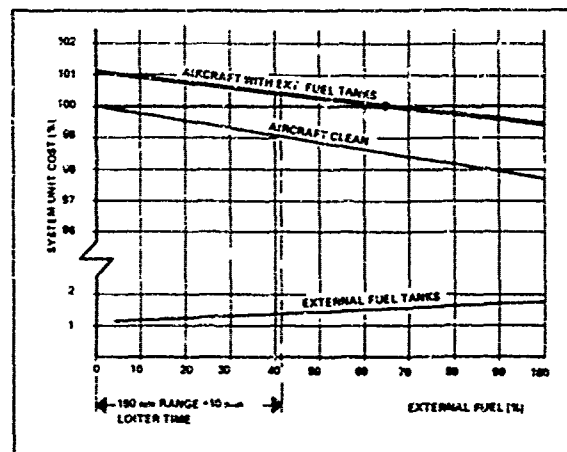


Figure 18. Internal/External Fuel Optimization

The analysis of requirements results in the set up of main requirements, as shown in figure 19. In order to minimize the take-off weight and cost, and corresponding to the design combat time of 2.5 min., two air-to-air missiles and one machine gun are chosen as normal design weapon load.

DESIGN		MISSION PERFORMANCE (WITH INTERNAL FUEL ONLY)	
• ONE SEAT		• RADIUS OF ACTION AT 30 000 FT	150 NM
• TWO ENGINES		• LOITER TIME AT 15 000 FT	10 MIN
• EXCELLENT VISIBILITY AND FLYING QUALITIES		• COMBAT TIME AT 15 000 FT	2.5 MIN
COMBAT PERFORMANCE		TAKE-OFF & LANDING PERFORMANCE	
• SPECIFIC EXCESS POWER (M = 0.8, H = 15 000 FT)	260 M ²	• T.O. - DISTANCE (ISA, S.L.)	400 M
• TURN-RATE (M = 0.8, H = 15 000 FT)	15 °/S	• LANDING DISTANCE (ISA, S.L.)	500 M
• MAX MACH-NUMBER AT 36 000 FT	2.3	WEAPON LOAD	
• COMBAT CEILING	56 000 FT	• TWO AIR TO AIR MISSILES	
		• ONE INBOARD MOUNTED CANNON	

Figure 19. Set up of Main Requirements

3. Evaluation Model

For the design work in the early design phases especially for concept and configuration studies, evaluation methods are needed besides the main requirements. These methods must explain to the design engineer the significance of various design parameters. An assessment of a solution alone by the help of an air combat simulation, for ex impie, is not complete; furtheron, such simulations are perhaps more appropriate in later design phases, where sufficiently detailed and precise data are available.

As the block scheme of figure 20 shows, the proposed method starts from the considered missions of the battle-field air superiority fighter. The flying qualities, the performance and other important aspects, as for example the cockpit visibility, the vulnerability etc. are analysed. Flying qualities and performance evaluation are combined using weighting factors. To this evaluation of the aircraft capabilities the evaluation of the further aspects is added by using a second weighting factor. The complete evaluation considers also the cost.

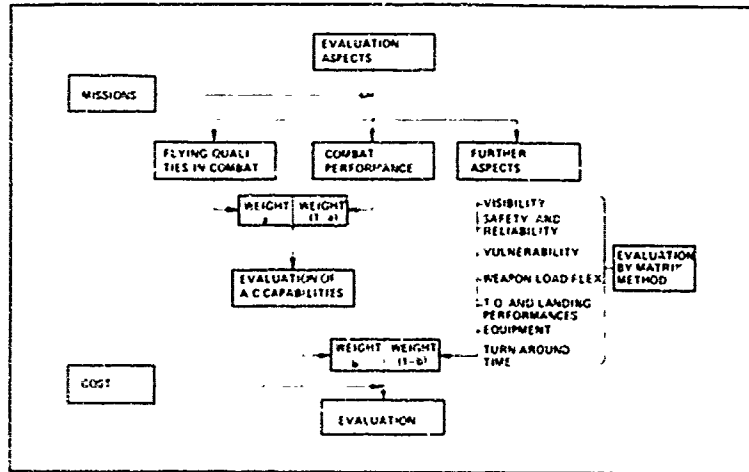


Figure 20: Block Scheme of Evaluation Method

Figure 21 helps to explain, how the importance of the flying qualities is derived. From the analysis of air combats, air combat simulations and pilots debriefings, but also by considering training programs of combat pilots, it is possible to get a representative set of flight maneuvers, which are essential for a battle field air superiority fighter (see column 1 of figure 21). Furtheron, such an analysis yields information to assume a relative frequency of the different maneuvers (column 2). This frequency is taken as weighting of the flight maneuvers. An analysis of the influence of the flying qualities for the different maneuvers allows an estimation of their relative value (see head row of figure 21). The multiplication of these valences with the corresponding weighting of the flight maneuver and the summation of all products for each parameter, results in an estimation of the importance of the parameter considered with respect to the tasks of the battle-field fighter (see lower row of figure 21). As shown in this example, maximum lift and control of normal acceleration have high rank of importance.

FLYING QUALITIES AIR COMBAT MANEUVERS	WEIGHTING OF FLIGHT MANEUVERS	CONTROL OF PITCH ATTITUDE															
		CONTROL OF NORMAL ACCELERATION	CONTROL OF YAW MOVEMENT FORCE	CONTROL OF ROLL MOVEMENT FORCE	MAXIMUM LIFT	PITCH UP TENDENCY, SUPER STALL	BUFFETING, BUFFET ONSET	STALLING BEHAVIOUR (WING ROCK, WING DROOP)	CONTROLABILITY ON STALLED FLIGHT	SPIN TENDENCY	SNITCH ROLL	TURN COORDINATION	ROLL PERFORMANCE	ROLL COUPLING	RUDDER EFFECTIVENESS	SPIRAL STABILITY	
HARD TURN	17	0,1	0,2	0,2	0,2												
BREAK	11		0,2	0,3	0,1	0,1	0,1										
SCISSORS	12		0,1	0,1	0,2												
YO YO	18	0,1	0,2	0,1	0,2												
LUFBERRY	5	0,1	0,2	0,1	0,2												
HIGH G (BARREL) ROLL	4	0,1	0,1							0,1							
VERTICAL REVERSAL	23	0,1		0,1		0,1		0,2	0,2	0,1		0,1	0,1				
HIGH G DIVING SPIRAL	4		0,2	0,1						0,1	0,2						0,4
TOTAL	1	6,7	13,2	13,2	14,9	3,4	1,1	5,7	5,0	3,7	0,4	12,8	7,2	6,7	4,0	5,8	

Figure 21: Weight of Flying Qualities

Analogous to the flying qualities, an assessment of the importance of the different performance parameters can be carried out (see figure 22). The determination of the significance of the other aspects considered is done by using a matrix method (see figure 23). All aspects are compared among one another. The matrix is worked out independently by some experienced people to assure an objective result.

COMBAT PERFORMANCE AIR COMBAT MANEUVERS	WEIGHTING OF FLIGHT MANEUVERS	ACCELERATION AT LOADFACTOR n = 1	ACCELERATION AT LOADFACTOR n = 1	DECELERATION CAPABILITY	MINIMUM SPEED	MAXIMUM SPEED	CLIMB CAPABILITY
HARD TURN	17	0.5			0.5		
BREAK	11		1.0				
SCISSORS	18		0.2		0.4		0.4
YO-YO	18		0.4	0.1	0.1	0.1	0.3
LUFBERRY	5		0.5		0.5		
HIGH G (BARREL) ROLL	4		0.6	0.2	0.1		0.1
VERTICAL REVERSAL	23		0.4			0.2	0.4
HIGH G DIVING SPIRAL	4		0.6	0.2		0.2	
WEIGHT OF PERFORMANCE	100	8.5	38.3	1.4	20.4	7.2	22.2

Figure 22: Weight of Aircraft Performance

	VISIBILITY	SAFETY/RELIABILITY	VULNERABILITY	WEAPON FLEXIBILITY	T/O AND LANDING PERFORMANCES	EQUIPMENT	TURN AROUND TIME
VISIBILITY		0.75	0.75	0.75	1.0	0.75	0.75
SAFETY/RELIABILITY	0.75		0.25	0.75	0.75	0.25	0.75
VULNERABILITY	0.25	0.75		1.00	1.00	0.50	1.00
WEAPON FLEXIBILITY	0.25	0.25	0.00		0.50	0.25	0.50
T/O AND LANDING PERFORM	0.00	0.25	0.00	0.50		0.25	0.50
EQUIPMENT	0.25	0.75	0.50	0.75	0.75		0.75
TURN AROUND TIME	0.25	0.25	0.00	0.50	0.50	0.25	
WEIGHT	0.23	0.14	0.22	0.08	0.17	0.18	0.08

Figure 23: Evaluation Matrix of further Aspects

For the assessment of different solutions of a battle-field fighter so called value functions are needed (see figure 24).

The evaluation of the solutions is carried out by determining at first the different aircraft characteristics. Then, the numbers of the corresponding value function is determined. The multiplication of these numbers with the corresponding weighting factors for each flight maneuver and the summation of all products yield an evaluation number for each solution. For this procedure, a table can be used as shown in figure 25.

The method described herein requires engineering skill and judgement. An appreciable insight into the design problems of a fighter can be gained. The method reveals that the flying qualities have to be considered and estimated already in the beginning of the design process of the battle-field fighter. In case of very new concepts, even wind tunnel measurements can be necessary in these early design phases.

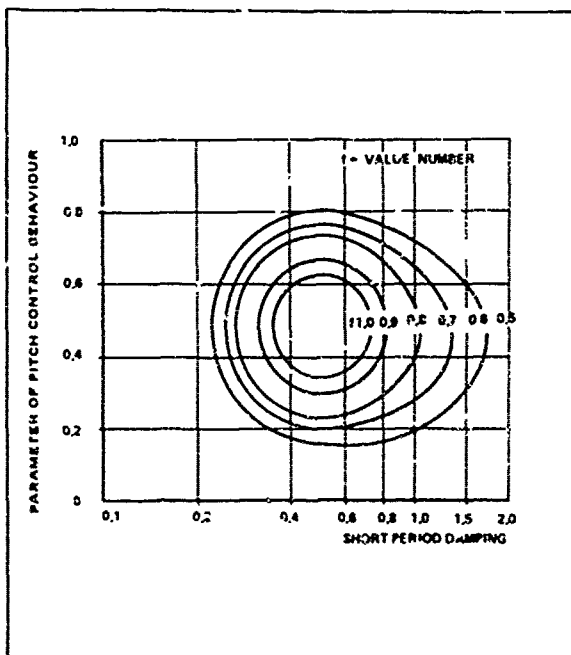


Figure 24: Example of Value Function




	WEIGHTING FACTOR			
CONTROL OF PITCH ATTITUDE	6.7			
CONTROL OF NORMAL ACCELERATION	13.2			
STICK MOVEMENT FORCE PER G	10.2			
MAXIMUM LIFT	14.9			
PITCH UP TENDENCY, SUPER STALL	3.4			
BUFFETING, BUFFET ONSET	1.1			
STALLING BEHAVIOUR (WING ROCK, WING DROP)	5.7			
CONTROLLABILITY IN STALLED FLIGHT	5.0			
SPIN TENDENCY	3.1			
DUTCH ROLL	0.4			
TURN COORDINATEN	12.8			
ROLL PERFORMANCE	7.2			
ROLL COUPLING	6.7			
RUDDER EFFECTIVENESS	4.0			
SPIRAL STABILITY	5.6			
TOTAL	100.0			

Figure 25: Scheme for Evaluation of Flying Qualities

4. Presentation of Concepts

In order to arrive to an optimal technical solution for a battle-field air superiority fighter, three fundamental concepts, the conventional dragon concept, the canard concept and the double delta concept, were considered and for each concept a preliminary design was worked out. These designs are still not optimized in each detail but serve for a preliminary evaluation.

The three-side view of the conventional design is shown in figure 26. The aircraft is a high wing solution with strakes and maneuver flaps. In figure 27, the canard concept is illustrated. The low delta wing allows a favorable arrangement of the landing gear. Furtheron, sufficient space is available for storing equipment and fuel. This concept has the known advantages regarding maneuverability and minimum speed.

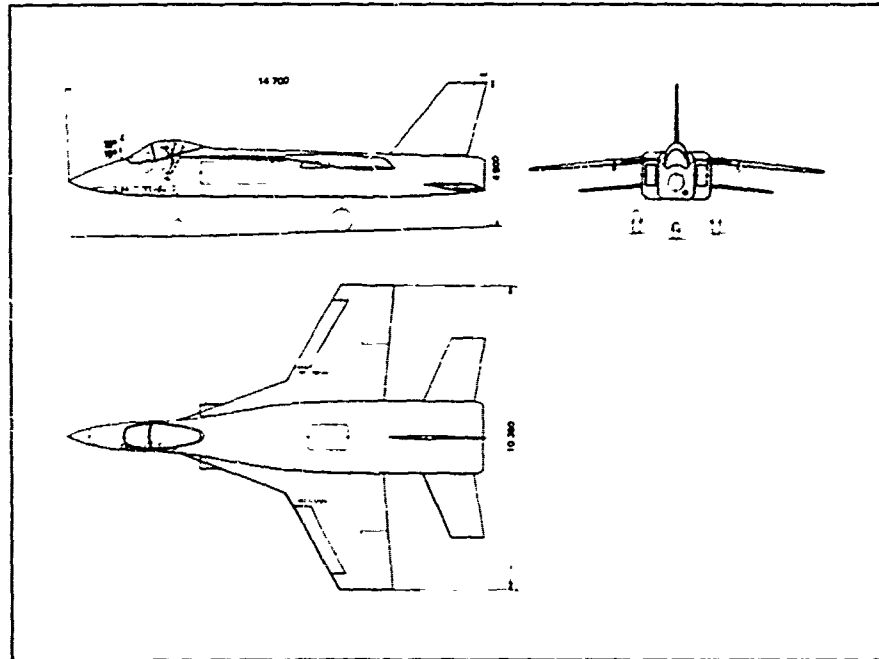


Figure 26: Conventional Concept

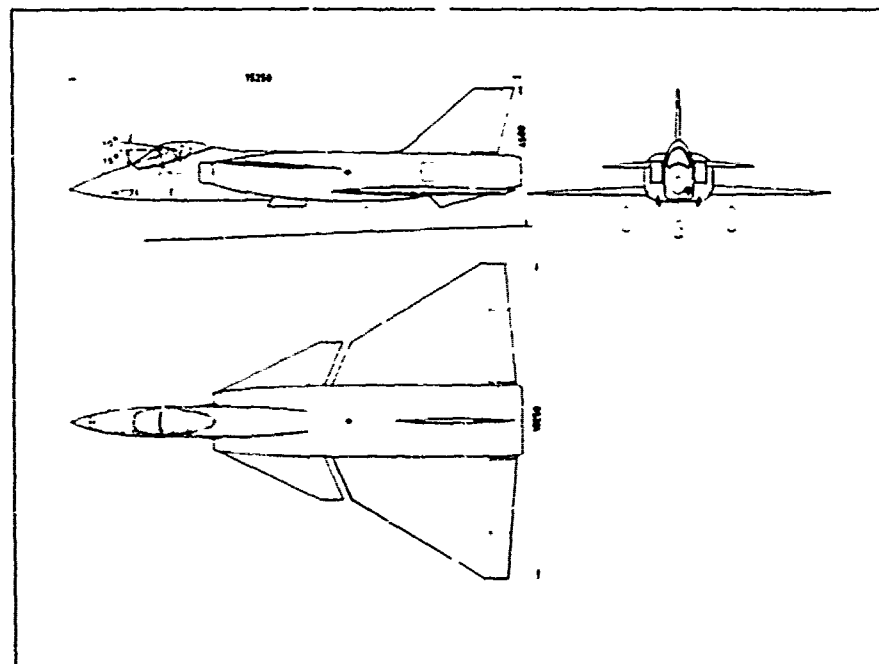


Figure 27: Canard Concept

The third concept, the double delta, is characterized by the fact that the outer wing can swivel (see figure 28). The swivel axis is perpendicular to the longitudinal axis of the aircraft. The outer wings are working as ailerons and as elevators. The arrangement allows not only to avoid the high trim losses of pure conventional configurations during high g-maneuvers but permits, by appropriate change of the wing twist in spanwise direction, to optimize the spanwise lift distribution with regard to minimum speed and induced drag. The chined forebody of the double delta aircraft serves to reduce trim drag at high Mach numbers and to increase directional stability with increasing angle of attack.

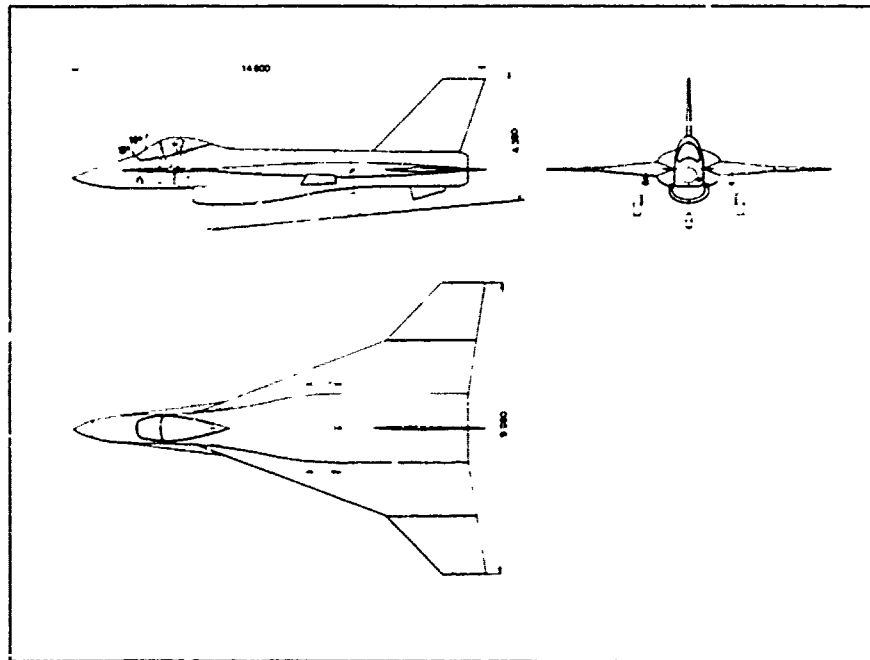





Figure 28: Double Delta Concept

All three concepts are characterized by cockpit layouts for excellent visibility.

A comparison of main data of the three concepts is given in figure 29. The double delta concept yields favorable results in dimensions, weight, thrust and flyaway price. The conceptual designs of the three solutions are worked out for equal thrust/weight ratio and equal span loading. Therefore they do not have exactly the same specific excess power and turn rate in the design point as required. This could be achieved by a further iteration, but even then the performance values would be different at any other point than the design point. Since the design point cannot fully reflect the flight regime of air combat with respect to speed and altitude, the differences in the performance values in the whole regime of speed and altitude have to be considered during an evaluation.

DATA	DIM			
SPAN	m	10.35	9.96	10.25
LENGTH	m	14.7	14.6	15.25
HEIGHT	m	4.9	4.38	4.5
GROSS WEIGHT	kg	10 980	10 100	10 800
COMBAT WEIGHT	kg	9 300	8 600	9 100
FUEL CAPACITY	kg	3 358	3 003	3 402
WING LOADING *	kg/m ²	260	165	190
THRUST/WEIGHT *	kg/kg	1.5	1.5	1.5
SPAN LOADING *	kg/m ²	86.7	86.7	86.7
STATIC THRUST WITH AB	kg	2 x 6975	2 x 6450	2 x 6825
FLY AWAY PRICE	MIO DM	11.1	10.5	10.9

* AT COMBAT (COMBAT WEIGHT INCLUDES 50% INTERNAL FUEL)

Figure 29: Comparison of Main Data

In figure 30, the flight envelopes are shown. The maximum Mach number is limited according to a dynamic pressure of 1 at or a total temperature of 125 °C. A comparison of SEP as function of Mach number at zero and 15000 ft altitude is shown in figure 31. In figure 32, the turn rate as function of Mach number is given. Finally figure 33 shows the specific excess power against the turn rate. The figures show favorable results for the double delta concept.

In order to enable a final choice between the three concepts considered here, the further evaluation, especially regarding the flying qualities has to be carried out as described already. The results available so far show that the double delta concept is very promising.

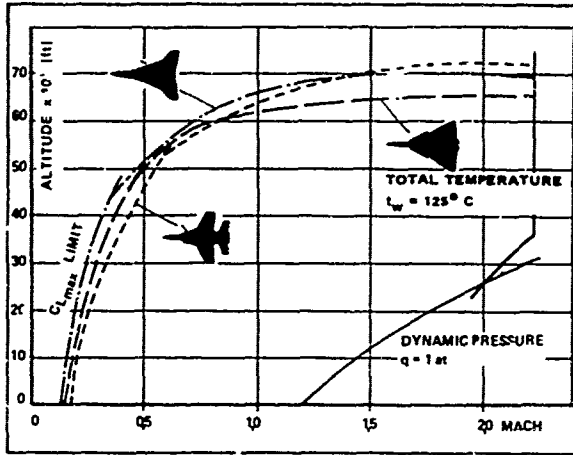


Figure 30: Flight Envelopes

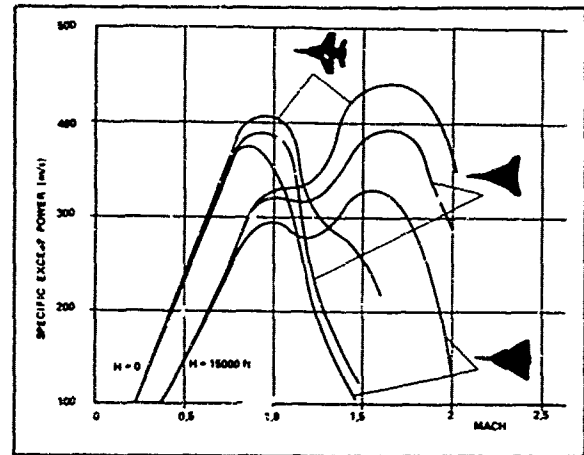


Figure 31: SEP vs Mach Number

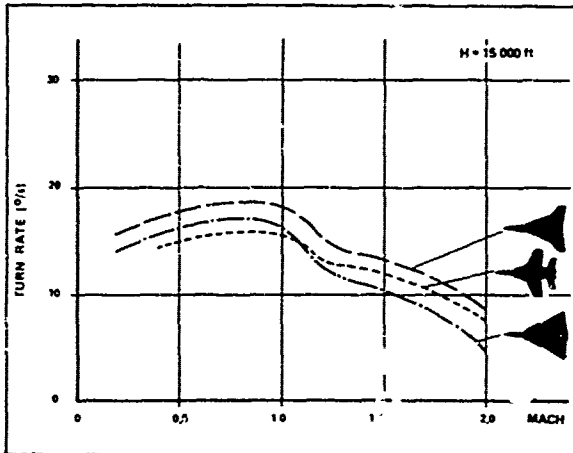


Figure 32: Turnrate vs Mach Number

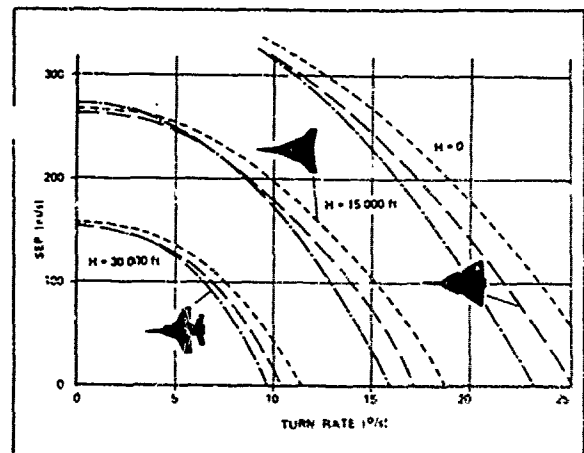


Figure 33: SEP vs Turn Rate

5. Concluding Remarks

Cost studies must be carried out already during the preparation of requirements together with the conceptual design studies in order to reduce the cost of new weapon systems and to avoid highly complex aircraft. Often primary performance requirements can however, not be compromised by cost considerations. For a given budget, the fleet effectiveness can be optimized by reducing the price of the aircraft, thus allowing for a higher number of aircraft. Above all a low cost design can be achieved by minimizing the take-off weight together with the equipment, especially the avionics.

For the evaluation of alternate solutions in the early design process and for optimization, methods are necessary which allow significant insight into the design problems. Such methods could not be substituted by air combat simulation, but complemented. The concept of a battle-field air superiority fighter, which was used here as example for system analysis and cost considerations, promises a highly effective weapon system for air defense at low cost. The applicability of advanced technology, e. g. CCV, could not be treated here. Further studies have to be carried out in this field in order to arrive to an optimum solution for a battle-field air superiority fighter.

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7 Acknowledgements

The author thanks the DORNIER Company and his colleagues for the support in preparing the paper. Special acknowledgements are given to Mr. Peter Starke and Mr. Wolfgang Blessing for their help in the elaboration of requirements and preparation of diagrams, Mr. Thilo Conrady for the formulation of the evaluation method and Mr. Wolfgang Haberland for the design work.

THE B-1 BOMBER - CONCEPT TO HARDWARE

by

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SUMMARY

This paper traces the B-1 from its initial conceptual studies to the hardware which will soon fly. The interaction of the B-1 requirements and advanced technology is given special consideration. Finally, the paper examines in depth the preliminary design process and then compares the hardware product with earlier designs. The "lessons learned" from the analysis are summarized.

The B-1 Strategic Bomber is the latest and most sophisticated aircraft in development. It is planned as a replacement for the B-52. The key requirements are thus initial survivability, improved penetration capability, improved payload-range, and through these greater cost-effectiveness than the B-52.

* * * * *

While the first B-1 bomber is rapidly taking shape as hardware, the airplane design continues to evolve. Since it is the most sophisticated aircraft under development, it deserves--and gets--much attention. Further, since the concept studies were started eight years ago, it is now worthwhile to look at the B-1 development as a case study of preliminary design.

The Advanced Manned Strategic Aircraft (AMSA) program, which led to the B-1, started in 1965. The primary requirement was for a cost-effective replacement for the aging B-52. Thus, the aircraft's mission was to deter general nuclear war by being capable of surviving an enemy first strike, successfully penetrating enemy defenses, and accurately delivering offset or laydown weapons on both industrial and military targets.

Since technology had expanded from 1950 (B-52 period) to 1965, many alternatives, such as V/STOL, all supersonic penetration, stand-off missile launchers, low altitude penetrators, etc., had to be reviewed. The results of these analyses quickly showed that low altitude penetration at high subsonic speed was the preferred mode--a supersonic, high altitude capability further provided flexibility and helped dilute enemy defenses. These modes became the prime requirements for the AMSA studies. The AMSA studies continued for four years--oriented at defining a cost-effective B-52 replacement. Three airframe contractors were funded throughout the AMSA studies. Figure 1 portrays this time history. It can be seen that the studies terminated in late 1969 with the B-1 requirement and Request for Proposal. At the end of the proposal evaluation, June 1970, North American Rockwell was selected to design and develop the B-1. First flight is scheduled for June 1974. The first aircraft is now being assembled at Palmdale, California.

The AMSA studies approached the conceptual definition with a three-pronged effort-- 1) detailed analysis on a point design aircraft, 2) parametric analyses centered around the point design, and 3) specific studies and/or tests in new technology and high risk areas.

I believe the B-1 is a classic example of the concept definition pre-design paradoxes:

- a) Must meet increasing threat vs. what threat can you prove.
- b) Must emphasize new technology vs. too much of an advance in state of the art.

B-1 PROGRAM HISTORY

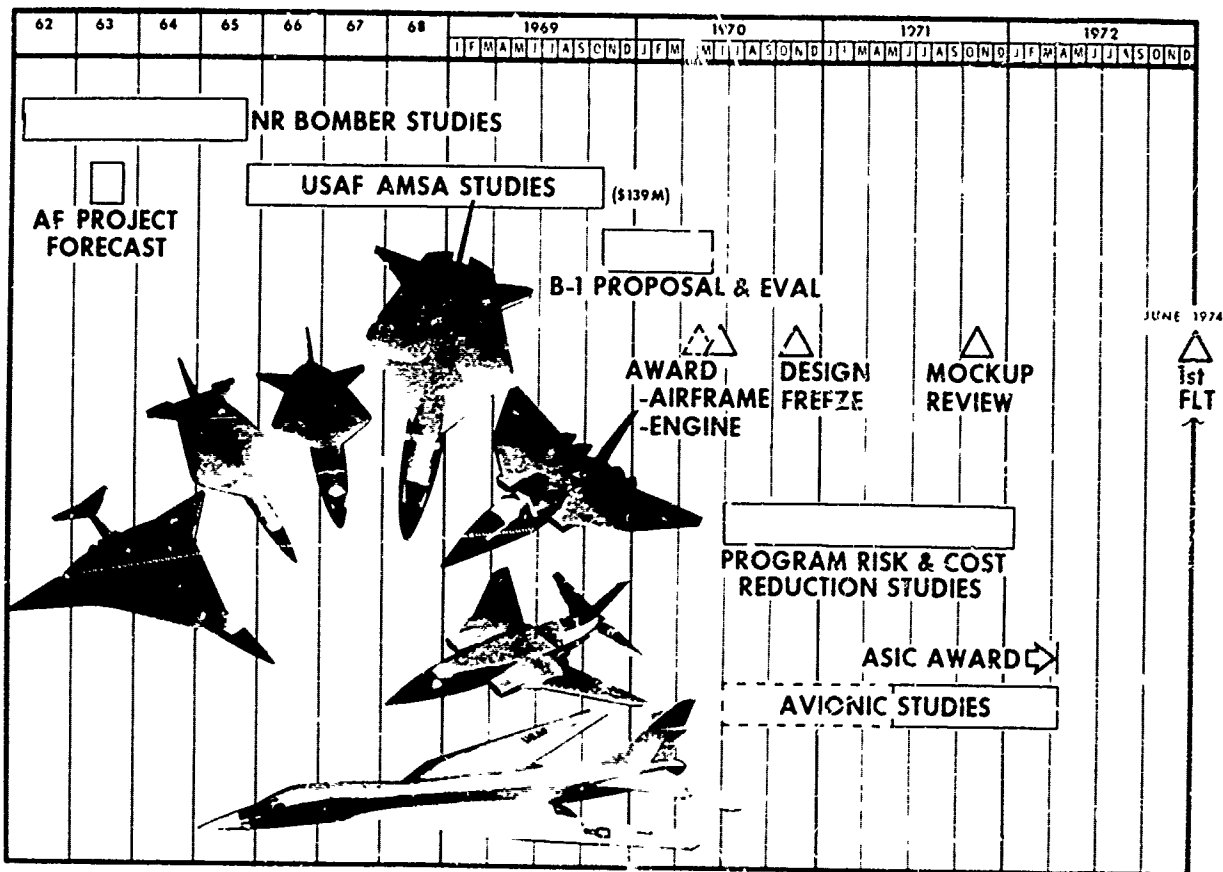


Figure 1

c) Present systems are good enough vs. not enough better than present system.

Accordingly, the concept definition phase was long and slow. The parametric studies were very extensive trying to provide data to support a firm requirement. These studies varied payload, military load, and design. A brief overview of this process is shown on Figure 2. Preliminary Design Review (PDR) is taken as the key time--that's when there finally became one configuration with its specific drawings. This doesn't mean that when PDR was over, we stopped making changes--but when changes were required (or desired) it did mean making drawings over.

Now I'd like to go back to those AMSA studies and look at some of the requirements, configurations, technology studies and results.

Since cost effectiveness was a major criteria, payload (SRAM - Short Range Attack Missiles - and SCAD - Subsonic Cruise Armed Decoy) was a major variable. The B-52 carries SRAM on a rotary rack with eight missiles attached. For the new bomber it was natural to look at multiples of this design, i.e., 16, 24, 32 SRAM's. With 24 SRAM's (three times B-52 payload) the impact is severe as shown in Figure 3. Here we can start to see the required advances in technology.

CONFIGURATIONS TO DRAWINGS

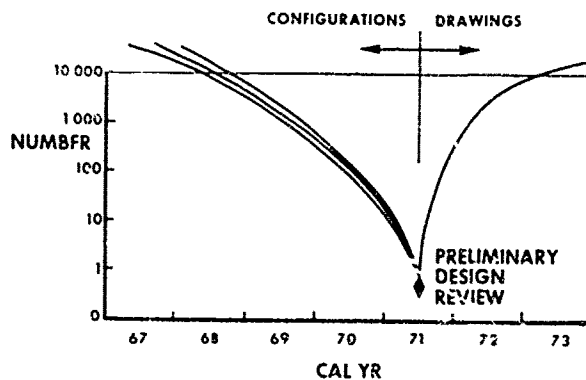


Figure 2

WEIGHT FRACTION COMPARISON

B-52	22%	20%	5%	53%
AMSA (B-1)	22%	22%	13%	41%
	STRUCTURE	SYSTEMS ETC	PAYLOAD	FUEL

a) Hold structural fraction at 22% while incorporating variable sweep and large payload bays.

b) Hold systems fraction to within 10% of that of B-52 while adding terrain following radar, improved sensors, additional penetration aids, etc.

c) Accomplish required range with 29% less fuel (41% vs. 53%).

Figure 3

In order to hold the structural fraction while incorporating large weapon bay cutouts and variable sweep, new materials and alloys were studied. Composites (both boron and carbon) were also investigated. High strength alloys of aluminum and steel were considered. Unique structural design concepts were also looked at--such as blended wing bodies to minimize wetted area and maximize structural depth. The primary effort was centered about titanium and the most efficient ways to use and manufacture it.

The segment called "systems" offered many challenges. This segment includes: Engines, offensive avionics, defensive avionics, electrical power, hydraulic, cooling, escape, etc. In each of these areas major developments were undertaken.

A lightweight augmented turbofan demonstrator program was initiated. The developments included-- a) high turbine temperature (cooled blades), b) short annular combustor, c) short mixer and augmentor, d) minimum length design. Two contracts were let for the building and running of technology demonstrator engines to incorporate the above features.

In the offensive avionics areas new technology offered promise of simplification, higher reliability and easier installation. Digital systems were seen as the key with emphasis on phased array type antennas doing multiple functions. A multi-mode radar hardware development program was established to pursue this possibility. A bread-board system was constructed and flown in a C-135.

Defensive avionics offered an even greater challenge. New concepts using digital technology appeared necessary to support the penetration requirements and provide adequate future growth and flexibility. Development programs for key technical features were supported. The technical feasibility was demonstrated.

The requirements for initial survivability lead to dispersal and hence to a need for a Central Integrated Test System (CITS). The primary purpose of CITS is to provide assurance while at a dispersed site that the airplane is ready to go (thus replacing flight line Aerospace Ground Equipment (AGE)). This new system was another challenge to define and implement--and also to the weapon system designer to include within weight and cost targets.

The last challenge was to the aerodynamicist--how to do the mission with 25% less fuel. The engine technology programs offered 10-15% improvement in specific fuel consumption. Variable sweep was the primary aerodynamic advancement. The AMSA designs were initially like the F-111's and started from the same NASA data base. As the configuration evolved it became apparent that the larger payload fraction and aircraft balance required the engines to be near the aerodynamic center rather than at the rear as in the F-111. This new arrangement led to problems of fuselage heating, horizontal tail placement, etc. Wind tunnel tests on the various configuration alternatives were conducted. It was evident that the requirements resulted in a difficult design problem--and that any selected configuration would require much tuning up.

Let us turn to the specific point design aircraft and their evolution. In 1967, the point design aircraft had the design shown on Figure 4 with the engines at the rear. The percentages for structure, systems, payload and fuel are shown on Figure 5. The gross weight was 350,600 pounds.

AMSA - 1967

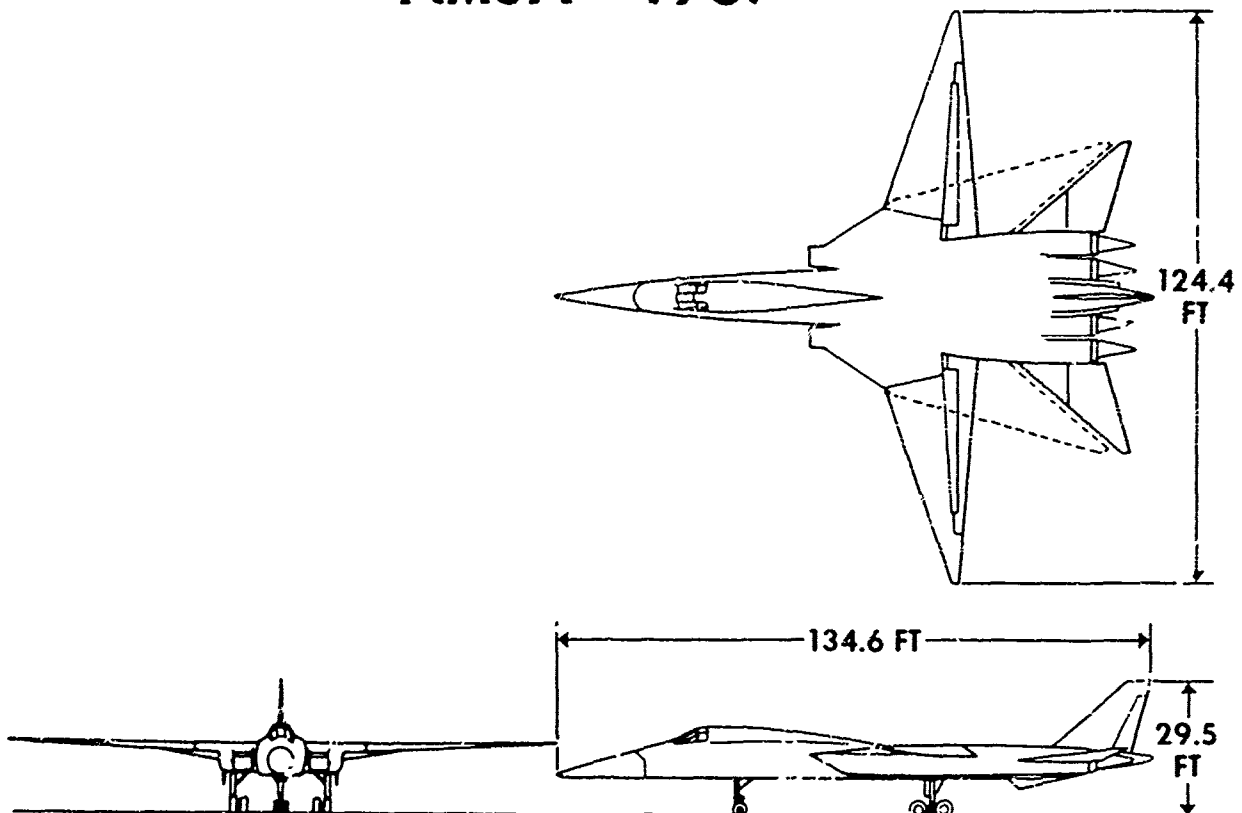


Figure 4

WEIGHT FRACTIONS

1967 AMSA

24%	15%	14%	44%
STRUCTURE	SYSTEMS ETC	PAY- LOAD	FUEL

Figure 5

(U) By 1968 the configuration had changed recognizing the need to separate the tail and wing (Figure 6). The required internal payload had been increased to 32 SRAM's as a part of the concept definition and the design. The primary variables in these studies were gross weight, range, and payload. But due to the many questions on requirements, a large number of trade off studies were accomplished--some to substantiate requirements, others to evolve and refine the requirements. Typical examples of these are:

- a) Landing gear flotation requirements vs. dispersal capability.
- b) Crew escape modules vs. ejection seats.
- c) On-board integrated test capability vs. ACE.
- d) Ride quality vs. crew effectiveness.
- e) Nuclear hardness vs. initial survival.

(U) Since cost effectiveness was the prime objective, these studies concerned both relative effectiveness in destroying a given target system, and the total system cost. The most difficult aspect was cost. For most of the studies gross weight, and its associated weight empty were the primary cost input. The point design aircraft were used to do a more complete cost analysis which became the base line for the trade off studies.

AMSA - 1968

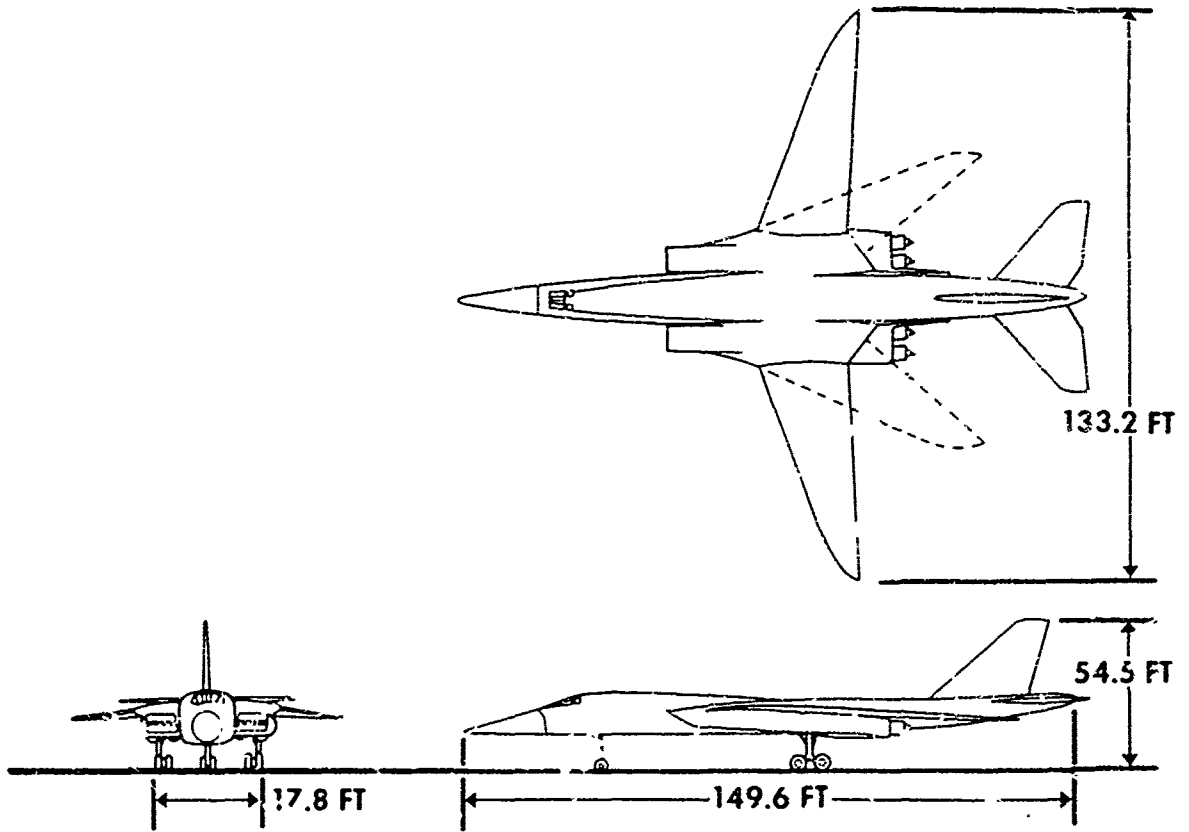


Figure 6

EFFECT OF CREW SURVIVAL SYSTEMS

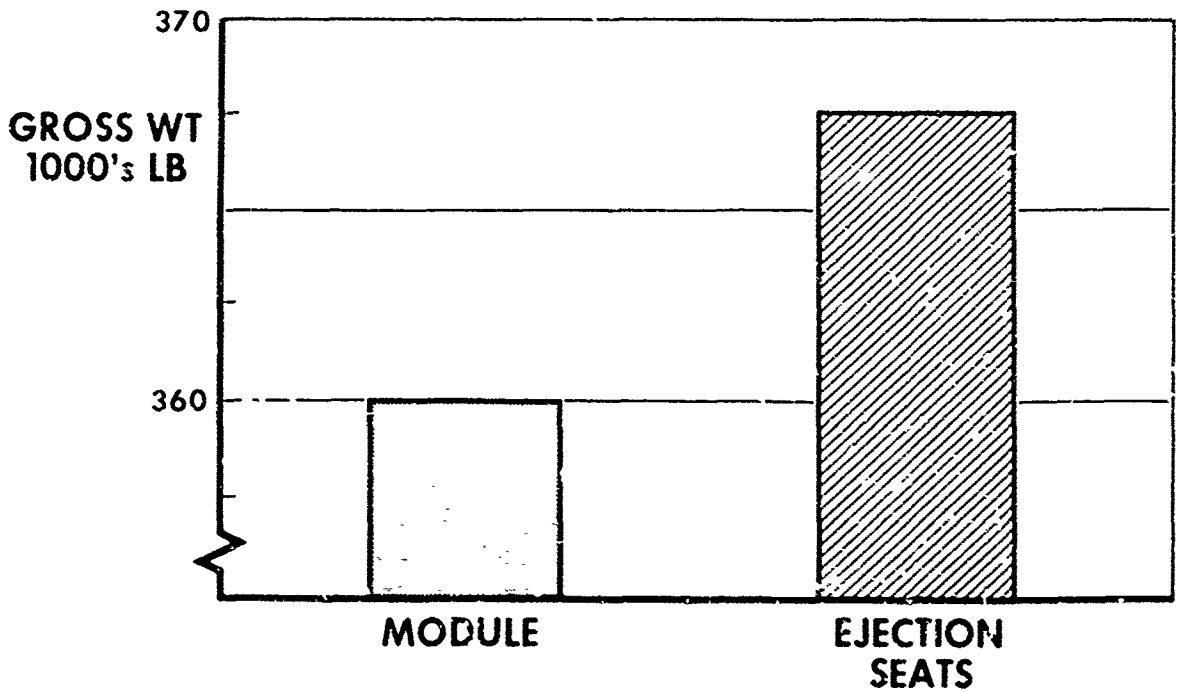


Figure 7

The results of the crew escape study are summarized in Figure 7. Alternate designs were made with--six ejection seats and six-man module (somewhat similar to that in the F-111). While the B-1 has a basic crew requirement for four, the training mission will often result in six being on board, accordingly a requirement for six men escaping. When these designs were then inputted to the parametric study program and the aircraft all sized to the same mission capability (range/payload), the aircraft with six ejection seats was heavier than that required with a module. Hence, the cost was greater even discounting the added cost for the module. The operating command felt that crew effectiveness would be better in the shirt-sleeve environment of the module. The requirement for a crew module was thus firmly established.

By 1969 thousands of configurations had been analyzed and detailed requirements established. Further, the B-52's were continuing to age. Accordingly, the Air Force requested Department of Defense (DOD) authority to proceed with the B-1. The next phase of the preliminary design began. Most of the studies in the concept definition phase were concerned with relative cost-effectiveness. Real world costs and programming considerations were now injected into the B-1. Undoubtedly, some of these negated assumptions made during the earlier studies. Efforts were made to account for all elements as could be identified.

The concept definition phase developed a design--including most detailed characteristics--responsive to the described threat, technical state of the art and in accordance with earlier type programming ground rules.

In the fall of 1969, the Air Force received permission to proceed with the B-1 program. Requests for Proposal were issued. The program was structured as a "Fly Before Buy" type. Hence, the request was limited to design, development, manufacture, and test of five test aircraft. The three companies which had participated in the study phases submitted bids. North American Rockwell was selected in June 1970 as the Weapons System contractor. General Electric was selected to develop the engine.

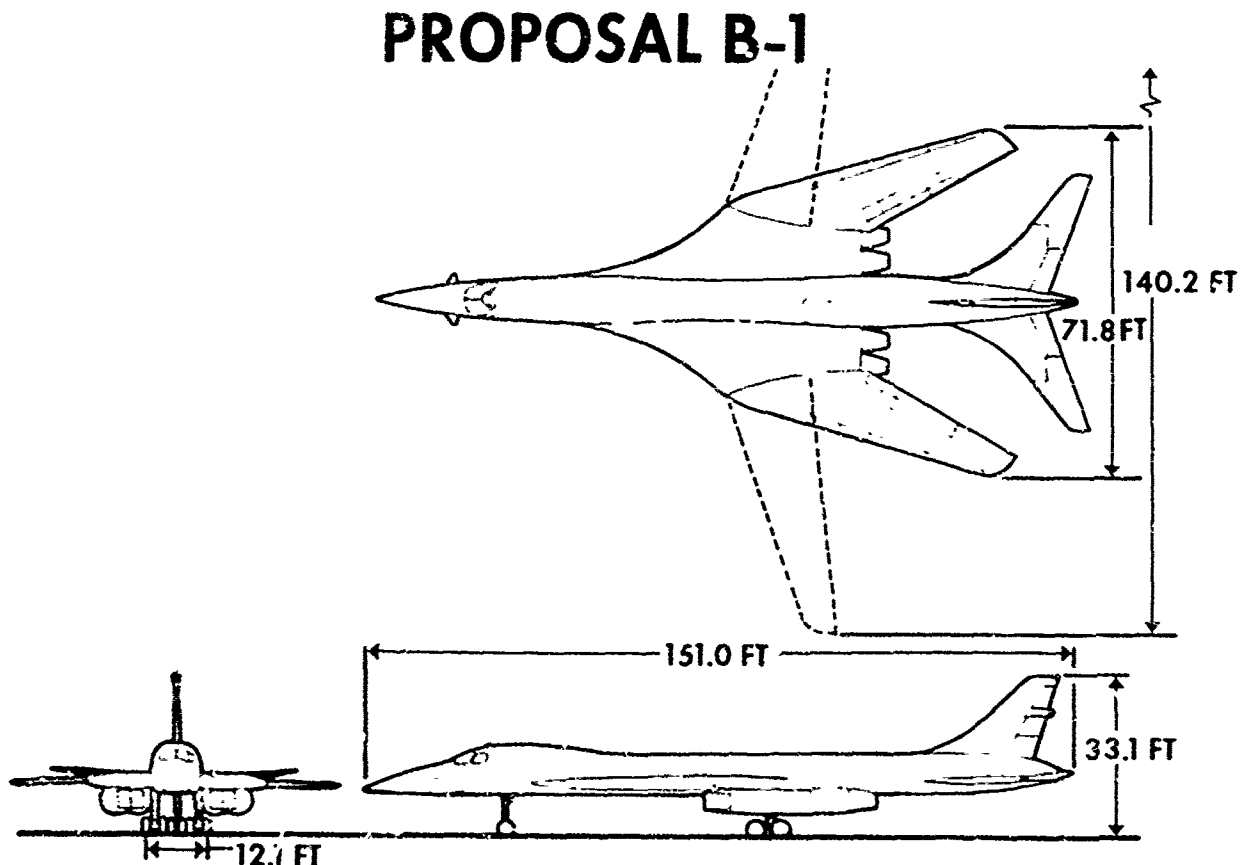


Figure 8

The winning configuration is shown in Figure 8. North American Rockwell included many features to minimize the size of their airplane. They proposed a 4,000-psi hydraulic system, a 230-volt electrical system, electrical multiplexing, structural mode control fins, maneuvering rocket control system for crew escape module, and a high percentage of titanium in the basic structure. The design featured a variable sweep wing, large horizontal tail, and mixed compression inlets (to satisfy the supersonic requirements).

When the Air Force sought approval to proceed in the spring of 1970, the programming realities of life further impacted the concept. The dollars available were less than those required. Conditional approval was given; but requiring major changes to get within the planned dollars. It was recognized at this time that the advanced avionics assumed in all the studies were beyond the available dollars (and beyond the immediate requirements). All avionics work was stopped--although the aircraft design was to continue using the advanced system definition. The assumption was that existing avionics would fit within the space, power, cooling limits and the aircraft would have potential to take advanced avionics later in its life.

The first major contractual task after contract award was to change the design and restructure the program to be compatible with the existing and projected money available. This effort was named Project Focus. The changed philosophy with regard to avionics has been mentioned. Other major requirements were also re-examined in line with money limitations rather than cost effectiveness. Finally, since a contractor had been selected and a specific design existed, more detailed cost trade offs could be made--weight was no longer synonymous with cost. The percentages of titanium, steel, and aluminum were varied, recognizing their respective impact on cost. The result of this trade study is shown in Figure 9. It can be seen that above about 20 percent titanium the unit cost increased; whereas the gross weight to do the mission continues to decrease. Accordingly, the percentage of titanium was dropped from about 40 percent to less than 20 percent.

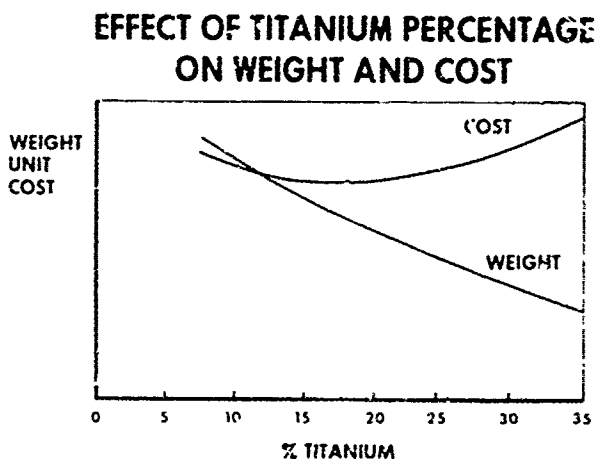


Figure 9

As a result of Project Focus, the requirements were modified as shown in Figure 10. The design gross weight was increased (350,870 lbs to 360,000 lbs). The modified design included other refinements--it is shown in Figure 11. The inputs from advanced technology had now been reduced in two key areas--avionics and materials. The reduction in use of titanium was strictly due to cost. It was retained in critical areas. Further, the North American Rockwell concept of Diffusion Bonding titanium was retained. In this same time period, a requirement for fracture mechanics was added (as a result of problems on the F-111 with high strength steel alloys). This requirement placed emphasis on new materials with superior fracture mechanics qualities. Advanced avionics technology was not entirely abandoned since the design was required to have a growth capability to include future all-digital systems.

The aerodynamic technology requirements were straightforward. Variable geometry was a necessity and leaned heavily on F-111 development. The take-off distance required a sophisticated flap/slot system. The structural mode control system was a new approach to minimize bouncing at the crew station and to save weight. Finally, the definition of the engine/inlet distortion recognized the high level of technology necessary in this area. Shortly after Project Focus there was an exercise involving program changes--the five flight test aircraft were reduced to three and much formal qualification work was postponed to agree with the funding plan. There still remained many technical challenges.

FOCUS CHANGES TO B-1 REQUIREMENTS

ITEM	CHANGE
• Takeoff Distance	Increased 500 Ft
• Supersonic Distance	Decreased 100 Mi
• Refuel Altitude	Decreased 500 Ft
• Thrust/drag at SL 0.85M	Decreased 10%

Figure 10

FOCUS AIRPLANE

-55B

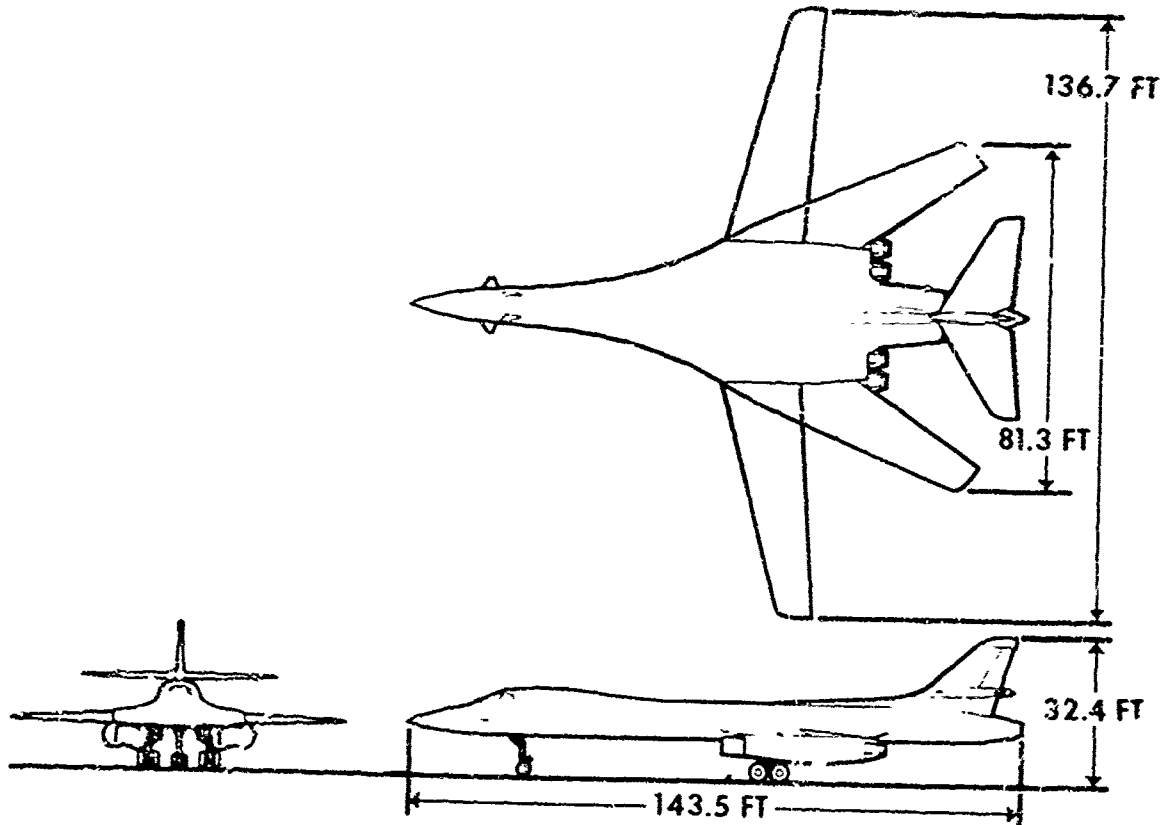


Figure 11

While some of the configuration technical problems were resolved during Project Focus (such as tail location), other details needed solutions. These were worked out during early 1971. The Preliminary Design Review was held in July 1971 and the Mock-up in October 1971. Thus by mid-1971, the configuration had finally settled down to a single basic design. From that time on changes became more costly--drawings had to be stopped, and with time, tooling changed and material scrapped. This third phase has continued to be dynamic because in some cases the detailed design wasn't adequate; in many other cases it became obvious the cost was greater than assumed in earlier studies--and hence some of the features were no longer desirable. Further, the emphasis on Design to Cost has continued to modify the program concept. With these changes has come a different relationship to advanced technology. Let us examine some of the specific examples of this period.

The flap system started out as a double slotted translating design. Detail design showed this to be very complex and heavy. Wind tunnel tests indicated a simpler, single slotted flap could be designed to meet the requirements. The slat was extended slightly inboard resulting in high lift capability in excess of requirements.

The crew escape system was proposed with a primary rocket motor and a gimbaled (both axes) maneuvering rocket motor. The system was simplified to one motor design--two motors with one for roll and one for pitch--resulting in a slight reduction in low altitude adverse attitude capability and a big cost saving.

The formulation of the avionics package was a major chore. Using the previous program guidance, a suite of off-the-shelf subsystems (forward looking radar, stellar/inertial platform, doppler, terrain following radar, radar altimeter, etc.) were proposed together with a central computer complex. This approach offered much flexibility and straightforward growth to more sophisticated systems. Proposals were requested and evaluated. Examination of them indicated that the development would be expensive and risky. Accordingly, the computer was revised to a modified off-the-shelf version. Further, the offensive and defensive segments were separated.

Before the avionics could get started on contract, two events occurred:

- (1) A complete reassessment of the inertial navigation requirement was requested, and
- (2) The defensive systems were put into a different category--a competitive initial phase to reduce both technical and cost risk.

The airplane design changes that occurred throughout this period are summarized on Figure 12.

The big challenge with the B-1 today is how to use technology to reduce or keep costs down. The concept of "Fly Before Buy" is fraught with problems--what's "off-the-shelf" today may be out of production tomorrow--when it's needed to support B-1 production. One way to avoid excessive cost being charged for "off-the-shelf" systems or sole source components is to use technological advancements. In the past, these have normally been oriented solely to improve performance. Today the B-1 SPO is working closely with the Air Force laboratories to insure that a number of their developments are "designed to cost" so that their price is no more than current subsystems. Further, significant improvements in reliability and maintainability are specified. These may detract from the performance improvement that could be gained, but often present a more severe challenge to the developer. A number of programs in the advanced development sphere are being closely watched for eventual incorporation into the B-1.

One of the first of these is the Advanced Metallic Structures wing carry through (WCT) box. Two concepts of WCT are being designed and evaluated--one will be built and tested. This box is being designed to the B-1 dimensional and structural requirements. By using advanced materials and concepts, a significant cost reduction (as well as weight reduction) is expected. Further, the Air Force will have an option so that this technology may be incorporated into later B-1's.

In the avionics area, a new, cheap, highly reliable inertial platform is under development. The B-1 avionics subsystem is being designed so that the inertial unit may be easily changed. If this development is successful, it will offer improved performance at lower cost.

DESIGN REFINEMENT STATUS

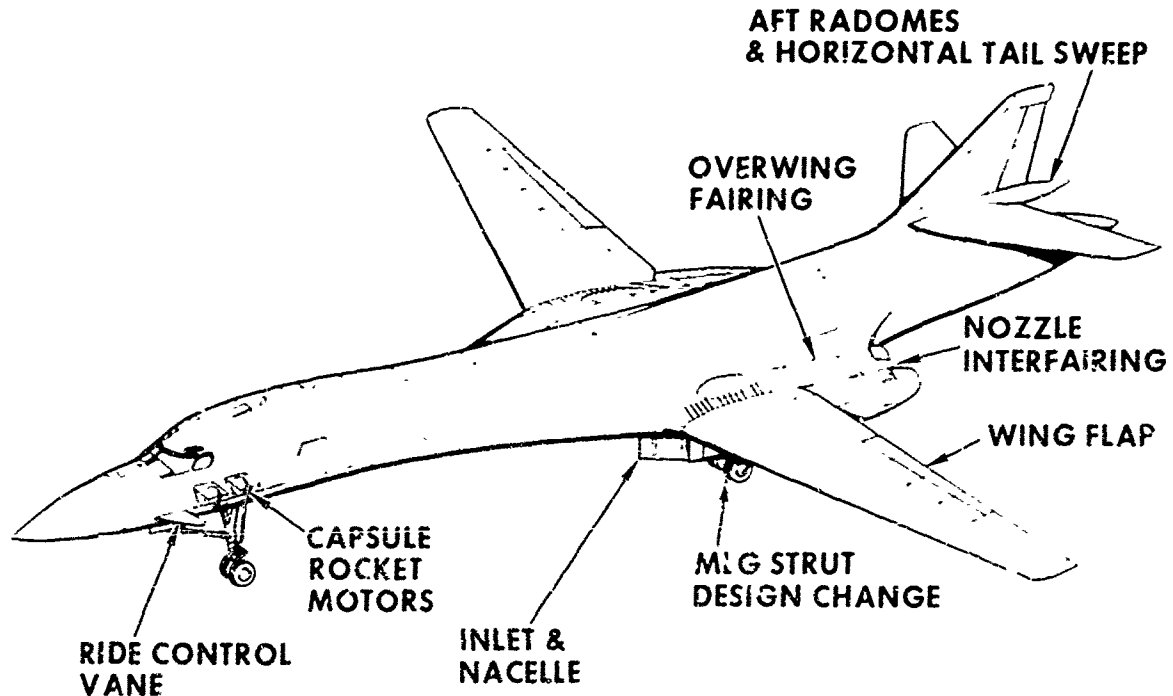


Figure 12

Another avionics area still of interest is that of computer controlled radars (using phase shifters). The Air Force has an advanced development program for such a radar. By using advanced technology, it is expected that a single forward looking radar with electronic scanning can replace the present forward looking radar, terrain following radar, and doppler radar. The new system will be nuclear hardened, be much simpler mechanically, and have considerably better reliability. This one development thus offers a cost-effective alternative to three current systems. The major difference from the earlier advanced development program is the emphasis on "Design to Cost."

The B-1 management is extremely interested in the Air Force/NASA Transonic Aircraft Technology program. This program is developing and testing a supercritical wing for the F-111. The supercritical wing thus offers growth performance capability (through higher cruise speeds) and possibly reduced cost through a simpler wing structural design. The variable sweep design makes it easy to switch to a new wing panel.

From the above examples, it should be evident that the B-1 program is vitally interested in new technology programs. The "Fly Before Buy" concept poses many problems in restraining cost when proceeding into the production program. New technology offers an excellent alternative to existing hardware which has gotten expensive.

The first B-1 airplane is going to fly next year. Subsequent aircraft will incorporate new features--to reduce cost, to improve performance. As problems are uncovered in the test program, new technology will be used to help solve them.

In looking back at the B-1 Concept Definition and Preliminary Design phases, what lessons would be considered most useful for future programs? First, the program effect on cost may invalidate many early preliminary design trade offs and conclusions. The remedy to this is to limit the extent of integration of this design--keep cleaner interfaces--propulsion, airframe, avionics, and weapons--and provide specified engineering type reserves in these areas. Second, recognize that technology can help reduce costs--particularly if properly motivated. Results on the B-1 indicate this is a new and promising discipline. It is challenging because an inadequate data base exists in the cost area. Engineers need to study and quantify the needed cost inputs. They must work closely with estimating personnel in order to properly guide the thrust of technology.

When one considers that the B-1 is 20 years newer than the B-52, it is a real bargain--technology has been used to improve performance and reduce cost. The B-1 has twice the payload, three times the altitude speed, half again the sea level speed and the same range capability of the B-52. We will continue to need technological breakthroughs to hold the price down.

DESIGN OF VERY LARGE AIRPLANES FOR LEAST SYSTEM COST

by

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SUMMARY

The commencement and success of a new program are virtually impossible unless the predicted system cost constitutes a step improvement over existing designs. The concepts proposed in this paper for very large airplanes stress lower speeds ($M = 0.65$) as the key to lower system cost. This speed leads to airplane designs with fewer and simpler parts. The lower cost of parts is further enhanced by the predicted reduction of engineering changes resulting from the simpler designs. Designing for adaptability to a wide range of missions without compromising the initial program is the other fundamental feature of low cost system design. It reduces the investment for new applications and spreads the original investment over a larger production. The application of the proposed design principles is illustrated with existing and proposed aircraft designs.

INTRODUCTION

Over the decades, the costs of new airplane programs have been escalating for a number of reasons. Worldwide inflation has played its part, but a parallel proliferation of requirements has acted as an even more significant cost booster. Our increasing knowledge and the computer have made it possible to analyze phenomena that were barely known to exist in earlier days. Increasing cost and the need to reduce program risks call for more and more safeguards, analyses and tests. This positive feedback loop further aggravates the outlook for future program costs. To justify the higher cost product, the designer is forced to press for more performance. This acts as a second, positive feedback loop, further increasing costs and calling for more safeguards against even minimal risks. The resulting cost trend is illustrated in Figure 1. This cost trend applies to military and commercial programs. Parallel with this cost escalation has gone a reduction in military program starts as shown in Figure 2

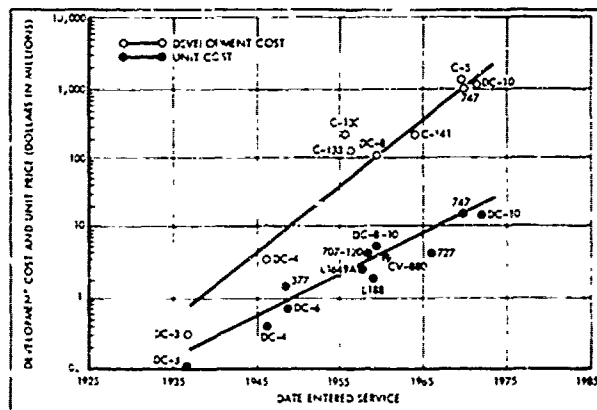


Figure 1: Airplane Program Costs

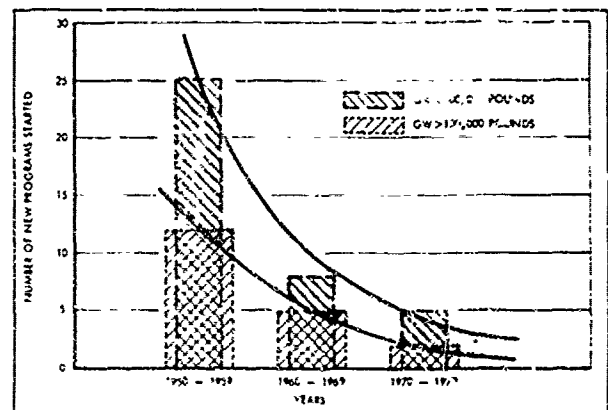


Figure 2: New U.S. Military Programs

It appears that reversing the trend of increasing cost is a necessary step to get a new, large airplane program started. Unless the predicted system cost and predicted return on investment show quantum jump improvements over existing systems, the new program simply will not happen.

In figuring out the return on investment, total system cost must be considered. The total system cost is composed of development cost, acquisition cost and operating cost. Expensive systems are often advocated and justified on the basis of savings in later operating costs. Discounting of future costs and benefits is used to compare systems on the basis of present value. The principle is illustrated in Figure 3. Discounting favors systems with low acquisition costs. Low initial investment also has a strong emotional appeal.

The need to reduce program costs applies equally to commercial and military systems. In the latter, operational capability takes the place of return on investment. A favorable ratio of operational capability to total life cycle cost is essential to a healthy national economy and a strong national defense. The next section presents some technical concepts that appear to have the potential to significantly lower development and acquisition costs for new, very large airplanes.

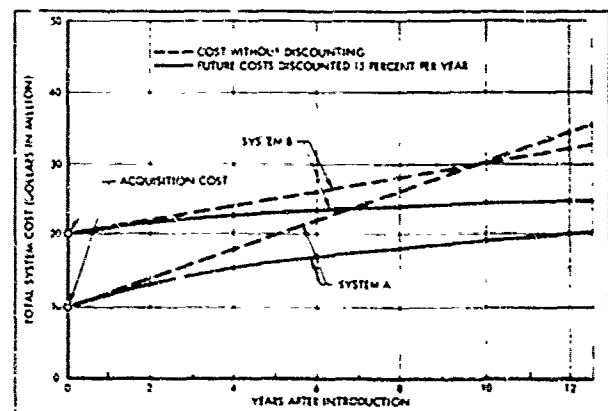


Figure 3: Total Life Cycle Cost

LOW COST CONCEPTS

Identification of the sources of high costs is necessary if a significant cost reduction is to be achieved. Past cost investigations conducted by The Boeing Company have yielded the picture shown in Figure 4. It may come as a surprise to some that management action taken after program go-ahead, when less than three percent of the eventual program cost has been spent, can influence the total

program cost by twenty percent at the most. The program cost is largely determined by the time the market is identified and the airplane concept formulated. Figure 3 illustrates the leverage management has on the unit cost of a mature program. While the foregoing charts were derived from commercial programs, the principles illustrated apply equally to military programs.

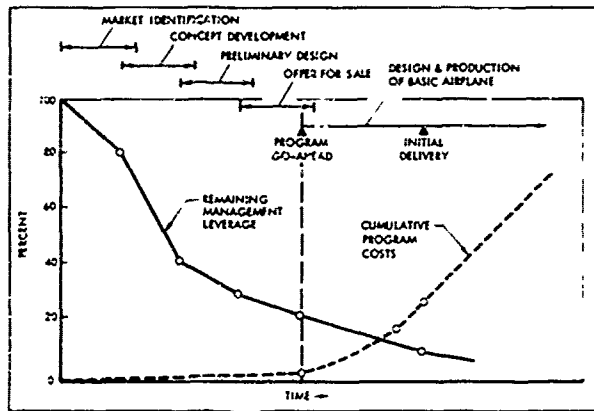


Figure 4: Cost Management of a Typical Commercial Program

MAJOR COST ELEMENTS	PERCENT OF TOTAL	DEGREE OF LEVERAGE
LABOR AND CONTROLLABLE OVERHEAD	33%	HIGH (± 20%) = 6.6
FIXED OVERHEAD	6%	LOW = -
MATERIALS (LESS ENGINES)	23%	MEDIUM (± 10%) = 3.3
ENGINES	23%	LOW (± 5%) = .7
OTHER	5%	MEDIUM (± 10%) = 0.5
TOTAL		11.6

Figure 5: Management Leverage on Cost of Mature Program

The market for the very large airplanes considered here is cargo, military and commercial. The former includes weapons carriers tankers and similar aircraft. Speed has been a major driving force in the design of passenger aircraft and will possibly continue to be. High speed is not a factor in the cargo market, military or commercial. Competition and comparison in the cargo market involve other transportation modes truck, railroad and ship. An hour's difference in transcontinental block time or comparable variations over intercontinental distances are irrelevant by comparison. Productivity, however, is very important, and it is affected by speed. It will be shown that, in the speed range proposed, the gain in payload fraction overpowers the effect of lower speed.

The absence of speed as a driving force is the basis for the airplane concepts expounded in the next paragraphs. Their common denominator is a cruising speed of approximately $M = 0.65$. It will be shown that selection of this cruising speed not only reduces development and acquisition costs but additionally lowers the operating costs.

When we examine the reasons for lower development and acquisition costs, we find that underlying all other effects are the simpler aerodynamics resulting from the slower speed. The wings can be straight and the fuselage a simple shape. Thicker airfoils and blunter bodies are acceptable. The flow is generally streamwise and simple tailoring of components is possible.

Frontal area is less critical than at higher speeds, making it possible to provide sufficient space for all systems and components. These facts add up to fewer conflicting constraints on the airplane designer, to greater design simplicity and to more freedom in selecting design solutions for low cost. Figure 6 depicts the weight trends as functions of speed. The higher payload fraction achievable at lower speeds makes the airplanes less sensitive to operating empty weight variations.

The fundamental simplicity of the airplane designed for the slower cruise speed is illustrated in the exploded view of Figure 7. Its major characteristics are straight wing, cylindrical fuselage, all identical propulsion installations, all identical landing gears, no leading edge devices, simple hinged flap, symmetrical, all moving horizontal stabilizer without separate elevator surfaces.

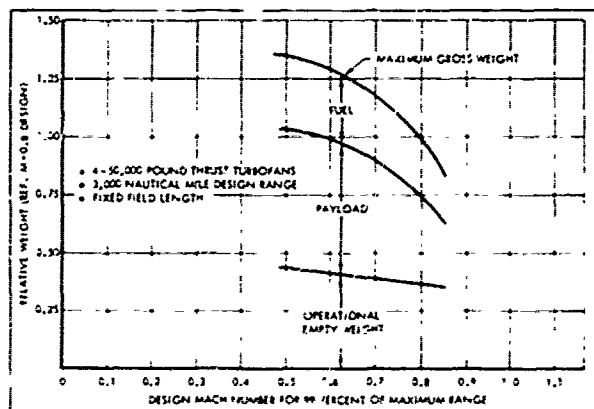


Figure 6: Weight Trends

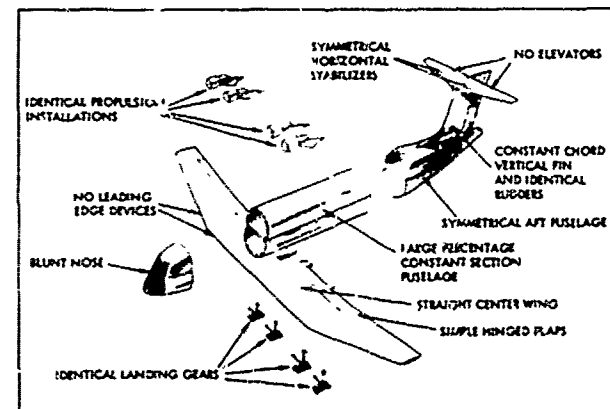


Figure 7: Concept Simplicity

Intuitively, we all feel that simplicity must be reflected in lower cost, but do the facts bear this out? It can be shown that there are solid, factual reasons why simplicity does reduce cost. Referring back to Figure 7, it is apparent that there are fewer different parts in the simple design. Fewer different parts means fewer tools, fewer setups, smaller inventory and less paperwork. A larger quantity of identical parts propulsion or landing gear components for example means lower cost because one moves down the learning curve more rapidly. The cumulative effect expected on cost is illustrated in Figure 8.

Similarly, one would expect simpler parts to cost less than complex parts. While this is undoubtedly true, the direct effect becomes relatively small with today's fabrication and machining methods. The indirect effect, however, that there are fewer errors in simple designs than in complex ones, is very significant.

The combination of fewer parts on which to make errors and simpler parts with fewer errors per part leads to a significant reduction of changes on the production floor. The effect of changes on manhours is shown in Figure 9. This is where the real cost saving due to simplicity occurs.

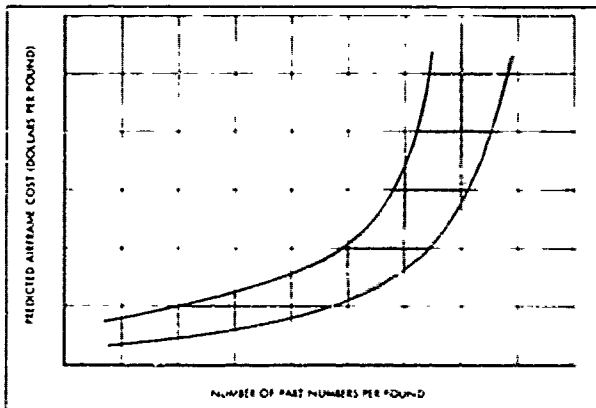


Figure 8: Effect of Part Count on Cost

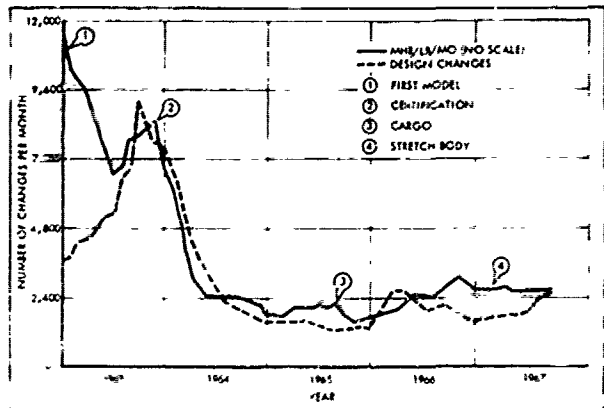


Figure 9: Effect of Design Changes on Production Manhours of Model 727 Aircraft

Indicative of the escalating development costs are the wind tunnel hours accumulated on past programs. Figure 10 shows that the wind tunnel hours spent in development have climbed from less than a thousand on the Model 314 Clipper to over 15,000 hours on the 747. This trend can be reversed with the proposed moderate speed concepts. The Compass Cope remotely piloted aircraft that made its first flight this summer had less than 100 wind tunnel hours prior to first flight. As will be shown later, it embodies some of the important concepts proposed here.

Successful airplane programs of the past, military and commercial, have all been characterized by the adaptation of the basic design to a multitude of different missions. This was as true of the DC-3 as it is true of the C-130 and the 707. Moderate speed design concepts enhance this adaptability. Figure 11 illustrates how one basic design adapts to the roles of cargo airplane, liquid natural gas transporter, missile carrier, tanker and sea control airplane.

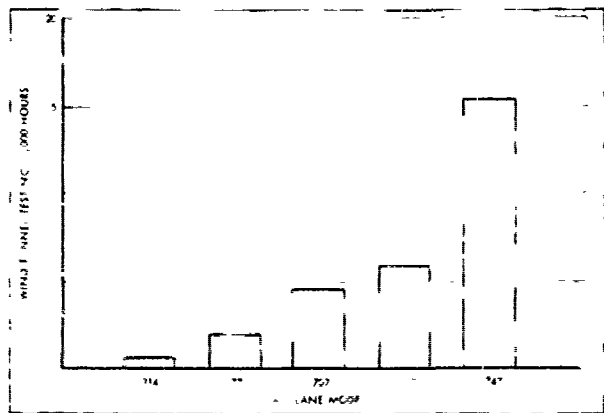


Figure 10: Wind Tunnel Development Hours

Where Figure 11 illustrates how the same basic airframe can be adapted to different missions, Figure 12 shows how some of the same basic components can be combined with other, different, components to make up new configurations. By changing the landing gear, a land plane can convert to a sea plane (or amphibian) with many components remaining practically unchanged. By extending the center wing span and by adding to the constant section fuselage length, the basic 1.2 million pound, 4-engine airplane can grow to a 2 million pound, 6-engine airplane. Identical landing gears and propulsion installations are used to support and power the fifty percent larger airplane.

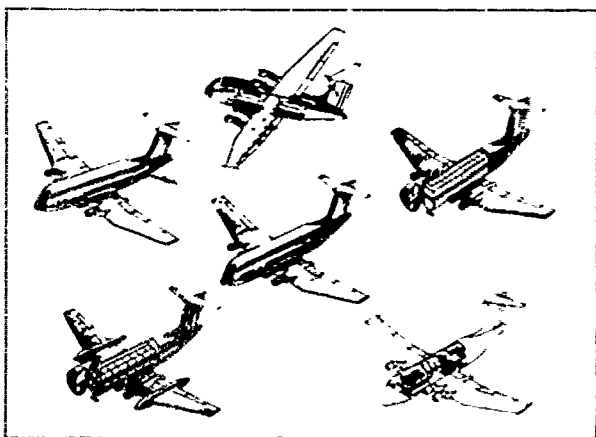


Figure 11: Mission Adaptability

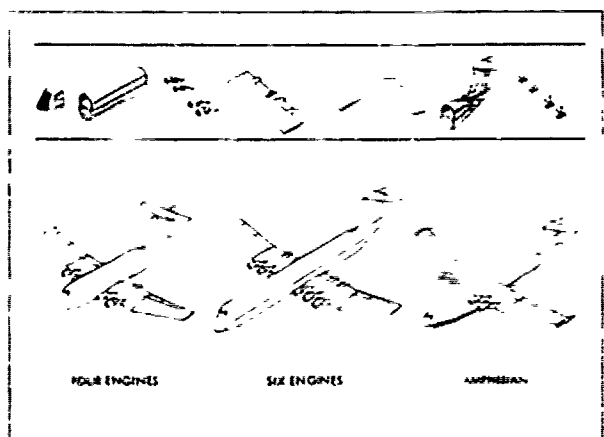


Figure 12: Component Adaptability

PRODUCTIVITY

In the preceding paragraphs, we have presented some of the reasons why moderate speeds should make it possible to keep the development and acquisition costs of new, large airplanes down. Low cost, however, constitutes only one half of the picture. The other half has to do with productivity, expressed, for example, by ton-miles per airplane per year in a range payload mission, or in ton-hours per year in an endurance mission, such as airborne alert. Payload capability and design cruise speed determine the productivity per flight-hour. Utilization enters as an additional factor in the determination of force size required to accomplish a specified task.

Figure 13 shows how airplane productivity has increased over the years, by a factor ten over the past twenty years alone. Increased productivity and lower operating costs have more than offset the higher development and acquisition costs of today's airplanes. The question of how the variation in cruise speed will affect the productivity of very large airplanes is answered in Figures 14 and 15.

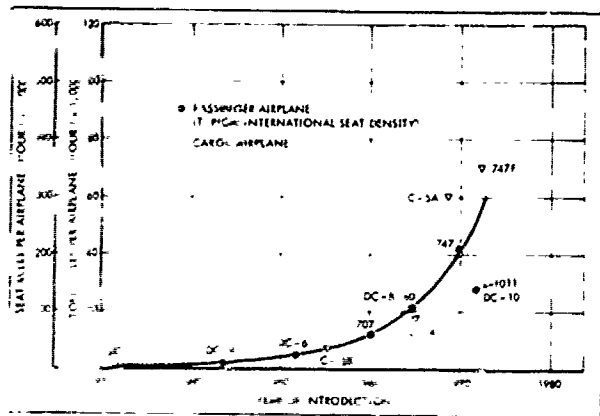


Figure 3: Growth of Airplane Productivity

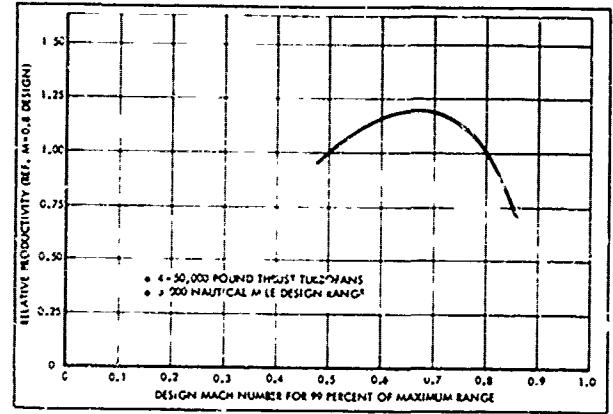


Figure 14: Military Cargo Mission

For the standard-range mission of Figure 14, a range of 3000 nautical miles, a payload density of 15 pounds per cubic foot, fixed length and identical engines (namely four 50,000 pound thrust turbofans) were assumed for all designs. With these assumptions, the airplane design has been optimized for each cruise Mach number. Moderate wing sweep, high wing loading and multiple slotted high lift devices were used at the high cruise Mach numbers. No wing sweep, low wing loading and no high lift devices characterize the designs for the lowest Mach numbers. Body volume is increasing with decreasing Mach number. A similar design optimization was followed for the endurance mission of Figure 15. In this case, the 3000 nautical mile range was replaced with a 20 hour endurance requirement. These curves show that productivity (ton mile/day) peaks at 0.6 to 0.7 Mach number, but that for endurance missions payload is still increasing at lower Mach numbers. The relative impact of the range-payload and the endurance missions affect the choice of a multi-mission design. These curves and other considerations, would imply that speeds in excess of 0.65 are required for military missions of both the payload and the endurance type.

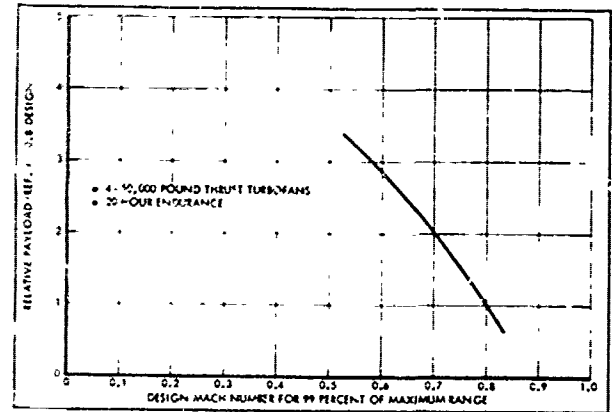


Figure 15: Endurance Mission

OPERATING COSTS

Operating costs of military systems are composed of personnel cost, petroleum-oil lubricant cost and depot maintenance. The very large airplane with high productivity minimize the number of aircraft required to do a given job. This has a major impact on operating cost because it reduces the personnel cost. The high productivity with a given propulsion installation implies a high ton-mile per pound of thrust, thus affecting the second operating cost factor favorably. Lastly, the reduced part count and simple components of the moderate-speed airplane should result in reduced maintenance manhours per flight hour. The actual data plotted in Figure 16 supports this judgment. This not only lowers operating costs, it also has the cumulative effect of increasing the average aircraft availability and utilization with a corresponding decrease in the number of airplanes required to produce the desired transportation effort.

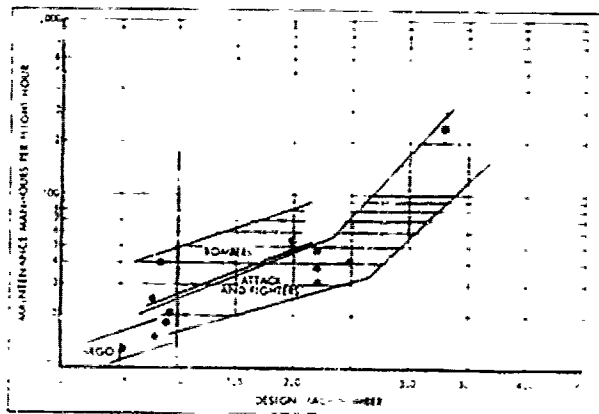


Figure 16: Effect of Speed on Maintenance

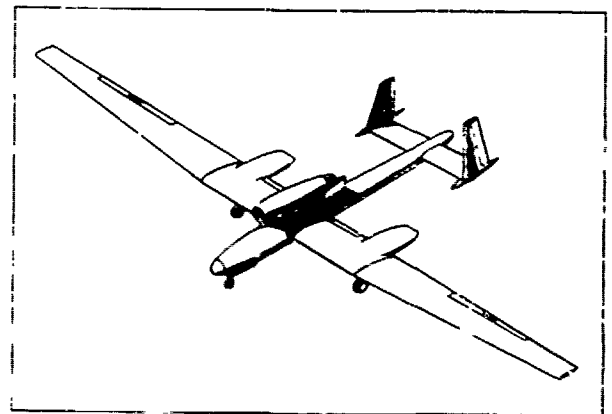


Figure 17: Compass Cope

APPLICATION OF PRINCIPLES

In the preceding sections low-cost concepts have been developed and explained. In this section, application of these concepts to executed and proposed airplanes is described.

The first example is Compass Cope, shown in Figure 17, a remotely piloted vehicle Boeing has built. While not a large airplane in the context of this discussion, it appears worth mentioning here because some of the concepts advocated for large airplanes have been applied to this vehicle. Cost was a paramount consideration on this program, and the search for low cost solutions led to the development of some of the concepts. Two design features are worth mentioning: the constant chord, unswept center wing and the U-tail.

The constant chord center wing was selected for the prototype to simplify wing construction and landing gear design. An additional consideration was the ease with which span and wing area could be modified later. It was found desirable near the end of the preliminary design phase to increase the gear tread by six feet. This was accomplished with no disturbance in the design effort by increasing the center wing span.

The U-tail was selected because it was possible to predict its characteristics analytically. A V-tail would have offered some operational advantages but was rejected because it would have entailed running blown nacelle wind tunnel tests. Selection of proven solutions kept the development cost low.

The next example, shown in Figure 18, deals with a study conducted for NASA. A carrier aircraft for the Space Shuttle Orbiter. The use of such a carrier aircraft is an alternate concept to equipping the shuttle orbiter with removable airbreathing engines for flight test and ferry missions. Cost will be the deciding factor in the ultimate concept selection. The effort to keep the carrier aircraft cost low is illustrated by the extensive use made of existing 747 parts off the production line. The center body portion is modified to permit attachment of the lift wing. A new straight center wing section is used to join the two fuselages and to suspend the orbiter. The center wing has neither leading edge nor trailing edge devices. The cost for two such carriers could be low enough to warrant serious consideration by NASA.

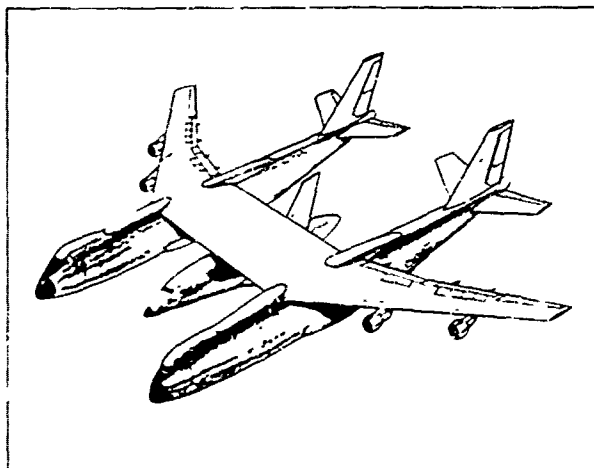


Figure 18: Space Shuttle Carrier

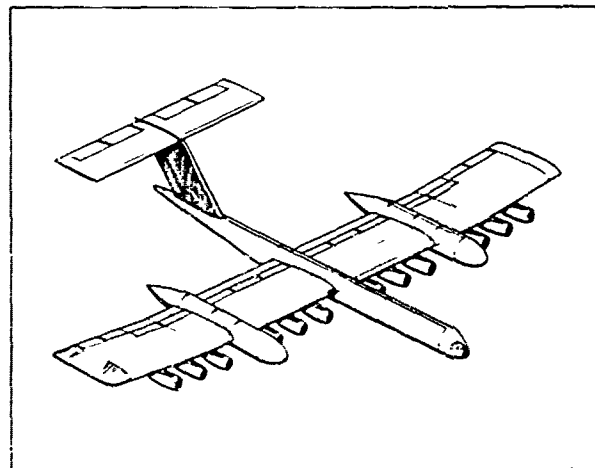


Figure 19: Large Resource Transport Airplane Concept

The resource carrier shown in Figure 19 emanated from a study started in early 1971. It was designed as a single-purpose dedicated airplane to transport crude oil, natural gas and valuable minerals from the source in the Canadian Arctic across the permafrost to a location where conventional means of transportation (pipeline, railroad or ship) would take over. The twelve-engine transport with a gross weight of 3.55 million pounds designed for a nominal range of 500 nautical miles carries a payload of 2.32 million pounds, a payload weight fraction of 65 percent. All the concepts proposed earlier in this paper were applied to this airplane.

- Straight wing with symmetrical structural box
- Identical engines, nacelles, and struts
- Identical landing gear and doors
- Identical diameter of fuselage and payload pods
- Unswept, symmetrical stabilizer
- No leading edge devices, simple flaps

Cost projections made at the time of the study indicated that a transportation system incorporating this airplane could deliver crude oil and liquid natural gas at prices competitive with arctic pipelines for oil and gas.

Very large aircraft such as the arctic resource carrier depend on the spanwise distribution of fuel, payload and landing gear for their structural efficiency. The straight wing becomes fundamental in this case since it accommodates the required number of landing gears in line at the optimal longitudinal position. The spanwise distribution of the landing gears on such very large aircraft leads to gear treads that cannot be accommodated on today's airport taxiways. Such very large aircraft are, therefore, initially limited to specialized tasks with specially designed landing fields.

The airplanes shown in the next figures are in the 1.2 million pound class and are compatible with existing ground facilities. The basic model of the airplane is the land-based missile carrier shown in Figure 20. It is characterized by a double deck fuselage, with weapons bays forward and aft of the wing box on the lower deck. The upper deck provides space for mission control center and crew accommodations.

The tanker shown in Figure 21 uses the same airframe. Used in conjunction with the missile carrier shown previously, long term airborne alert of strategic deterrent forces is possible. Rapid resupply of overseas forces with fuel is possible using a tanker with an offload capability of this size.

The ocean surveillance and sea control airplane of Figure 22 once again uses the same airframe. The weapons bays hold anti-ship missiles, torpedos and sonobuoys. The upper deck accommodates mission control, communications, electronic warfare and other mission related equipment and operators stations. The long range and long endurance of this airplane make possible dispatch into distant waters and extended loiter in operational areas.

The addition of a swingnose and the elimination of the bomb bays convert the basic airframe to a cargo airplane, as shown in Figure 23. The fuselage cross section with its 12-foot radius upper and lower lobes can accommodate 48 8 x 8 x 10 foot containers.

At a design range of 3000 nautical miles, the payload varies from 466,000 pounds (low density cargo, 289-foot fuselage length) to 528,000 pounds (high density cargo, 205-foot fuselage length).

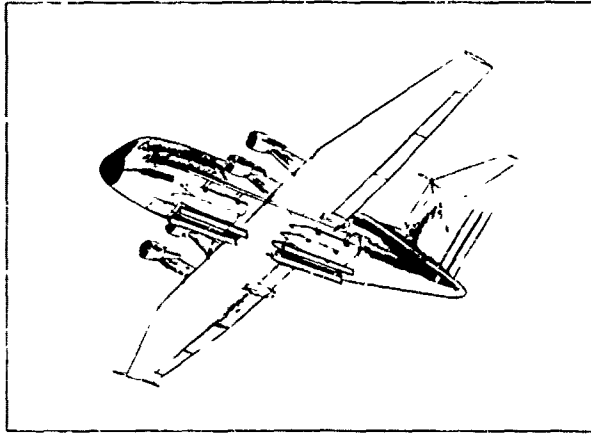


Figure 20: Missile Carrier

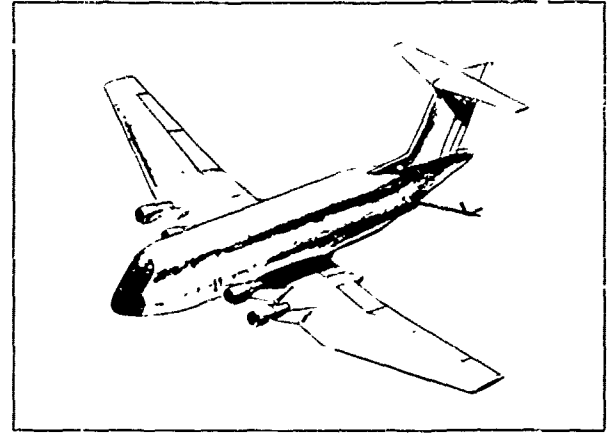


Figure 21: Aerial Tanker

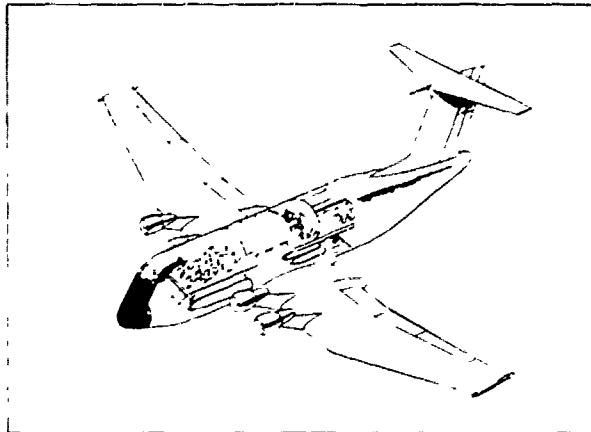


Figure 22: Sea Control Airplane

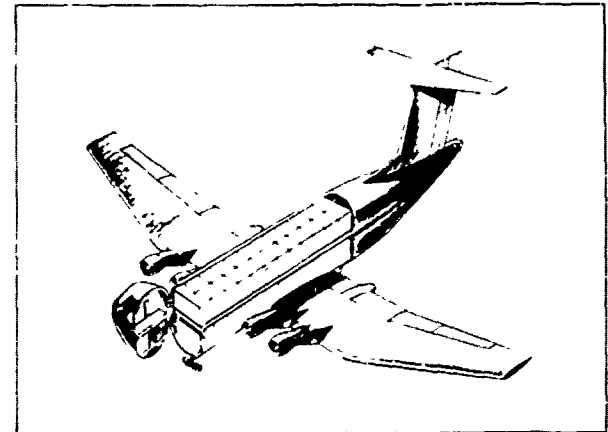


Figure 23: Cargo Airplane

Figure 24 shows a highly specialized liquid natural gas carrier. The payload, 674,000 pounds of liquid natural gas at cryogenic temperatures, is carried in the fuselage and in wing pods. Interaction between structural wing box and cryogenic tanks is thus avoided.

A variation of the landbased missile carrier of Figure 20 is the amphibian missile carrier shown in Figure 25. It uses the same wing, empennage, propulsion installation and cockpit as all the other airplanes, but for obvious reasons it requires a totally different fuselage. The amphibian missile carrier can use existing bases, but it has the advantage over landbased designs that it can loiter with an almost unlimited dispersal at minimum expenditure of fuel. Its capability to sit on any large body of water gives the system an extended post-attack rundown time. This makes a controlled response to an attack feasible, a desirable feature of a strategic deterrent system.

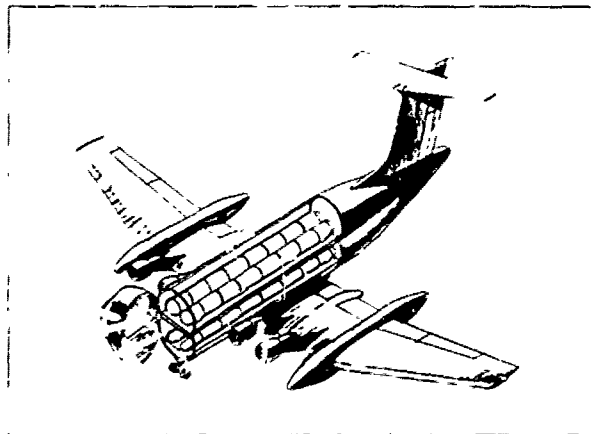


Figure 24: Liquid Natural Gas Carrier

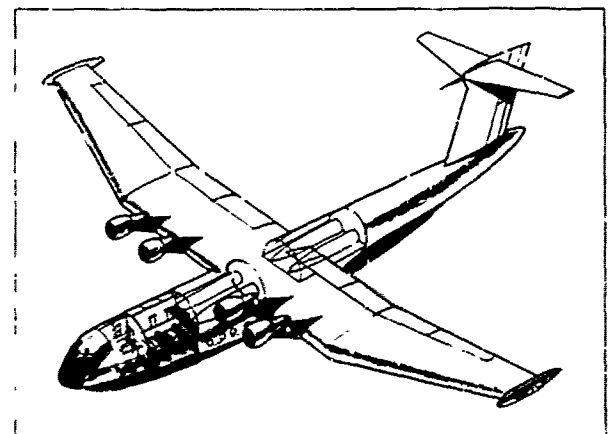


Figure 25: Amphibian Missile Carrier

CONCLUSIONS

Designing large airplanes for cruise speeds of approximately Mach = 0.65 is a prerequisite to low cost, very large airplanes. The large payload weight fraction achievable with a given installed thrust at the lower speeds makes the productivity of the airplane less sensitive to small variations in structural weight.

The basic aerodynamic flow characteristics at $M = 0.65$ permit configuration solutions combining simple component parts with high aerodynamic and structural efficiency. The total effect of simplicity is a major reduction of development, acquisition, and operating costs.

INTEGRATED, COMPUTER-AIDED DESIGN OF AIRCRAFT

by

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ABSTRACT

The design process for conceptual, preliminary, and detailed design of aircraft is discussed with emphasis on structural design. Problems with current procedures are identified and improvements possible with an optimum man-computer team using integrated, disciplinary computer programs are indicated. Progress toward this goal in aerospace and other industries is reviewed, including NASA investigations of the potential development of Integrated Programs for Aerospace-Vehicle Design (IPAD). The benefits expected from IPAD lead to the conclusion that increased use of the computer by a man-computer team that integrates all pertinent disciplines can create aircraft designs better, faster, and cheaper.

INTRODUCTION

Current requirements to produce technically superior aircraft at lower cost force the generation of optimized designs of greater technical depth in less time than in the past. Automation of the design process via computer-aided design systems that integrate all the pertinent disciplines can provide a solution to this problem. In this paper I will present a philosophical discussion of why we should automate the design process, how far we have come, and where we should be going. It is the result of several years of observation and participation in the automation of analysis and design of aerospace vehicles, particularly on structures.

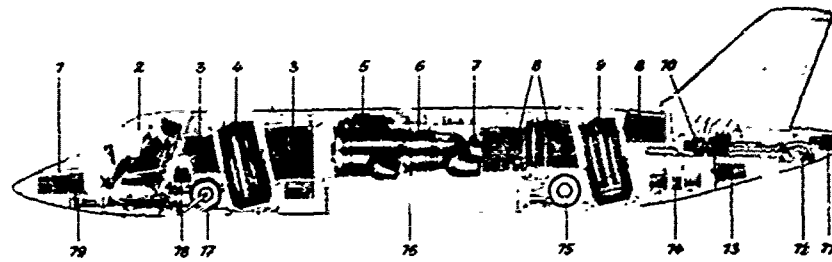
This paper starts with a discussion of the current design process and suggests the needs and payoffs from automation. Brief definitions of design automation and integrated computer-aided design are included. Then the nature of the design process is reviewed and followed by a description of progress already made toward automated design. Some additional steps toward greater automation, now in the planning stage, are described and their potential benefits are indicated. Finally, a few concluding remarks are presented.

The opinions presented in this paper are my own and do not necessarily reflect those of NASA. Because of my personal experience in structures, the discussion will be biased toward structural design, but the total aerospace vehicle design process will be considered.

THE NEED FOR AUTOMATION

Interest in computer-aided design is a logical consequence of the increasing cost and complexity of aerospace vehicles and systems and the related increase in the size of design staffs and in the complexity, cost, and time required for design, Reference 1. One of the factors contributing to both development and unit cost is the cost of design, which is increasing as illustrated in Figure 1. The cost of manpower and computer time required in a typical airplane company for designing one pound of aircraft structure is plotted against calendar year. It shows that we are paying about four times as much today for manpower as for computer time. Is this the best use of our available resources? I think not. Computers should have a larger share. If these trends continue for about a decade, expenditures for structural design will be equally divided between men and computers. However, I believe the trends will change and total design costs will grow less rapidly as we make more and better use of both men and computers. Hopefully, total costs could level off or turn downward in the future, but this is unlikely.

Another aspect of design that affects vehicle cost is illustrated in Figure 2 where planned and actual vehicle costs are plotted against time to design and manufacture a prototype. The vehicle cost increment is due to untimely engineering that produced results too late, caused out-of-sequence and repeated work, and resulted in unnecessary manufacturing changes. The cumulative effect is a magnified cost increment. Untimely engineering occurred because some phases of the design process did not go into adequate technical depth and/or because human limitations to deal with the volume and complexity of information involved were exceeded. A strong need exists to reduce human activity by providing some computer assistance on all routine functions and to use computerization to add greater technical depth and optimization in the early stages of design where basic concepts are selected.



- | | |
|-------------------------------|---------------------------------------|
| 1 ELECTRONICS COMPARTMENT | 11 DRAG CHUTE |
| 2 COCKPIT | 12 PITCH AND YAW CONTROL NOZZLES |
| 3 FRONT FUEL TANKS | 13 INTEGRATED HYDRAULIC COMPONENT |
| 4 FRONT LIFT ENGINE | 14 ELECTRONICS COMPARTMENT |
| 5 ACCESSORY DRIVE | 15 MAIN UNDERCARRIAGE |
| 6 LIFT-THRUST ENGINE | 16 LOAD COMPARTMENT |
| 7 SWIVELLING NOZZLES DRIVE | 17 NOSE UNDERCARRIAGE |
| 8 REAR FUEL TANKS | 18 AIR CONDITIONING AND OXYGEN SUPPLY |
| 9 REAR LIFT ENGINE | 19 PITCH CONTROL NOZZLES |
| 10 AUXILIARY POWER UNIT (APU) | |

Fig. 19 INBOARD PROFILE

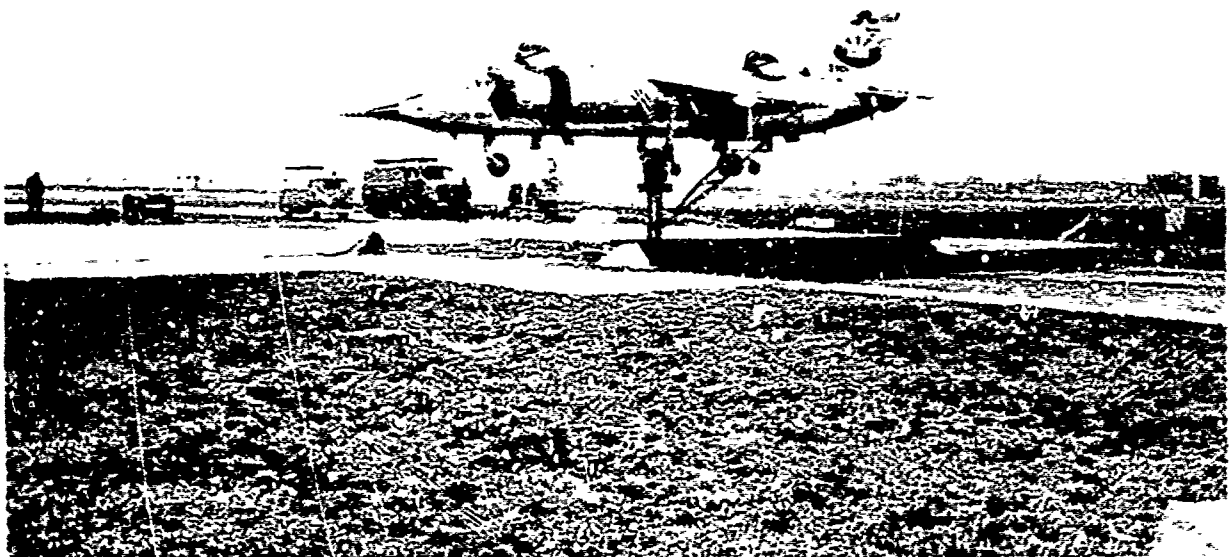


Fig. 20 AIRCRAFT MOUNTED ON PEDESTAL

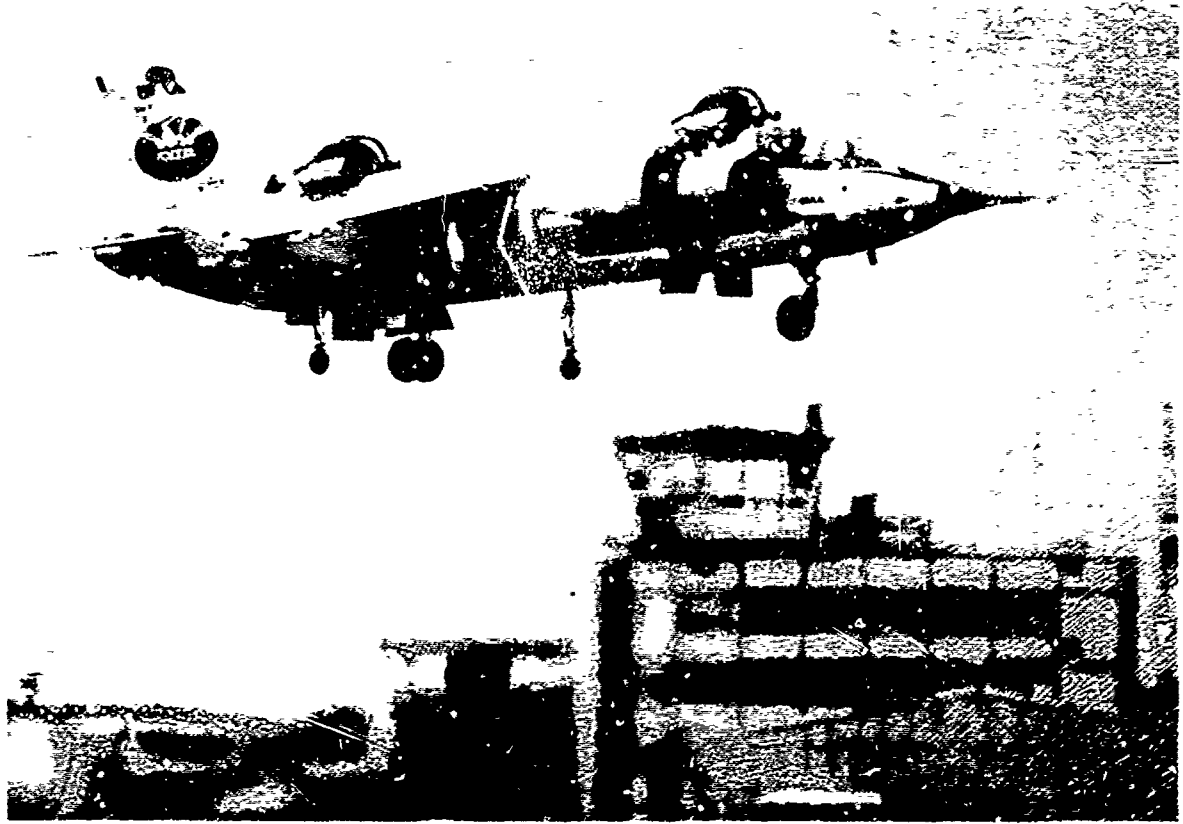


Fig. 21 HOVERING TEST

CONCEPT CCV ET SPECIFICATIONS

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RESUME

Le concept CCV tel qu'il est présenté actuellement consiste, au stade de la conception d'un nouvel avion, à tenir compte des possibilités offertes par quatre systèmes :

- stabilité artificielle ;
- répartition des charges en manoeuvre ;
- antiturbulence ;
- antifiottement.

En réalité, le concept CCV doit être compris dans un sens beaucoup plus large ; cette philosophie peut se résumer ainsi : profiter des derniers progrès technologiques de l'électronique (fiabilité et miniaturisation en particulier) et utiliser des types nouveaux de gouvernes de façon à satisfaire au mieux le compromis performance, qualités de vol, durée de vie, prix.

Une des conséquences de cette philosophie est l'abandon des exigences de stabilité "naturelle" de l'avion (stabilité statique et stabilité de structure, c'est-à-dire absence de flottement dans le domaine de vol).

Doit-on en conséquence créer un règlement nouveau pour les avions conçus selon les principes CCV ? Telle est la question à laquelle nous nous efforcerons de répondre.

SUMMARY

The C.C.V. concept and specifications

The C.C.V. concept, as defined at present, consists in taking account, at the design stage of a new aircraft, of the possibilities offered by four systems :

- static stability compensation,
- manoeuvre load control,
- active ride control,
- active flutter control.

Actually, the C.C.V. concept should be understood in a much wider context ; this philosophy could be summed up as : to take advantage of the most recent technological progress in electronics (e.g. reliability and miniaturization) and to make use of new types of control actuators in order to satisfy at best the compromise between performance, handling-qualities, lifetime, cost.

One of the consequences of this philosophy is the abandonment of the aircraft "natural" stability requirements (static stability and structure stability, i.e. absence of flutter in the flight range).

Should we, therefore, write new specifications for aircraft designed along the C.C.V. concept? That is the question that we shall endeavour to answer.

En gestation depuis quelques années, le concept CCV et son équivalent français, l'intégration des systèmes, quitteront bientôt le domaine des spéculations et de la recherche pour être réellement utilisés pour la conception des avions d'arme de la prochaine génération et en partie du moins pour la conception des futurs avions de transport.

Utilisés jusqu'à présent sur des avions purement expérimentaux, les systèmes CCV n'étaient pas assujettis à d'autres règles de sécurité que celles habituellement en vigueur sur ce type d'appareil où les risques de panne ou de défaillance sont compensés par des règles d'emploi et de maintenance particulières (pilotes d'essais entraînés, écoute radio par des spécialistes, bonne météo, guidage possible sur des terrains de secours, trafic aérien local neutralisable, enregistrement permanent des divers paramètres de fonctionnement et vérification des systèmes entre chaque vol, etc.).

Par contre, lorsque ces systèmes seront utilisés sur avions d'arme ou sur avions de transport, ils devront satisfaire à des règles précises permettant de s'assurer que le niveau de sécurité que l'on est en droit d'attendre sur les avions de cette génération n'en est pas diminué. C'est alors que se pose le problème de la réglementation applicable aux avions bâtis sur les principes CCV. Devrons-nous modifier les MIL SPEC 8785 B et 87300, l'AGARD 577, l'AvP 970, la FAR 25 ou le TSS 3 pour tenir compte de ces nouveaux systèmes ? Telle est la question à laquelle nous nous efforcerons ici de fournir une réponse après avoir dégagé les principes de base du CCV.

Bien souvent, le concept CCV est présenté comme l'utilisation de quatre systèmes principaux permettant de se libérer d'un certain nombre de contraintes au moment de la conception d'un nouvel avion. Ces quatre systèmes sont :

- a) la stabilité artificielle (aircraft static stability compensation) ;
- b) le système antiturbulence (active ride control) ;
- c) le système de réduction des charges en manoeuvre (manoeuvre load control) ;
- d) le système antifiottement (active flutter control).

Le système de stabilité artificielle permet d'une part de se libérer partiellement du problème de centrage au moment de la conception, d'autre part, de réduire la dimension des empannages et la traînée d'équilibrage.

Le système antiflottement autorise, comme le système de réduction des charges en manoeuvre, une diminution de la masse de structure.

Enfin, le système antiturbulence réduit la fatigue de la structure et améliore le confort de pilotage.

L'utilisation de ces quatre systèmes sur bombardier ou avion de transport permet d'obtenir des gains importants en ce qui concerne la masse au décollage et donc la puissance installée (gain de masse de structure, gain de traînée donc de masse de combustible). Par contre, le concept CCV appliqué aux avions de chasse conduit à l'utilisation de systèmes soit différents soit analogues mais conduisant à des gains pour des raisons différentes.

C'est ainsi que l'adjonction de gouvernes de portance directe et de force latérale permet d'accroître la manoeuvrabilité de l'avion en combat et rend les assiettes partiellement indépendantes de la trajectoire (amélioration de la plateforme de tir). La stabilité artificielle permet de réduire les surfaces d'empannage et de réduire les traînées d'équilibrage en manoeuvre, ce qui augmente le facteur de charge maximal équilibré. Associé à ce dernier système, le système antiflottement permet de dessiner l'avion lisse sans se soucier du problème des charges extérieures (l'adjonction de charges externes en général recule le centrage, avance le foyer et réduit la vitesse critique de flottement).

En prenant un peu de recul, la philosophie de base du CCV paraît en définitive être la suivante "réduire au mieux la masse au décollage et la puissance installée, c'est-à-dire réduire les coûts d'un projet tout en obtenant des performances données, en oubliant quelques préjugés ou habitudes qui coûtent généralement cher".

Autrement dit, concevoir un avion sur ce principe revient :

- à ne pas s'imposer a priori la stabilité naturelle (stabilité autour du centre de gravité, stabilité de la structure, c'est-à-dire non flottement à l'intérieur du domaine de vol) ;
- à créer des gouvernes nouvelles pour répondre à des besoins nouveaux (antiflottement, portance directe, force latérale, répartition des charges en manoeuvre, etc.) ou des gouverner dont les défauts rédhibitoires peuvent être compensés par le système de stabilité artificielle (gouverne canard par exemple) ;
- à accepter l'utilisation de chaînes de commandes électriques et des commandes pilote nouvelles (micro-manipulateur) ;
- à présenter au pilote des informations nouvelles.

Nous pouvons même ajouter à cette liste le multiplexage des informations, c'est-à-dire la transmission des informations par des barres omnibus au lieu de circuits spécialisés.

Quels sont les points pour lesquels les problèmes de sécurité vont alors se poser ?

Le premier point, le plus évident, est le problème de fiabilité des systèmes. Le deuxième est celui des qualités de vol qui sont influencées par l'apparition de gouvernes nouvelles, de commandes pilote nouvelles et de nouveaux systèmes de présentation des informations.

Regardons tout d'abord le problème de la fiabilité des systèmes.

En ce qui concerne les chaînes de commandes électriques, nous nous trouvons aujourd'hui dans une position analogue à celle où nous étions il y a une vingtaine d'années au moment où l'on envisageait de ne plus lier mécaniquement le manche et le palonnier aux gouvernes et à compter uniquement sur une transmission hydraulique. Il faut néanmoins reconnaître que nous sommes maintenant dans une situation beaucoup plus favorable pour sauter le pas car nous disposons de bases bien plus solides qu'alors en ce qui concerne les études de fiabilité. Par ailleurs, les règlements déjà modifiés pour pouvoir accepter la transmission hydraulique pure pourront être facilement adaptés à la transmission électrique. Le TSS 3 et les MIL SPEC 8785 B et 83300 qui reposent sur les mêmes principes de base ne faisant pas d'hypothèse a priori sur la fiabilité des systèmes n'ont pas à être modifiés. Par contre, des méthodes de démonstration de conformité devront être mises au point de façon à obtenir une démonstration satisfaisante de la fiabilité du système. Rappelons une fois de plus qu'aucune démonstration globale de fonctionnement réel en vol ne peut fournir la réponse. Il s'agit en effet de démontrer des probabilités de panne de l'ordre de 10^{-6} à 10^{-7} par heure de vol. Or la démonstration d'une probabilité de panne inférieure à 10^{-n} par heure avec un niveau de confiance de 0,9 exige un fonctionnement pendant $2,3 \cdot 10^n$ heures sans rencontrer la panne, de $3,9 \cdot 10^n$ heures en rencontrant une fois seulement la panne, de $5,3 \cdot 10^n$ heures en rencontrant deux fois seulement la panne, etc. Il est donc strictement impossible de démontrer directement 10^{-6} ou 10^{-7} . On ne peut atteindre expérimentalement que des probabilités de l'ordre de 10^{-2} à 10^{-3} par heure. En conséquence, on ne peut estimer la fiabilité du système complet que par le calcul fondé sur la redondance d'éléments dont la probabilité de panne de l'ordre de 10^{-2} est démontrée expérimentalement. Encore faut-il prendre garde à ce que la redondance des systèmes respecte l'indépendance des éléments ; or deux éléments issus d'une même chaîne de fabrication, soumis à la même ambiance peuvent-ils être considérés comme réellement indépendants ? L'expérience a montré qu'il n'en était rien et que la probabilité de la panne simultanée de deux éléments est nettement supérieure au carré de la probabilité de la simple panne (par panne simultanée, nous entendons panne de deux éléments au cours d'un même vol). Les progrès remarquables de la miniaturisation en électronique autorisent une multiplication des chaînes, circuits et systèmes beaucoup plus importante que celle concevable actuellement sur les systèmes mécaniques et hydrauliques ; mais la solution n'est peut-être pas dans la redondance de nombreux systèmes identiques mais plutôt dans l'utilisation de plusieurs chaînes parallèles remplissant la même fonction (ou pour quelques vnes remplissant la même fonction mais simplifiée), construites sur des principes, des schémas et des technologies différentes et installées dans des parties différentes de l'avion. Cette méthode a pour but de rendre les chaînes réellement indépendantes et par ailleurs de diminuer la vulnérabilité des avions d'arme.

Nous venons de rappeler que la démonstration de fiabilité ne pouvait être obtenue directement

par des essais en vol des prototypes ou avions de présérie. N'en déduisons cependant pas que les essais en vol soient inutiles : après 1000 ou 2000 heures de vol d'essais sans incident, il est impossible de conclure directement que la fiabilité des systèmes est suffisante ; c'est par contre la seule méthode pour vérifier le fonctionnement en ambiance réelle de chacun des systèmes et d'en évaluer la probabilité de panne. Mais, seule l'analyse théorique de la façon dont les systèmes élémentaires composent le système complet permet de conclure quant à la fiabilité de l'ensemble à partir des estimations des probabilités de pannes élémentaires obtenues au banc d'essai et confirmées par les essais en vol.

Bien entendu, ces méthodes d'analyse de la fiabilité n'ont pas atteint le concept CCV pour être étudiées et développées. En particulier, la certification de Concorde repose en grande partie sur ces méthodes ; rappelons par exemple que Concorde vole en supersonique à un centrage qui rendrait l'avion instable en vol subsonique. Il est bien évident que pour accepter qu'un avion vole à un centrage pour lequel l'approche et l'atterrissage sont impossibles, il faut démontrer que la probabilité de panne du système de transfert de combustible est raisonnablement faible (la probabilité d'une seule panne interdisant le retour au centrage subsonique au cours de la vie de l'ensemble des avions en service doit être suffisamment faible pour que cet événement puisse être considéré comme improbable).

Il n'en reste pas moins nécessaire d'améliorer les méthodes d'analyse sur de nombreux points : recherche systématique des cas critiques, réduction des temps de calcul, estimation des probabilités élémentaires (calcul, essais en simulation, essais en vol) etc., toutes ces recherches étant destinées à bâtir des méthodes plus sûres de démonstration de conformité au règlement.

Avant de passer aux problèmes posés par les qualités de vol, une dernière remarque s'impose en ce qui concerne le niveau de probabilité de panne à exiger des différents systèmes afin d'obtenir un niveau de sécurité global acceptable pour les avions civils et militaires et une probabilité raisonnable de réussite de la mission pour les avions militaires.

Étudions tout d'abord le cas des avions militaires.

Il est parfaitement admis actuellement de perdre quelques appareils d'une flotte par suite d'incidents interdisant la poursuite de la mission (panne de réacteur sur mono-moteur, panne totale du circuit hydraulique des servo-commandes, etc.) dans la mesure toutefois où le pilote dispose de moyens lui laissant une chance raisonnable de ne pas être tué dans la catastrophe (c'est ainsi que les pilotes de certains chasseurs monoréacteurs ont la consigne de s'éjecter en cas de panne de moteur).

Le choix du niveau de fiabilité du système CCV peut alors se faire de la façon suivante.

Un système CCV "de principe", c'est-à-dire comportant les circuits minimaux réalisant les fonctions CCV, permet, à performances données, de réduire la masse de structure et la quantité de carburant nécessaire à la mission ; il en résulte un gain global de $x\%$ sur le prix d'achat et le coût d'exploitation d'une flotte d'avions (par coût d'exploitation, nous entendons les dépenses nécessaires au fonctionnement de chaque avion pendant toute la durée d'exploitation de ce type d'appareil). Si nous améliorons la fiabilité du système CCV en augmentant la redondance des divers circuits, il est bien évident que nous augmentons le prix d'achat de chaque avion ainsi que son coût d'exploitation (augmentation des heures de maintenance des systèmes). Ainsi le gain par rapport à l'avion classique diminue alors que la fiabilité des systèmes augmente*.

Par ailleurs, à toute probabilité p (par heure) de panne du système CCV conduisant à la perte de l'avion correspondent des probabilités Q_k d'avoir entre 0 et k pertes d'appareils pour une flotte de N avions exécutant chacun n heures de vol. En prenant $Q_k = 0,99$ on a une bonne estimation du nombre k de pertes par panne du système CCV que l'on peut raisonnablement escompter observer pendant la durée d'exploitation de la flotte (on a une chance sur cent seulement d'observer un nombre réel d'accidents supérieur à k).

Si p est la probabilité de panne par heure, la probabilité de non accident au cours des n heures de vol de l'avion est $I_1 = (1-p)^n$ et la probabilité d'accident $P_1 = 1 - I_1$. Ainsi la probabilité Q_k d'avoir entre 0 et k accidents pour N avions de la flotte est :

$$(1) \quad Q_k = \sum_{m=0}^k C_N^m I_1^{N-m} P_1^m$$

Lorsque le produit $N I_1$ est suffisamment grand, une bonne approximation du nombre k correspondant à une valeur donnée de Q_k est donnée par :

$$(2) \quad k = \lambda \sqrt{N I_1 P_1} + N I_1$$

λ étant déterminé par :

$$Q_k = \int_0^\lambda \frac{e^{-x^2}}{\sqrt{2\pi}} dx \quad (\text{loi normale})$$

(pour $Q_k = 0,99$; $\lambda = 2,3264$ et pour $Q_k = 0,5$; $\lambda = 0$)

Pour fixer les ordres de grandeurs, regardons la loi $k(p)$ correspondant à $Q_k = 0,99$ pour une flotte de 1000 avions devant effectuer chacun 5000 heures de vol (le calcul a été effectué avec la formule 1 pour $p < 10^{-6}$).

* - Il faut prendre garde en effectuant ce bilan à ne pas oublier les améliorations du système d'éjection qui peuvent se révéler nécessaires (siège zéro zéro par exemple qui n'est pas indispensable sur un multimoteur classique).

$Q_k = 0,99$									
p	10^{-4}	$5 \cdot 10^{-5}$	10^{-5}	$5 \cdot 10^{-6}$	10^{-6}	$5 \cdot 10^{-7}$	10^{-7}	$5 \cdot 10^{-8}$	10^{-8}
k	430	252	65	37	11	7	3	2	1

Pour voir l'influence de Q_k sur ce résultat, donnons également les lois $k(p)$ pour $Q_k = 0,5$ (une chance sur deux d'avoir plus de k pertes) et $Q_k = 0,999$ (une chance sur mille d'avoir plus de k pertes) (pour $Q_k = 0,999$; $\lambda = 3,0902$).

$Q_k = 0,5$									
p	10^{-4}	$5 \cdot 10^{-5}$	10^{-5}	$5 \cdot 10^{-6}$	10^{-6}	$5 \cdot 10^{-7}$	10^{-7}	$5 \cdot 10^{-8}$	10^{-8}
k	394	222	49	25	5	2	0	0	0

$Q_k = 0,999$									
p	10^{-4}	$5 \cdot 10^{-5}$	10^{-5}	$5 \cdot 10^{-6}$	10^{-6}	$5 \cdot 10^{-7}$	10^{-7}	$5 \cdot 10^{-8}$	10^{-8}
k	442	262	70	40	13	5	4	3	2

Il ressort de ces chiffres que l'objectif de fiabilité pour le système CCV se situe raisonnablement au voisinage de 10^{-6} : 1 % de perte est acceptable par les équipages sur le plan psychologique dans la mesure où la probabilité de perte de l'équipage lui-même est beaucoup plus faible (d'où le problème de la fiabilité du système d'éjectica) ; bien entendu, cet objectif n'est valable que si le gain global α dû au système CCV est supérieur à 1 %.

Or, il faut reconnaître que pour atteindre une probabilité de panne de 10^{-6} par heure, il est nécessaire de tripler les systèmes* (nous avons vu plus haut que l'on peut tout juste atteindre expérimentalement des probabilités de systèmes élémentaires de l'ordre de 10^{-5} ; pour démontrer que p est inférieur à 10^{-6} , il faut donc plus de deux systèmes indépendants).

Il n'est donc pas évident que le coût de système CCV ainsi conçu autorise un gain global supérieur à 1 %.

En ce qui concerne les avions civils, le problème se pose de façon différente car il s'agit dans ce cas d'assurer la sécurité de passagers payants n'ayant pas la possibilité d'évacuer l'avion en cas de panne totale du système. Par ailleurs, un avion civil est destiné à accomplir un nombre d'heures de vol beaucoup plus important qu'un avion militaire, de l'ordre de 30 000 au lieu de 5000.

Actuellement, l'objectif de fiabilité global imposé par les règlements civils est de l'ordre de 10^{-7} par heure (pour les avions de la génération en service aujourd'hui sur les lignes aériennes, la probabilité de panne catastrophique est plutôt de l'ordre de 10^{-5} par heure). Voyons à combien de catastrophes cela conduit pour une flotte de 1000 avions.

Pour $n = 30000$ heures ; $N = 1000$; $p = 10^{-6}$; on obtient

pour $Q_k = 0,5$ $k = 30$
 $Q_k = 0,99$ $k = 42$
 $Q_k = 0,999$ $k = 50$

Pour $n = 30000$ heures ; $N = 1000$; $p = 10^{-7}$; on obtient

pour $Q_k = 0,5$ $k = 3$
 $Q_k = 0,99$ $k = 7$
 $Q_k = 0,999$ $k = 9$

* - Les circuits hydrauliques ne sont généralement que doublés mais il faut reconnaître que les constructeurs ont accumulé un grand nombre d'expériences en vol permettant de mieux qualifier la fiabilité de ces systèmes et que les règles de l'art assurant la constance de ce niveau de fiabilité sont maintenant solidement établies.

On voit que l'objectif à 10^{-7} représente encore une perte comptée de 7 avions sur 1000 en service. Il est cependant difficile pour l'instant d'exiger une fiabilité des systèmes CCV conduisant à une probabilité élémentaire inférieure à 10^{-7} : il n'y a pas de raison de demander plus à ce système qu'aux systèmes classiques. Notre propos est tout au plus de montrer que l'objectif à 10^{-7} raisonnable sur le plan faisabilité pour la décennie à venir n'est qu'une étape dans la recherche de la sécurité (l'accroissement de sécurité doit d'ailleurs dans une première étape venir plus de l'amélioration de la navigation et du pilotage au décollage, en approche et à l'atterrissage, que de l'amélioration de la fiabilité des systèmes).

Nous venons de voir quelle doit être la probabilité de panne conduisant à une catastrophe. Cet objectif est valable pour le système de commande électrique ou le système de stabilité artificielle, mais la panne de chacun des systèmes CCV n'a pas toujours des conséquences catastrophiques.

Il est évident que les pannes du système antiturbulence et du système de répartition des charges en manœuvre dans la mesure où elles ne mettent pas en cause la sécurité immédiate par un braquage intempestif de gouverne rendant l'avion incontrôlable n'ont pas de conséquences graves puisque ces dispositifs ont pour seul but d'améliorer le confort ou la durée de vie en fatigue de l'avion. Des chaînes simples sont donc parfaitement acceptables du seul point de vue sécurité (par contre, des considérations de confort en exploitation ou de réussite de mission opérationnelle peuvent amener à améliorer la fiabilité du système antiturbulence).

En ce qui concerne le système antiflottement, on serait tenté de conclure rapidement à exiger une fiabilité du même ordre que celle des systèmes de commandes électriques ou de stabilité artificielle.

Il faut en réalité tenir compte des conditions d'emploi d'un tel système. Sur un avion de transport civil, le système antiflottement peut être utilisé pour repousser la vitesse critique plus loin dans le domaine périphérique, autrement dit autoriser le vol avec une marge réduite par rapport à la vitesse critique de flottement naturel. Dans ces conditions, une catastrophe ne peut se produire que s'il y a pénétration dans le domaine périphérique au-delà de la vitesse critique naturelle et panne simultanée du système antiflottement pendant l'excursion. La probabilité de catastrophe est donc le produit de la probabilité d'excursion et de la probabilité de panne du système. Compte tenu de la faible probabilité d'excursion et du faible temps passé au-delà de la vitesse critique naturelle, la probabilité de panne du système peut être relativement élevée (de l'ordre de 10^{-3} à 10^{-4}). Ce problème est très voisin de celui rencontré pour tous les dispositifs de sécurité, destinés à fonctionner dans le domaine périphérique comme par exemple le "stick shaker" ou le "stick pusher".

Par contre, le système antiflottement peut être amené sur avion d'arme à fonctionner à l'intérieur même du domaine autorisé. La fiabilité exigée dépendra cette fois des conséquences de la panne : si cette conséquence en est le déclenchement d'un flottement explosif, il est bien évident que la probabilité de panne doit être de l'ordre de celle exigée pour le système de commandes électriques ou le système de stabilité artificielle ; par contre, si le dispositif est destiné à combattre un flottement dû à la présence de charges extérieures, le problème peut se poser de façon différente ; ce type de flottement est en général relativement mou et peut laisser suffisamment de temps à un dispositif de sécurité pour larguer les charges extérieures. Dans ces conditions, là encore, une fiabilité réduite du système peut être acceptée.

Venons en maintenant au deuxième point intéressant la sécurité. Comment le concept CCV peut-il réagir sur les exigences de qualités de vol ou plus exactement sur les exigences de pilotabilité.

L'objectif général de toute exigence de pilotabilité est donnée par le TSS 3 :

"l'avion doit avoir des caractéristiques de pilotabilité suffisamment bonnes pour que l'exécution de chaque sous-phase et des manœuvres qui s'y rapportent ne soit pas trop difficile et fatigante pour l'équipage compte tenu de la durée de la sous-phase et de la probabilité de l'état de l'avion et de l'état de l'atmosphère. Autrement dit, l'ensemble des activités physiques et mentales nécessaires à l'exécution de chaque sous-phase ne doit pas entraîner de fatigue excessive pour l'équipage de façon à limiter les risques d'erreur de jugement et les risques de fausses manœuvres".

En ce qui concerne les commandes, le TSS 3 précise que "toute manipulation de commandes, conforme au manuel de vol, doit pouvoir se faire sans gêne excessive pour l'équipage. En particulier, les efforts entraînés par la manipulation autorisée des commandes ne doivent pas être trop grands compte tenu de l'emplacement, de la forme et des dimensions des commandes ainsi que de la durée d'application de ces efforts. Cela s'applique aussi aux efforts apparaissant sur les commandes à la suite de la manipulation d'un sélecteur lors d'un changement de configuration affichée."

Ces principes de base étant dégagés, deux méthodes de démonstration de conformité sont alors proposées :

- la méthode classique que l'on retrouve de façon sensiblement équivalente dans tous les règlements (les exigences diffèrent quant à leur sévérité suivant qu'elles proviennent d'une spécification militaire ou d'un règlement de certification civile ; en effet, un règlement civil a pour seul objectif la sécurité des personnes transportées ou survolées alors qu'une spécification militaire a pour objectif non seulement la sécurité mais également la bonne efficacité de la mission) ;
- une méthode basée sur l'évaluation de la charge de travail par une échelle du type Cooper-Harper.

Regardons tout d'abord comment la méthode classique peut s'appliquer aux avions bâtis sur les principes CCV.

Quatre types de spécifications de pilotabilité sont exigés :

- a) des spécifications de stabilité ;
- b) des spécifications concernant la réponse de l'avion aux commandes ;
- c) des spécifications relatives aux efforts aux commandes (efforts maximaux et possibilité d'annulation) ;
- d) des spécifications concernant les effets des changements de configuration sur le comportement général

de l'avion (trajectoire, attitude, efforts aux commandes).

En ce qui concerne les exigences de stabilité, il n'y a aucun problème à les appliquer à un avion CCV. La seule difficulté pourrait provenir de la justification des exigences de stabilité elles-mêmes dans le cas des phases de vol où la vitesse évolue relativement rapidement ; ce cas est prévu dans le TSS 3 mais les méthodes de démonstration de conformité ne sont pas encore au point ; ce problème n'étant pas spécifiquement CCV, nous le traiterons pas ici (il est traité en particulier au sein du "handling quality committee").

De même, les spécifications concernant l'effet des changements de configuration s'appliquent sans difficulté au CCV. Il est à prévoir d'ailleurs qu'il sera relativement plus facile de satisfaire ces exigences dans le cas du CCV ; la modification des positions d'équilibre des gouvernes lors de la sortie du train, des volets, des becs, des aérofreins, etc. peut être plus facilement modulée en tenant compte des conditions de vol (vitesse, altitude, masse et centrage) du fait que les gouvernes sont commandées électriquement ; seul se pose le problème de fiabilité de la conjugaison.

Par contre, les lois d'efforts sur les micromanipulateurs devront faire l'objet d'études pour tenir compte, comme nous l'avons précisé plus haut, des dimensions de la forme et de l'emplacement de ces commandes. Des études au simulateur et en vol doivent permettre de résoudre facilement ce problème.

En définitive, c'est dans le domaine de la réponse de l'avion aux commandes que les problèmes les plus délicats vont se poser dans la mesure toutefois où le pilote disposera de commandes particulières pour actionner des gouvernes non classiques comme la gouverne de force latérale et la gouverne de portance directe. Bien entendu, si le pilote dispose des commandes classiques de roulis, lacet, tangage (même sous forme d'un micromanipulateur reproduisant les fonctions du manche et du palonnier), les exigences classiques s'appliquent telles quelles car le pilote n'a pas à savoir si ses ordres sont directement transmis à des gouvernes classiques ou si son action sur la commande provoque le braquage conjugué de plusieurs gouvernes ; c'est bien d'ailleurs le cas actuellement pour la commande de gauchissement qui peut actionner avec des taux de conjugaison différents suivant les conditions de vol, divers élévons et spoilers.

Il est cependant possible d'envisager deux commandes particulières pour agir sur les gouvernes de force latérale et de portance directe, ou plus exactement pour créer une force latérale et une force de portance permettant d'agir sur la trajectoire sans modifier l'incidence et le dérapage. Le pilote abandonne alors le micromanipulateur et le pilote automatique transparent maintient le dérapage nul, les ailes "horizontales" et l'incidence à la valeur recommandée pour cette phase de vol (approche, attaque au sol, etc.). Le pilote agit sur un deuxième micromanipulateur dont les déplacements "verticaux" et "latéraux" font varier portance et force latérale et permettent d'agir sur la trajectoire. C'est dans ce cas que doivent impérativement être fournies au pilote des informations concernant sa trajectoire, par exemple la trace à l'infini du vecteur vitesse ; le pilote en effet ne dispose plus du retour classique d'information fournie par les variations d'assiettes lui permettant de doser ses actions sur les commandes.

Il est bien évident qu'un tel mode de pilotage n'a pas été prévu par les différents règlements et que des spécifications nouvelles devront être établies pour juger de la qualité des réponses de l'avion à ces nouvelles commandes.

Deux méthodes sont alors applicables : ou bien établir ces nouvelles spécifications par des essais au simulateur et en vol sur avions expérimentaux ou bien appliquer directement sur l'avion à certifier les principes du TSS 3 en évaluant la charge de travail du pilote au cours des phases où le nouveau mode de pilotage est utilisé. Etant donné que les formes, les dimensions et l'emplacement du nouveau système de commandes, la nature même des nouvelles informations à fournir au pilote, sont loin d'être déjà figées, il nous semble plus sage d'appliquer la seconde méthode, la première ne pouvant intervenir que lorsqu'un certain nombre de règles générales d'emploi auront été trouvées par l'expérimentation des premières réalisations. Notons d'ailleurs que la première méthode consiste à déterminer un certain nombre de critères généraux fondés sur une expérimentation utilisant l'évaluation des charges de travail par l'échelle de Cooper-Harper.

En conclusion, le concept CCV pose deux types de problèmes en ce qui concerne les spécifications de sécurité :

- a) la démonstration de la fiabilité des systèmes : les méthodes d'estimation des probabilités globales de panne doivent être perfectionnées ;
- b) l'établissement de nouveaux critères de qualité de vol pour les avions équipés de commandes spéciales "de trajectoire" (force latérale et portance directe).

Par ailleurs, l'ensemble des critères classiques de qualité de vol s'appliquant sans modification aux avions bâtis suivant le concept CCV.

INTRODUCTION OF CCV TECHNOLOGY INTO AIRPLANE DESIGN

by

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1.0 INTRODUCTION

The airplane designer reaches for all the tools at his command when starting a new design. In the past the fundamental aeronautical sciences of aerodynamics, propulsion and structures have been his principal technological disciplines. More recently the development of modern control system technology has offered capability to extend the limits of aerodynamic and structural technology, and achieve superior airplane performance. Application of active control systems to improve airplane performance has become known as CCV (Control Configured Vehicle) technology, due to the leadership of the U.S. Air Force Flight Dynamics Laboratory, where the terminology was conceived, and which has sponsored most of the programs described herein.

2.0 THE CCV DESIGN APPROACH

The traditional airplane design process produces a configuration definition essentially as shown in Figure 1. Mission performance requirements such as takeoff, landing, payload/range, speed versus altitude, and endurance are assessed as the first step in establishing a new airplane configuration. Trade studies of these factors are performed and constraints applied to define a parametric airplane. This includes definition of the wing area, maximum weight, minimum weight, thrust, volume of the vehicle, etc. From this point, the propulsion, aerodynamic, and structural designs proceed, combining into a first configuration which is assessed for performance. This process is iterated several times to define a vehicle which meets all the specified mission performance requirements and which satisfies a minimum weight or minimum cost criterion.

When the configuration is defined, the traditional approach then moves to system design. At this point, flight control design usually begins. If flight control is not considered from the outset in a design, the result can be an airplane with handling qualities which are somewhere between acceptable and unacceptable, rather than optimum. The airplane may have none of the favorable structural/aerodynamic interactions which are possible with modern control systems.

In contrast, the CCV design approach shown in Figure 2 capitalizes on the potential of considering advanced flight control concepts during the initial parametric trades. The ability to perform various advanced control functions can influence formulation of mission requirements. Using the traditional approach the aircraft user is somewhat constrained to small improvements over previous designs, whereas the CCV approach may allow a more significant improvement in aircraft capability.

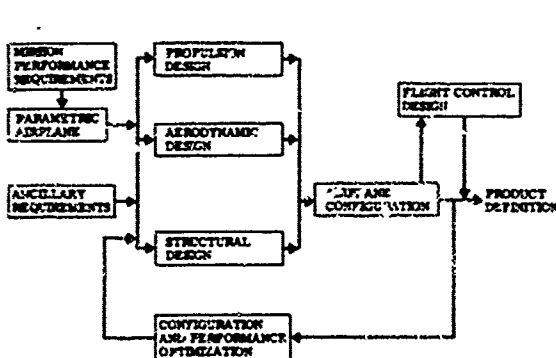


FIGURE 1. TRADITIONAL AIRPLANE DESIGN CYCLE

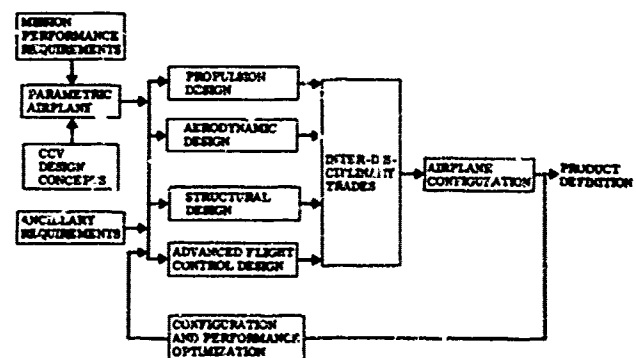


FIGURE 2. CCV AIRPLANE DESIGN CYCLE

Formulation of the parametric airplane can also be altered by the CCV approach, since traditional empirical design procedures no longer necessarily apply. For example, wing loading and/or aspect ratio for a specific load capability can be significantly changed by incorporating CCV concepts.

Flight control design is included on an equal basis with the other major technologies of propulsion, aerodynamics and structure. The large vertical block (Figure 2) emphasizes the interdisciplinary trades between these four technologies which are possible and beneficial in defining the final aircraft configuration. This process is iterated until the best possible configuration is achieved.

Analytical CCV studies indicate that the most significant performance improvements are achieved from six control functions:

- Augmented Stability (AS)
- Gust Load Alleviation (GLA)
- Maneuver Load Control (MLC)
- Fatigue Reduction (FR)
- Ride Control (RC)
- Flutter Mode Control (FMC)

Augmented Stability is a technique for eliminating the requirement for inherent aircraft static and dynamic stability and augmenting the stability with an active control system to a level that provides desirable handling qualities. This approach permits a smaller empennage, in that the empennage is sized to provide only the trim plus maneuver requirement. AS provides better control response, which improves maneuvering performance. Relaxed Static Stability (RSS) is another name which has been applied to AS.

Gust Load Alleviation is a technique for reducing airframe peak transient loads resulting from gust disturbances. It encompasses control of rigid body and/or structural flexibility components of the airplane gust response.

Maneuver Load Control is a method for redistributing wing lift during maneuvering flight. Incremental stresses may be reduced by deflecting wing control surfaces symmetrically in response to load factor commands in a manner that shifts the wing center of lift inboard, thus reducing wing root bending moments.

Fatigue Reduction is a technique for reducing fatigue damage rate by using active controls to reduce the amplitude and/or number of transient bending cycles to which the structure is subjected during turbulence.

Ride Control is a technique for improving crew and passenger ride comfort by reducing objectionable rigid body and structural vibrations through control surface deflections.

Flutter Mode Control is a technique for actively damping flutter modes using aerodynamic surfaces, providing potential weight savings and/or extending flutter placard speeds.

It is significant to note that the application of these concepts differs depending on the mission involved. For a fighter aircraft, AS and MLC most likely would be applied to achieve superior maneuvering performance. For a bomber or transport the same two concepts would be applied to improve cruise efficiency, by reducing weight and drag.

Bomber and fighter aircraft normally require ride improvement only at crew stations, whereas transport aircraft (military and civilian) may require ride improvement along the entire length of the fuselage. Benefits of ride control are improved man-machine mission effectiveness, a by-product of reduced crew fatigue. FMC might be applied to a large transport to save wing weight, while for a fighter FMC could increase the number of different wing-mounted stores that could be carried.

3.0 STATUS OF CCV TECHNOLOGY

In the evolution of aircraft design many technology innovations have been incorporated to achieve improved performance. Today, before any innovation is accepted, the aircraft designer must have confidence that the new concept will:

- Be safe
- Achieve predicted performance
- Be cost effective

These three criteria are not always weighted equally. For military aircraft, for example, attainment of significantly improved performance may be paramount, while for a civil transport safety is of primary importance. Cost effectiveness is important in either case.

The next several pages will describe the genesis and current status of CCV studies and applications, and then summarize the current position of CCV technology.

3.1 Genesis of CCV Technology

Requirements to fly faster and farther have resulted in flying higher and designing larger more flexible airframes. This trend has made it more challenging to design aircraft having acceptable performance, stability and handling qualities throughout the mission profile. Large, high speed, flexible aircraft normally have inadequate short period and Dutch roll damping, resulting in objectionable handling qualities. In addition, elastic modes of such aircraft are normally strongly coupled throughout the frequency spectrum, resulting in increased response to longer gust wavelengths and higher dynamic structural loading. Although automatic pilots have been used to control rigid body aircraft dynamics for over fifty years, systems to control aircraft elastic modes have been considered seriously only during the past decade (References 1 - 23).

During one of the earliest programs, active control of a B-52 lateral body bending mode to reduce aft body structural fatigue damage rate was studied (Reference 1). A system was synthesized to control a 1.25 Hz antisymmetric mode using the rudder with an aft body accelerometer for feedback. This system was not implemented, however.

The B-52 was initially designed as a high altitude bomber, but mission requirements were later expanded to include low-altitude, high speed flight. Increased turbulence at the low-altitude environment results in larger peak loads, increased fatigue damage rate and reduced controllability. Severity of the low-level environment was vividly illustrated in 1964 by a low-level flight test incident near the Colorado Rocky Mountains in which a severe turbulence encounter broke the vertical tail.

This incident and recommendations from a special Air Force committee emphasized the need for an advanced flight control system on the B-52. As a result, the ECP 1195 Stability Augmentation System (SAS) was developed and by mid 1971 was installed in the B-52G and H fleet (References 7 - 17). The system was designed to reduce peak loading, reduce fatigue damage rate and improve controllability in turbulence.

Airplane angular rate and linear acceleration information are sensed, processed and fed to wide bandpass hydraulic actuators for the elevator and rudder control surfaces. Triply redundant electronics and dual redundant hydraulic power provide system fail-operate reliability. The resulting system performance is illustrated in Figure 3 and Table I. Typical flight test results of aft body side displacement in random turbulence with and without the SAS are shown in Figure 3. With the SAS off the 1.4 Hz body lateral bending mode dominates the response. The SAS reduces lateral RMS displacements by a factor of six. As a result mission flexibility and aircraft structural life are significantly improved. The upper portion of Table I compares the relative time for a B-52G or H airplane to accumulate equal fatigue damage operating at low level with and without the system. The lower portion of the table compares relative time to reach an overload occurrence operating at the same conditions. Since its installation the system has demonstrated excellent performance and reliability characteristics under field operational conditions.

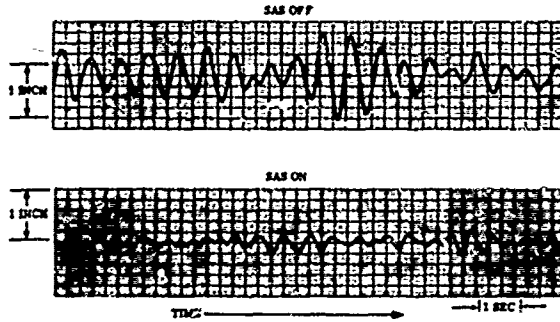


FIGURE 3. AFT BODY SIDE DISPLACEMENT IN RANDOM TURBULENCE WITH ECP 1195 SAS

Two additional structural mode control research programs were directed by the Air Force Flight Dynamics Laboratory (AFFDL) concurrently with the ECP 1195 program. North American Rockwell conducted a program on the XB-70, "Gust Alleviation and Structural Dynamic Stability Augmentation System (GASDSAS)", (References 18 - 24). Boeing and Honeywell jointly conducted a program using the B-52, "Load Alleviation and Mode Stabilization (LAMS)", (References 25 - 28).

The objective of the XB-70 GASDSAS program was to design a control system to reduce rigid body and structural mode accelerations of a large, flexible, low load factor aircraft, flying at high speeds in turbulence. Two concepts were investigated. One concept employed Identical Location of Accelerometer and Force (ILAF) techniques using linear accelerometer blended signals. A second approach used blended signals from two remotely located angular accelerometers, known as the Differential Angular Acceleration (DAA) system.

Both systems used the elevons for force producers. In addition, small horizontal and vertical canards were shown to be effective for structural mode control. The ILAF system provided significant reductions in fuselage vertical accelerations as indicated in Figure 4.

The LAMS program (References 25 - 29) was initiated in 1966 to demonstrate the capability of an advanced flight control system to alleviate gust loads and control wing structural modes on a large, flexible subsonic aircraft. One specific goal was to flight demonstrate on a B-52 airplane a measurable reduction in overall airplane fatigue damage rates caused by turbulence while retaining or improving aircraft handling qualities. The LAMS system was designed to use existing ailerons, spoilers, elevators, and rudder surfaces with new wide bandpass electrohydraulic actuators. The system was synthesized and demonstrated at three typical flight conditions. For comparison a baseline SAS representing a contemporary stability augmentation system was designed to control only rigid body motions. Fatigue damage rate improvements with the LAMS flight control system are shown in Figure 5, based on an annual usage consisting of 4.3 percent time at a low level, high speed condition, 6.8 percent time at a low level, low speed condition, and 88.9 percent time at high altitude cruise. The LAMS system reduced controls locked (no SAS) wing fatigue damage rate approximately 50 percent.

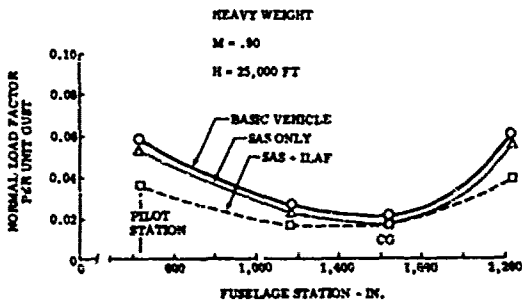


FIGURE 4. XB-70 ACCELERATION REDUCTION WITH ILAF SYSTEM

TABLE I MISSION FLEXIBILITY WITH ECP 1195 SAS B-52G AND H FLEET 350,000 LB GROSS WEIGHT LOW LEVEL ENVIRONMENT

	SPEED	WITHOUT SAS	WITH SAS
TIME TO EQUAL FATIGUE DAMAGE	325 KEAS	1	11
	400 KEAS	0.32	2.7
TIME TO EQUAL OCCURRENCE OF OVERLOAD	325 KEAS	1	1000
	400 KEAS	0.26	26

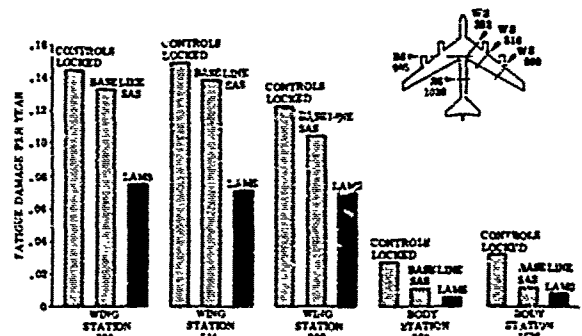


FIGURE 5. B-52 FATIGUE DAMAGE RATE REDUCTION WITH LAMS (FATIGUE DUE TO TURBULENCE)

Another goal of the LAMS program was to demonstrate that the LAMS technology is applicable to aircraft other than the B-52 test aircraft. An analytical study was conducted to determine the benefits that a LAMS system could provide on the C-5A aircraft (Reference 29). The study defined a longitudinal axis and a lateral-directional axis LAMS flight control system providing reduced fatigue damage rates and improved ride qualities at a representative flight condition. System performance is illustrated in the comparison of baseline SAS and LAMS FCS fatigue damage rates and improved ride qualities at a representative flight condition. Typical system performance is illustrated in the comparison of baseline SAS and LAMS FCS fatigue damage rates shown in Table II. Fatigue damage rates are reduced by several orders of magnitude at critical wing locations. The small increase in horizontal tail root fatigue damage rate with the LAMS system was not considered significant because of the relatively low damage rate. It is not unusual for LAMS type systems to exhibit performance degradations at noncritical locations.

The Lockheed-Georgia Company in 1969 designed, developed, and flight tested an Active Lift Distribution Control System (ALDCS) for the purpose of reducing the design limit maneuver loads on the wing of the C-5A airplane (Reference 56).

The fail operative system controlled the ailerons with signals obtained from accelerometers located near the node of the first bending mode and included active control of the inboard elevators for pitch compensation. The system as developed made maximum use of existing system components and required the addition of the sensors, a load alleviation computer system, and new interfaces to existing components. Flight tests showed the system to be highly satisfactory, both from the standpoint of achieving the desired static load relief as well as from a handling quality standpoint.

According to Reference 30 a version of the ALDCS will be utilized for the purpose of extending the fatigue life of the C-5A airplane by reducing maneuver and gust loads.

TABLE II C-5A FATIGUE DAMAGE RATES

VEHICLE STRESS STATION	FATIGUE DAMAGE RATE PER HOUR	
	BASELINE SAS	LAMS PCS
WING STATION 748	0.388×10^{-4}	0.924×10^{-8}
WING STATION 120	0.468×10^{-4}	0.576×10^{-9}
FUSELAGE STATION 1108	0.1048×10^{-11}	0.328×10^{-13}
FUSELAGE STATION 1804	0.208×10^{-7}	0.224×10^{-8}
HORIZONTAL TAIL ROOT	0.405×10^{-9}	0.491×10^{-8}
VERTICAL TAIL ROOT	0.635×10^{-7}	0.704×10^{-9}

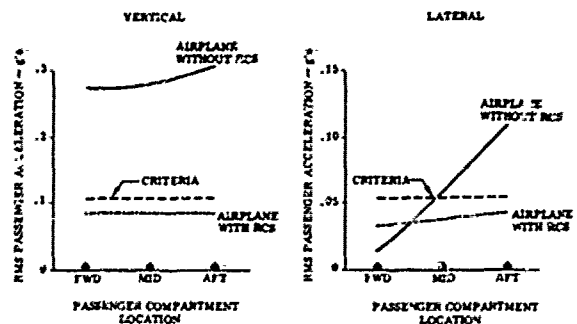


FIGURE 6. STOL PASSENGER RIDE QUALITIES DURING DESCENT

An early Honeywell study of the use of elastic mode suppression for ride quality improvement is described in Reference 31. Vertical accelerations on a double-delta SST configuration were shown to be reduced 50 percent by means of a suitably designed control system. Both rigid body and structural modes were suppressed.

In 1969 the first paper was published which described performance benefits that might be achieved by using the CCV design approach (Reference 32). The study was conducted for a B-52 bomber aircraft and the tail surface areas were reduced 50 percent with the augmented stability concept. This reduction in tail area reduced drag three percent, operating weight empty (OWE) nine percent (16,000 pounds), and gross weight 11 percent (53,000 pounds) for the same mission.

This study was later repeated in more depth, and the results were similar (Reference 33). Design gross weight was reduced 13.7 percent and OWE was reduced 13 percent for the same mission. The large gross weight and OWE reductions resulted from the relative insensitivity of range with gross weight, which is typical of long-range designs with small payloads.

Augmented directional stability was adopted on the British TSR2 low-level strike aircraft, using a small all-moving vertical fin, to provide lower gust sensitivity in high speed low-level flight, reduce aeroelastic effects and ease expert problems of yaw trim and loss of rudder effectiveness at transonic speeds (Reference 34). The moving fin was half the size of a conventional fixed fin, resulting in a seven percent decrease in profile drag, a 10 percent decrease in aircraft weight, and corresponding cost savings.

In 1970 the NASA Langley Research Center contracted with Boeing-Wichita to conduct a preliminary design study of a commercial Short Takeoff and Landing (STOL) airplane incorporating an advanced control system (Reference 35). The objective was to explore the feasibility of CCV technology providing satisfactory ride qualities and competitive high speed cruise performance with a low wing-loading STOL aircraft. The potential advantages of simplicity, reliability, and low noise of a low wing-loading STOL transport suggested an examination of this class of aircraft relative to other design approaches. The airplane was designed for 130 passengers, 2,000 feet field length, cruise Mach number of 0.80 and a range of 750 NM. The parametric study produced a configuration having a 0.35 thrust-to-weight ratio, a 50 psf wing loading, and which satisfied specified mission requirements and airworthiness standards. Passenger compartment vertical and lateral accelerations during descent (the highest acceleration condition) with and without the ride control system are shown in Figure 6. Design criteria were based on acceptable acceleration levels established during previous moving base simulator ride quality tests. The STOL design with the ride control system meets the acceleration criteria at all passenger locations in both the vertical and lateral axes.

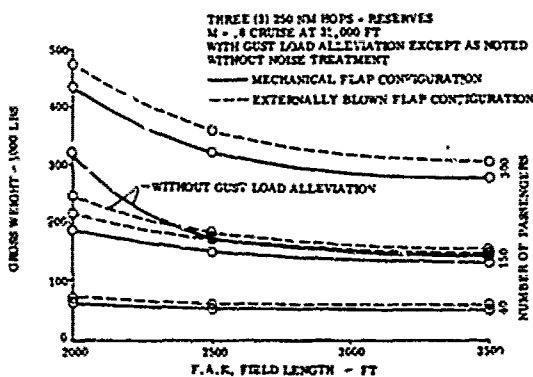


FIGURE 7. STOL TRANSPORT - SIZE COMPARISON

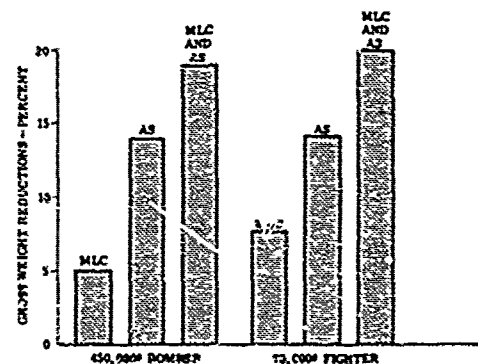


FIGURE 8. GROSS WEIGHT REDUCTIONS WITH MANEUVER LOAD CONTROL (MLC) AND AUGMENTED STABILITY (AS)

Because the results of the Reference 35 study appeared promising, NASA-Langley sponsored a broader scope effort which is reported in Reference 36. In the latter study STOL aircraft which used conventional mechanical flaps and low wing loading were compared against STOL aircraft which used externally blown flaps and higher wing loadings. All aircraft were designed to the same mission and employed an active GLA system. Results are shown in Figure 7.

It was concluded that through use of an active control system for both GLA and RC, low wing-loading airplanes with mechanical flaps are lighter, quieter and more economical than externally blown flap airplanes over a wide range of payloads and design field lengths. On the average the externally blown flap airplanes were about 12 percent heavier than the mechanical flap airplanes for the same mission and field length. GLA provided a large gross weight reduction for airplanes with design field lengths shorter than 2,500 feet. Without GLA the mechanical flap airplanes were heavier than the externally blown flap airplanes for field lengths less than about 2,400 feet.

In 1971 the U.S. AFFDL initiated a series of study contracts to investigate the compatibility of augmented stability and maneuver load control on different types of aircraft (Reference 37). Previous studies investigated potential performance payoffs of

individual CCV concepts and found that the benefits are strongly configuration and mission dependent. This study investigated the benefits which result from simultaneous application of the two concepts. Two classes of military aircraft were used in the Boeing study: a 450,000-pound B-52 bomber and the Boeing Model B18, a 70,000-pound fighter. Results of this study (Reference 38) indicated that the two concepts are compatible for both configurations. As shown in Figure 8 performance benefits, in terms of potential gross weight reductions, from each concept incorporated independently are essentially additive when both systems are incorporated simultaneously. In addition, performance improvements are approximately equal for both aircraft classes.

Results of a similar study conducted by McDonnell Aircraft Company personnel are reported in Reference 39. This study pointed out that "the basic performance improvement objectives to be achieved through the use of MLC and RSS are not the same for fighter aircraft as they are for bomber/transport aircraft. The objective for bomber/transport aircraft is to improve cruise efficiency as measured by performance parameters such as range and payload. For fighter aircraft, the design objective is to improve maneuvering performance as measured by performance parameters such as specific excess power (P_s) and maximum normal load factor ($n_{z_{max}}$). While the RSS design concepts are essentially the same for both fighter and bomber/transport aircraft, the MLC design concepts are notably different."

Results of the McDonnell study showed that, "1) significant performance benefits can be realized for a fighter aircraft through judicious application of MLC and RSS aircraft design concepts; 2) these design concepts are compatible; and 3) aircraft configured using these design concepts can be adequately controlled using existing control system technology."

Figure 9 "shows the effect of the MLC configurations on specific excess power at three typical combat flight conditions. It can be seen that the configurations with leading edge slats provide the best subsonic improvements in specific excess power. At $M = 0.90$, alt = 37,000 ft and $n_z = 3$ g, the leading edge slats result in an improvement in P_s of 140 fps. Supersonically, the canards increase specific excess power by 210 fps at $M = 1.45$, alt = 35,000 ft and $n_z = 5$ g."

Further examples of CCV benefits to fighter aircraft are presented in References 40 and 41, while Reference 42 discusses active wing/store flutter control for an F-4. The latter study indicated the possibility of expanding the F-4 permissible flight flutter envelope by 150 knots using the existing aileron control surfaces. Reference 43 presents results of a study to determine methods of improving efficiency of a 700,000-pound class jet transport by using wing maneuver load alleviation. Results showed that the airplane studied could have its wing span increased 10 percent for the same wing weight if MLC were used. The corresponding airplane performance improvement was 13 percent, resulting in a 10,000-pound payload increase. The use of the MLC system for gust alleviation also appeared promising.

The largest challenge that the designers of the U.S. Supersonic Transport (SST) faced was to achieve a design that was economically competitive with current large subsonic commercial transports. Configuration studies indicated that substantial improvements in weight, drag and balance could be obtained if conventional stability requirements were relaxed and the operational cg range was shifted aft (References 44 and 45). Coincidentally, design development of the stability augmentation system progressed to the point where complete confidence existed in the ability to develop a backup stability augmentation system (SAS) with the same reliability as the structure. System reliability was provided through design simplicity, component quality, design installation quality and employment of multiple, redundant, independent channels. Engineering analyses showed that sufficient reliability could be developed to fully rely on such a system for acceptable handling qualities in the event of shutdown of the normal augmentation system. The backup SAS, being likened to structure, was christened "Hard SAS", or HSAS. Adoption of the HSAS permitted a five percent aft shift in the operational cg range. This aft shift in balance resulted in weight savings and drag reduction and provided full utilization of body length with improved loading flexibility and reduced ballast requirements. The net result was a 150-inch reduction in body length and 100 square feet reduction in vertical tail size. Horizontal tail area remained unchanged, since the change in control requirements was met with a change in stabilizer deflection range. These configuration changes, plus additional minor system changes, culminated in a total weight savings of about 6,000 pounds. In addition, significant drag reduction also resulted from the reduced surface area and improved trim drag situation. For example, for cruise at Mach 2.7, drag was reduced about 2.5 percent. As a result of adopting the HSAS approach, aircraft range was increased 225 nautical miles. To achieve this same range without the HSAS, the 48,806 pounds of payload would have to be reduced about 30 percent.

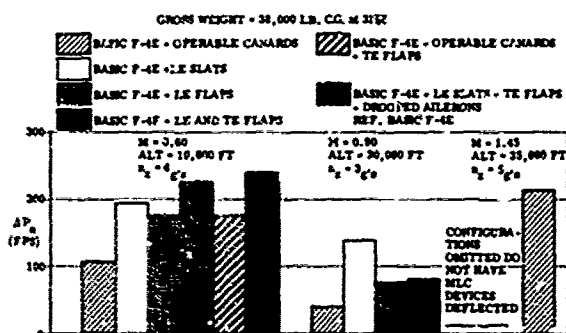


FIGURE 9. MODEL F-4E - EFFECT OF MLC DEVICES ON ENERGY MANEUVERABILITY

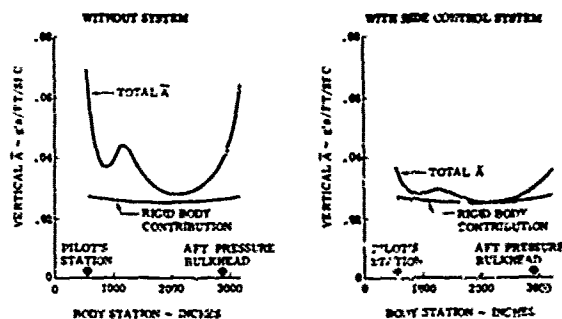


FIGURE 10. SST VERTICAL ACCELERATIONS WITH AND WITHOUT RIDE CONTROL SYSTEM (SUBSONIC DESCENT CONDITION)

Potential benefits of ride control and flutter mode control systems were also studied during the U.S. SST program (Reference 46). The ride smoothing analysis indicated that the first four body modes in the vertical plane, covering a frequency range from approximately 1.4 to 3.8 Hz, contributed significantly to gust induced accelerations. The ride control system was designed to suppress these elastic vibration modes without significantly affecting rigid body dynamics. Figure 10 illustrates fuselage vertical accelerations for a typical subsonic descent condition with and without the system.

The upper curves are total vertical accelerations, the lower curves are rigid body contribution, and the differences between the curves are aircraft flexible mode contributions. The system significantly reduces vertical accelerations at the forward and aft ends of the aircraft, where accelerations are largest.

The flutter mode control studies were directed toward actively damping two flutter modes on a delta wing SST strength

designed configuration. At Mach 0.90 the configuration had a 3.5 Hz wing mode and a 2.8 Hz body-wing mode with zero damping at speeds less than $1.2 V_{DIVE}$ (V_D), requiring over 19,000 pounds of additional stiffness structure to provide an adequate flutter margin. Studies were conducted to determine the feasibility of damping these modes with a flutter mode control system to eliminate weight. Figure 11 shows damping ratios of the two flutter modes, with and without the flutter mode control system. The system used wing tip aileron surfaces and two wing mounted pitch rate gyros. Without the system, the body-wing mode becomes unstable at $0.96 V_D$. The system extends flutter beyond $1.2 V_D$. Root loci of the two flutter modes as a function of velocity are shown in Figure 12.

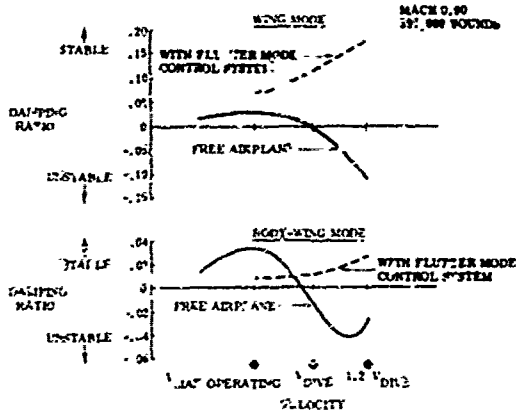


FIGURE 11. EFFECTS OF FLUTTER MODE CONTROL ON SST

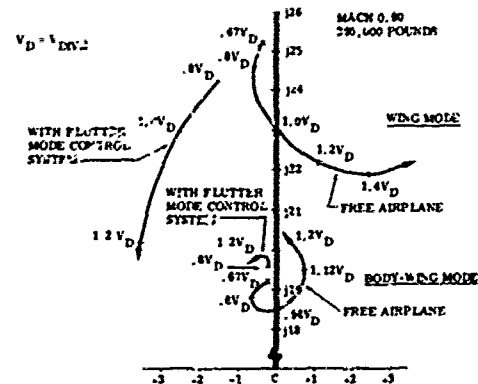


FIGURE 12. SST ROOT LOCI WITH FLUTTER MODE CONTROL SYSTEM

The NASA-sponsored Advanced Technology Transport studies examined the potential payoff of "active controls concepts" which are similar to the CCV concepts listed previously. Results from Boeing studies, Reference 47, are quoted below.

"Although the problem of aerodynamic center shift does not affect the subsonic airplane to the same extent as it does the SST, there are still substantial gains to be realized through the use of longitudinal SAS. As will be shown, these are associated primarily with reduced horizontal tail size and lower trim drag in cruise. However, the benefits include reduced trim drag for takeoff, climbout, and landing approach and, in some cases, reduced vertical tail size.

"Tail size selection criteria are illustrated in Figure 13, which superimposes the landing approach trim requirement and the criterion for static stability with a three percent stability margin, shown as solid lines. It should be noted that the static stability criterion corresponds to the critical (at least) stable combination of airplane configuration and flight condition. These criteria are expressed in terms of the cg position allowed as a function of horizontal tail volume coefficient. Conventional airplane design standards usually require a three percent static stability margin measured from the neutral point. A further requirement fixes the airplane's cg range required for loadability. The tail volume and, therefore, tail size can then be selected as the value that simultaneously satisfies the three above criteria. Relaxing the stability requirement to that corresponding to neutral maneuver stability, for example, allows a substantial reduction in tail size and permits operation of the airplane in a more aft cg range to reduce trim drag. For such operation, a highly reliable stability augmentation system must be incorporated as an integral part of the control system design. The possibilities of operating in the unstable region indicated by the dashed line may be attractive but require correspondingly greater attention to SAS concept selection with appropriate levels of redundancy. To achieve the same aft movement of the cg with normal static stability criteria would obviously require large increases in tail size. Although the loadability of the airplane might thereby be improved, it is doubtful that the cg range available could be used effectively.

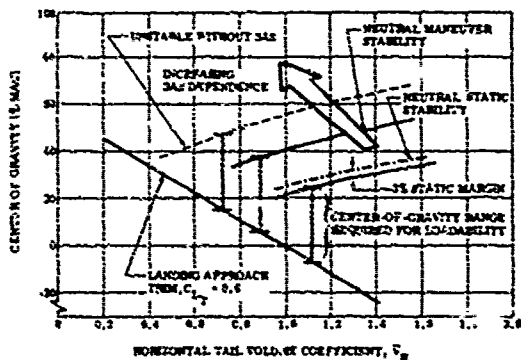


FIGURE 13. HORIZONTAL TAIL SIZING CRITERIA

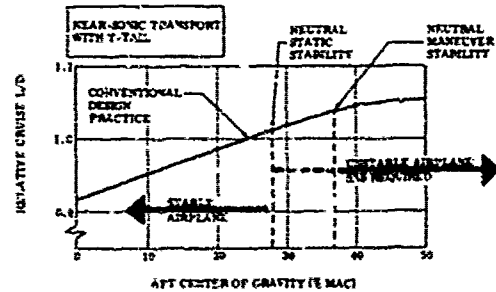


FIGURE 14. CG EFFECTS ON AERODYNAMIC EFFICIENCY

"A major portion of the gains referred to above are associated with improvement in cruise L/D of the airplane. The sensitivity of this parameter to cg location is shown in Figure 14 for the near-sonic transport with a T-tail. It is apparent that substantial gains in aerodynamic efficiency of the airplane from that corresponding to neutral static stability are possible through the use of longitudinal SAS. It is also obvious that cg movement aft beyond 40 percent MAC yields relatively little gain in cruise. This is due to the fact that continued increases in positive tail load eventually start increasing trim drag. Finally, it is worth noting that improvements in L/D and $C_{L_{max}}$ with more-aft cg positions for both takeoff and landing approach can be significant in terms of noise or field length. All of the above benefits finally appear as improved cruise efficiency and reduced airplane size to perform a given mission.

"The use of active controls for improved lateral-directional characteristics, particularly directional SAS, can provide substantial reductions in vertical tail size and corresponding reductions in drag and weight. In most designs incorporating flight-critical directional SAS, the minimum size of the vertical tail is likely to be governed by engine-out criteria, but some directional stability would normally remain (e.g., $C_{n\beta} \approx 0.002/\text{degree}$). However, an unstable Dutch roll mode would result for most of the flight envelope. This would have to be offset by including a highly effective yaw damping function in the lateral-directional SAS, possibly including the use of ailerons in addition to the rudder.

"Active controls can also be used to save wing weight (Reference 48) by shifting the lift center of pressure inboard to reduce bending moment through appropriate aileron and/or flap motions. This could be effective for both maneuver and gust loads and would result in weight reductions because of reduced strength requirements. A further gain results from lessened fatigue damage, primarily from gusts, provided that the system can respond effectively to acceleration and/or other signals generated at appropriate points on the airplane. Flutter suppression can be implemented by appropriate motions of wing control surfaces in response to similar signals and may become important as the wing stiffness is reduced to achieve weight reductions through load alleviation. When improved ride quality is also sought as an active controls benefit, it becomes obvious that a very sophisticated, highly integrated, and reliable system must be developed to achieve the combined goals discussed above.

"The impact of the application of active controls for both longitudinal and directional SAS, combined with load alleviation, is shown graphically in Figure 15. The effects are apparent throughout the entire configuration reflecting in changes in weight, balance, inertial characteristics, aerodynamic efficiency, and, finally, the overall size of the airplane. When the cumulative effects of these changes are assessed in the design cycle, the airplane takeoff gross weight is reduced 13 percent for a given mission with corresponding gains in DOC and return on investment (ROI) (Table III).

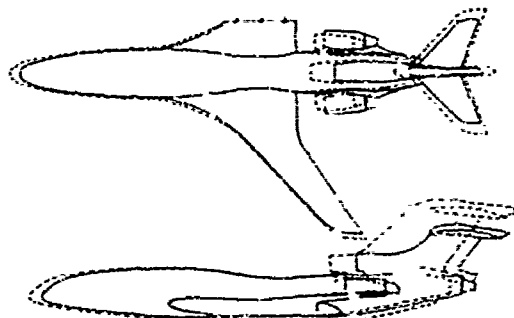


FIGURE 15. SUMMARY OF ATT ACTIVE CONTROLS BENEFITS

TABLE III ACTIVE CONTROLS BENEFITS FOR ADVANCED SUBSONIC TRANSPORT

CONFIGURATION	RELATIVE TOGW	RELATIVE DOC	RELATIVE ROI
CONVENTIONAL TECHNOLOGY • STATICALLY STABLE • FAIL-SAFE LAW DAMPER • MINIMUM ACCEPTABLE EXALL	1.00	1.00	1.00
ADVANCED TECHNOLOGY • FLIGHT-CRITICAL SAS • MANEUVER LOAD ALLEVIATION • GUST LOAD ALLEVIATION • OPTIMIZED CROSS CO • REDUCED CG RANGE	0.87	0.98	1.14

"While the means for implementation of some of the techniques discussed above continue to be actively pursued, the potential gains as well as the difficulties from the application of others, e.g., load alleviation systems, are only now becoming appreciated. Since their implementation is less straightforward and the alternatives less apparent, their ultimate impact on airplane design is difficult to assess. This points up the need for adequate design and operating criteria and more reliable means of evaluating the performance gains before active controls technology can be successfully applied. Also, the need for continued development of more reliable systems and components cannot be overemphasized."

3.2 Fly-By-Wire Technology Development

Practical realization of CCV functions on production aircraft depends on Fly-By-Wire (FBW) relative systems with a reliability consistent with the function criticality. Two programs, the Air Force 680J Survivable Flight Control System and the NASA F-8 Fly-By-Wire program, are directed toward the development and flight demonstration of fly-by-wire systems (References 49 - 51).

The Survivable Flight Control System (SFCS) program is an advanced development program being conducted by McDonnell Aircraft Company under contract to the Air Force Flight Dynamics Laboratory. The principal objective of this program is the development and flight test demonstration of an F-4 aircraft with a Survivable Flight Control System utilizing fly-by-wire techniques.

The SFCS is a three-axis, fly-by-wire primary flight control system which functions to command aircraft motion, instead of surface position, as a direct function of pilot applied inputs. Improved stability characteristics are provided through the use of feedback control, proper placement of aircraft motion sensors, and the application of structural mode filters to attenuate aircraft resonant frequencies.

Quadruplex (four channel) redundancy is used in all feasible system components, including power supplies, to obtain improved system reliability and increased mission completion probability. The system is designed to sustain two similar failures per axis without significant degradation in performance.

The highly successful flight test program consisted of 84 flights totalling 88.5 flight hours. The first 27 flights were flown with a mechanical backup control system in the pitch and yaw axes. The backup system was removed as soon as adequate system performance and reliability were demonstrated. The remaining 57 flights explored virtually the entire flight envelope of the airplane, including flight at Mach 2 and air-to-air and air-to-ground attack mission evaluations (Reference 64).

The NASA F-8 fly-by-wire program objective is to prove the feasibility of digital fly-by-wire systems. An Apollo LEM digital computer and associated equipment provides three axis command augmentation control on an F-8 via a single digital channel backed by triplex fail-operational/fail-safe electric command channels in all three axes. A total of 18 flights and 25 flight hours have been accrued on the Phase I digital FBW system as of March 29, 1973. A total of 2,200 hours of operating time, counting both flight and iron bird simulator time, has been accomplished with no failures in the primary digital system (Reference 63).

The Sperry Rand Corporation, which is a participant in the F-4 and F-8 programs, has reached the following conclusions on the status of fly-by-wire technology (Reference 49):

- Mission reliability requirements of fly-by-wire systems for fighter aircraft can be met within today's state-of-the-art redundant system techniques using quadruplex and possibly triplex channels.
- Today's state-of-the-art electronic component technology, combined with redundant system mechanization techniques, permits implementation of electronic command augmentation channels that track with sufficient accuracy to satisfy fly-by-wire system failure transient and nuisance susceptibility requirements.

- Built-In-Test-Equipment (BITE) for preflight testing of a fly-by-wire system is an essential requirement for fly-by-wire systems.

3.3 Current CCV Developments

As evidenced by the foregoing discussions there has been considerable effort devoted to developing advanced flight control technology. The results have led to the emergence of a "Control Configured Vehicle (CCV)" technology directed toward optimizing an aircraft configuration for a given requirement, including controls and control systems capabilities as primary design variables, rather than considering only aerodynamics, propulsion and structures in the initial design stages. The Air Force Flight Dynamics Laboratory awarded a CCV contract to Loening Wichita (Reference 52) in 1971 to design, install and flight demonstrate four advanced flight control systems on a B-52 test airplane (Figure 16). These systems can be classified by function as: 1) ride control, 2) augmented stability, 3) maneuver load control and 4) active flutter suppression. System performance will be demonstrated individually and simultaneously to verify compatibility of all concepts. System performance goals are as follows:

- The Ride Control (RC) system will reduce fuselage vertical and lateral accelerations at the pilot's station at least 30 percent during flight through atmospheric turbulence.
- The Augmented Stability (AS) system will provide satisfactory flying qualities for an aircraft configuration with neutral static stability.
- The Maneuver Load Control (MLC) system will reduce wing root bending moments at least 10 percent at the flaps up flight condition that produces maximum wing root maneuver loads.
- The Flutter Mode Control (FMC) system will extend the basic airplane flutter placard speed (V_{FO}) at least 30 percent.

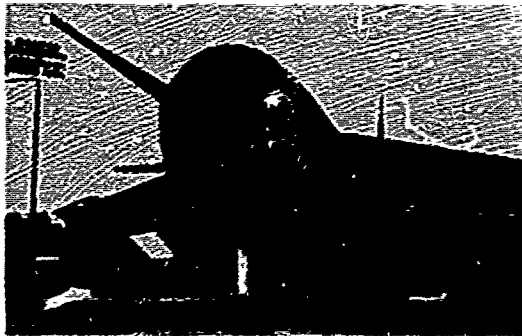


FIGURE 16. B-52 CCV FLIGHT VEHICLE

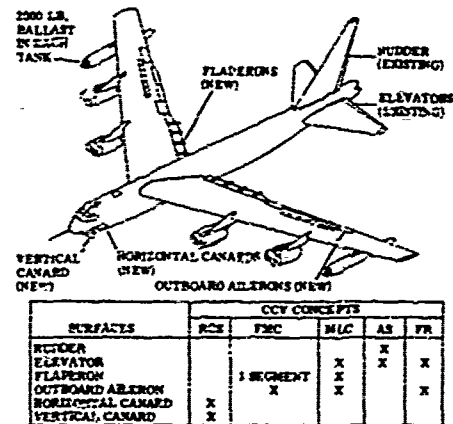


FIGURE 17. B-52 CCV CONTROL SURFACES

Analyses have been conducted to determine optimum control surfaces and surface locations and sizes for each CCV concept. New control surfaces consist of forward body horizontal and vertical canards, three-segment flaperons and outboard ailerons used with the existing rudder, elevators and inboard ailerons. Control surfaces required by each CCV system and surface locations are shown in Figure 17. Demonstration of a flutter mode control system necessitated ballasting the external fuel tank to induce a flutter condition within the B-52 flight envelope.

The Ride Control system, designed by the AFFDL, was flight tested early in 1973. Typical flight test results, shown in Figure 18, verify that the performance goal of 30 percent vertical acceleration reduction was attained. Similar results indicate that lateral accelerations were reduced 34 percent. The Flutter Mode Control system was first demonstrated on 2 August 1973 with a flight 10 knots above the airplane flutter speed. Flight demonstration of all concepts will be completed late in 1973 (Reference 52).

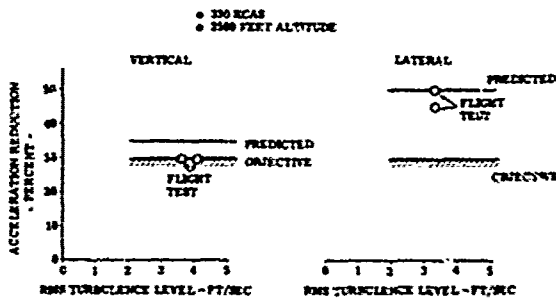


FIGURE 18. B-52 CCV RIDE CONTROL FLIGHT TEST RESULTS (PILOT'S STATION)

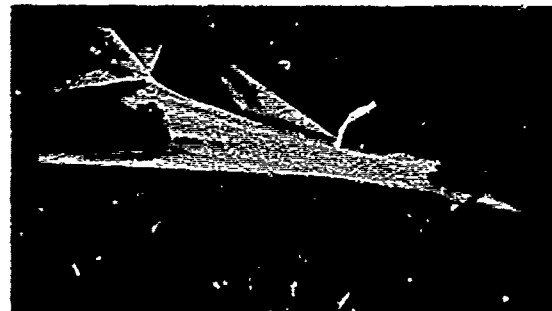


FIGURE 19. B-1 SUPERSONIC BOMBER

A structural mode control system is being designed by North American for the B-1 strategic bomber to improve crew ride qualities during turbulence and terrain following (References 53 - 55). A system based on the ILAF technique will use anhedral canards (Figure 19) completely dedicated to the ride smoothing function. Canard surface deflection and rate limitations are +20 degrees and 200 degrees per second respectively. Preliminary estimates of system performance at the crew station are shown in Figure 20. The system is not effective at low gust magnitudes because of nonpiston threshold and hysteresis, nor at high gust magnitudes because of saturation. The system is designed to provide approximately 70 percent reduction in acceleration within the most meaningful gust range in terms of human sensitivity and probability of gust encounter.

The Lockheed L-1011 commercial transport lateral axis was designed around the CCV concept (Reference 56). A mission

All six CCV concepts have achieved "proof of concept" flight test status through the B-52 ECP 1195, LAMS or CCV programs. Augmented stability and ride control will achieve prototype test status through the YF-16 and B-1 programs, respectively. The concepts of fatigue reduction and gust load alleviation achieved production status first with the B-52 ECP 1195 program. The Active Lift Distribution Control System (ALDCS) currently being designed for retrofit to the C-5A fleet accomplishes fatigue life improvements by maneuver and gust load reductions. The Lockheed L-1011 lateral gust design loads were reduced 20 percent to take credit for the use of the yaw damper. The use of active controls for flutter suppression has been demonstrated in the B-52 CCV Program for a 2.4 Hz symmetric wing bending and torsion mode.

It should be noted that the use of active controls for fatigue life improvement and ride control imposes a much less severe reliability criteria on the system design than does use of controls for augmented stability, maneuver load control, gust load alleviation or, especially, flutter mode control.

To date the prototype YF-16 airplane is the most advanced example of CCV technology, incorporating a fly-by-wire control system with no manual backup, and using augmented stability for improved maneuvering capability to a greater degree than ever before.

Of those concepts which have achieved production status, only the L-1011 usage described above actually affected the design of structure in a new airplane. The B-52 ECP 1195 and C-5A ALDCS usages were to alleviate existing problems and were not used to reduce the amount of structure in the airplane. Nevertheless the success of these applications provides confidence towards using them to a greater degree in future aircraft.

4.0 CCV TECHNOLOGY DEVELOPMENT REQUIREMENTS

Having discussed the CCV state-of-the-art from the point of view of the airplanes which have been studied and/or modified, each of the six CCV concepts will now be examined separately to determine where further technology advances must be made to encourage wider application of CCV technology in airplane design.

Performance benefits which accrue from application of CCV concepts are strongly configuration dependent. For bombers and transports, the largest payoff appears to be offered by AS, through reduced empennage size, with the associated drag and weight reductions. For the U.S. SST both AS and FMC offered very significant weight advantages, and for STOL aircraft GLA appears quite fruitful. MLC and RS offer less significant payoffs.

For fighters the benefits of AS and MLC are more likely to be of importance in improved maneuvering and excess specific power advantages. FMC may allow a wider selection of external weapons to be carried. Since fighters tend to be designed to higher load factors than bombers or transports, the benefits of GLA and FR are likely to be less significant. RC offers crew comfort and performance benefits during low altitude high-speed flight.

4.1 Augmented Stability (AS)

The most common use of AS in aircraft today is yaw damping to improve Dutch roll characteristics. In some cases safety-of-flight considerations cause limiting of the aircraft flight envelope with one channel of the yaw damper failed.

Technology improvements in several areas must be accomplished before an AS system will be utilized for anything except limited application. Design criteria for such systems need to be developed, to fully address on a mission basis the questions of redundancy and reliability. The "Hard SAS" criteria of the U.S. SST; i.e., no failures expected in the total lifetime of an SST fleet (Reference 45) is realistic, though perhaps difficult to achieve. To fully utilize the CCV preliminary design philosophy, methods of developing accurate, rapidly iterative aeroelastic mathematical models must be improved in order to assess the effects and payoffs of the AS system. Detailed, accurate weights estimations of the AS system during preliminary design become increasingly significant in that the payoff analyses are critically dependent on them. There is an additional requirement to develop improved methods for predicting nonlinear aerodynamic characteristics for elastic vehicles at high angles of attack and yaw, because nonlinearities complicate the design of the control system. The use of aeroelastic wind tunnel models with active control systems is a tool which offers considerable power in CCV airplane design, but one which needs development to a level commensurate with the risks being assumed.

When the system design is accomplished, hardware reliability must be assured, both from a component and total system point of view. This may be accomplished by use of service proven components with stringent requirements on selection, derating, acceptance, testing and burn-in. Use of functional full scale "iron bird" flight control system mockups provides a means of debugging the system and achieving confidence in its satisfactory operation.

Control system fabrication techniques must be extremely high quality. This will require increased rigor in assembly, inspection and testing prior to flight. Improved maintenance and testing techniques will be called for once the system is installed. Built-in test equipment which utilizes self-test techniques will be an integral element of the total flight control system.

4.2 Gust Load Alleviation (GLA)

The GLA feature of the B-52 ECP 1195 and LAMS programs must be considered non-safety-of-flight critical because no structure was removed when the systems were incorporated. The reduction of lateral gust design loads in design of the Lockheed L-1011 more fully utilized the benefits of GLA and safety was provided with a fail operative yaw damper. Reliance on a full-time GLA system to prevent catastrophic vertical gust overloads, with corresponding benefits in lighter structure, is yet to be accomplished. In general the requirements for technology development to support the use of GLA are similar to those for AS, above.

4.3 Maneuver Load Control (MLC)

The CCV B-52 MLC system has demonstrated the feasibility of the MLC concept. The C-5A Active Lift Distribution Control System is the first fleet application of the concept. Incorporation of MLC into preliminary design of new airplanes requires development of technology similar to that already discussed above for GLA and AS.

4.4 Fatigue Reduction (FR) and Ride Control (RC)

These two concepts are listed together because they are both nonsafety-of-flight and are ready today for production commitment for those applications where they can be shown to pay off. These two concepts are regarded as nonsafety-of-flight because failure of a properly designed FR or RC system merely reverts the airplane back to a higher fatigue damage rate (or rougher ride) for that period of time when the system is off.

4.5 Flutter Mode Control (FMC)

Of all CCV concepts, FMC is perhaps the most sensitive to configuration, especially wing planform and thickness. Because of this sensitivity, the potential payoffs must be investigated in great depth with high-quality trade studies prior to acceptance of FMC in a design. Improved prediction methods for three-dimensional unsteady aerodynamic forces are needed for such analyses.

The effects of manufacturing tolerances on flight control system operation are likely to be more critical for FMC than for the other CCV concepts. Methods of predicting these tolerances and accounting for their effects must be developed. If a FMC is relied upon full time within the normal flight envelope, the system must meet the HSAS reliability criteria. Less reliability may be required if the system is needed only in emergency conditions with a low probability of occurrence.

It may be possible to design a FMC system which is self-adaptive to changes in flight environment and airplane configuration. Confidence in such a system would be strengthened by wind-tunnel and subsequent flight test demonstration of a FMC system which stabilizes higher frequency flutter modes than the 2.4 Hz mode demonstrated in the CCV B-52 program.

4.6 Summary of CCV Technology Development Requirements

Table V summarizes briefly those areas of technology where advances are needed before wide-spread use of the CCV design philosophy will occur. These requirements may be summarized into three main areas:

- Reliability development and verification
- Economic payoff trades development
- Improved preliminary design techniques

TABLE V SUMMARY OF CCV TECHNOLOGY DEVELOPMENT REQUIREMENTS

REQUISITE	REQUIRED FOR			
	GUST LOAD ALLEVIATION (GLA)	AUGMENTED STABILITY (AS)	MANEUVER LOAD CONTROL (MLC)	FLUTTER MODE CONTROL (FMC)
DEVELOP RAPIDLY ITERATIVE MATH MODELS	X	X	X	X
DEVELOP SYSTEM DESIGN CRITERIA	X	X	X	X
DEVELOP IMPROVED AEROELASTIC WIND TUNNEL MODELS WITH ACTIVE CONTROLS	X			X
IMPROVE PRELIMINARY DESIGN PAYOFF ANALYSIS METHODS (ESPECIALLY WEIGHT ESTIMATIONS)	X		X	X
IMPROVE NONLINEAR AERODYNAMIC AND CONTROL SYSTEM PREDICTION METHODS	X	X	X	X
DEMONSTRATE HARDWARE RELIABILITY	X	X	X	X
DEVELOP IMPROVED FABRICATION TECHNIQUES	X	X	X	X
DEVELOP IMPROVED FAILURE DETECTION TECHNIQUES				X
DEVELOP IMPROVED MAINTENANCE & TEST TECHNIQUES	X	X	X	X
EVALUATE IMPACT OF MANUFACTURING TOLERANCES				X

Development of a "core" of CCV technology is envisioned as a means of encouraging greater use of CCV concepts in aircraft design. The core requirements are summarized in Table VI, where it is seen that the principal technologies involved are structures, controls and aerodynamics/propulsion.

TABLE VI "CORE" CCV TECHNOLOGY REQUIREMENTS

REQUISITES	STRUCTURES	CONTROLS	AERO/PROPULSION
I RELIABILITY DEVELOPMENT AND VERIFICATION		<ul style="list-style-type: none"> ● CONTROL SYSTEM COMPONENTS ● IMPLEMENTATION <ul style="list-style-type: none"> - DIGITAL VS. ANALOG - REDUNDANCY - MAINTAINABILITY ● POWER SYSTEMS IMPLEMENTATION <ul style="list-style-type: none"> - HYDRAULIC - ELECTRIC 	<ul style="list-style-type: none"> ● SAFETY CRITICALITY
II ECONOMIC TRADES DEVELOPMENT	<ul style="list-style-type: none"> ● STRUCTURAL TRADES 	<ul style="list-style-type: none"> ● CONTROLS TRADES ● SYSTEMS TRADES 	<ul style="list-style-type: none"> ● PAYOFF TRADES - PERFORMANCE VS. COST
III IMPROVED PRELIMINARY DESIGN TECHNIQUES			
A. DESIGN CRITERIA	<ul style="list-style-type: none"> ● STRUCTURES DESIGN CRITERIA 	<ul style="list-style-type: none"> ● CONTROLS DESIGN CRITERIA ● PASSENGER RIDE QUALITY TRADES ● SYSTEMS DESIGN CRITERIA 	<ul style="list-style-type: none"> ● CONTROL POWER DESIGN CRITERIA
B. IMPROVED ANALYSIS TECHNIQUES	<ul style="list-style-type: none"> ● MATHEMATICAL MODELLING 	<ul style="list-style-type: none"> ● RELIABILITY AND SAFETY CRITICALITY ANALYSIS TECHNIQUES ● POWER SYSTEMS ANALYSIS TECHNIQUES ● AUTOMATIC C.G. CONTROL 	
	<ul style="list-style-type: none"> ● AEROELASTIC WIND TUNNEL MODELS WITH ACTIVE CONTROLS ● WEIGHTS PREDICTION AND VERIFICATION 		

Figures 23, 24 and 25 are block diagrams representing possible methodologies of achieving technology advancements toward wider application of CCV concepts. Research results would be applicable to the entire field of advanced flight control system design as well as to CCV technology.

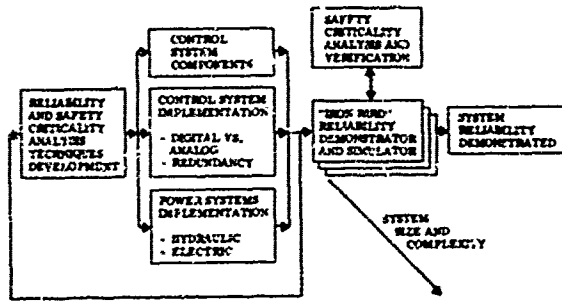


FIGURE 23. CCV TECHNOLOGY DEVELOPMENT REQUIREMENTS I - RELIABILITY DEVELOPMENT AND VERIFICATION

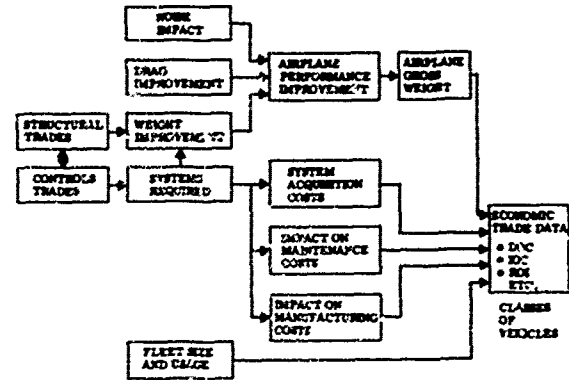


FIGURE 24. CCV TECHNOLOGY DEVELOPMENT REQUIREMENTS II - ECONOMIC PAYOFF TRADES

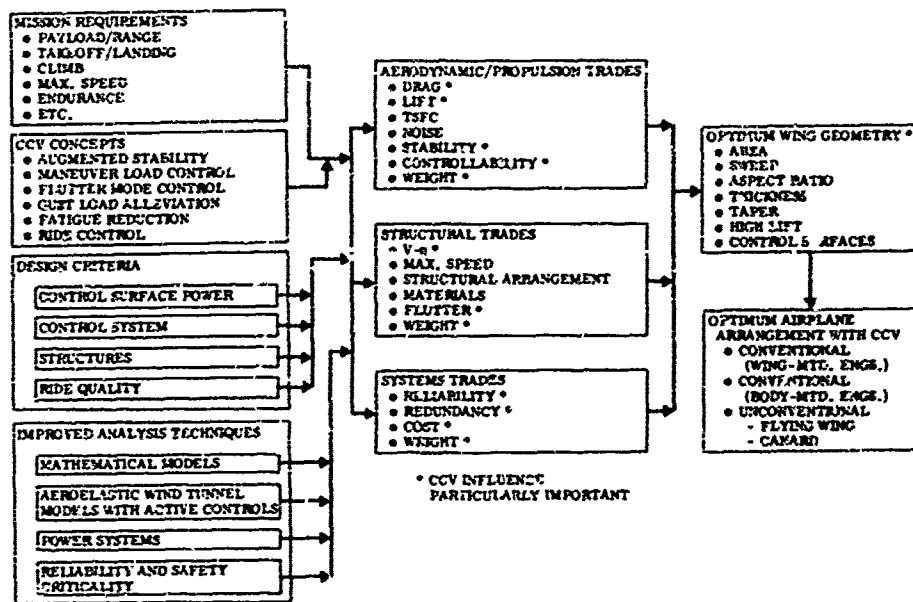


FIGURE 25. CCV TECHNOLOGY DEVELOPMENT REQUIREMENTS III - IMPROVED PRELIMINARY DESIGN TECHNIQUES

Present methods of using "iron birds" to develop confidence in system reliability can be easily expanded to incorporate CCV systems (Figure 23). Some improvements in analytical techniques for predicting component and system reliability and safety criticality would be helpful. Use of flight simulators for evaluation of the safety criticality of any particular system or component failure will continue to be useful.

CCV concepts will be used only when clear-cut advantages accrue from doing so. Economic payoff trades (Figure 24) must be of highest possible quality, which means that trade study inputs such as system weights and costs must be well defined. The costs of additional analyses, laboratory tests, improved fabrication techniques and new maintenance requirements may be increased and must be accounted for. Impact of CCV systems on cost of ownership is difficult to assess, but must be done in order to measure the long-term advantages.

Figure 25 summarizes some of the areas where improved preliminary design techniques are needed to aid incorporation of CCV concepts. There presently is a lack of design criteria for airplanes which include CCV concepts. For example, how should control surface power be split between control functions and CCV functions? Should "dedicated" CCV surfaces be used instead of one surface being used to accomplish several functions? What criteria for acceptability of failure (and hence, redundancy) should be applied to a CCV system? Obviously, the criteria is not the same for a FMC system and a FBW system. Should several CCV systems be combined in a single "black box" or should they be separated physically and functionally? These are only a few of the many criteria questions which need to be addressed. The revised version of MIL-F-9490C which is now being prepared by Boeing-Wichita under AFFDL contract will provide guidance toward the development of flight control system reliability and redundancy criteria. The specification is being designed to be compatible with FBW and CCV concepts and will describe the analyses, ground tests and flight tests necessary to demonstrate adequate flight control system performance and reliability.

The interplay between control system criteria and structural criteria becomes extremely important in the airplane design if the GLA system is to be relied upon to prevent catastrophic overloads. How much authority should the GLA system assume? For maximum weight reduction the GLA system may be very authoritative. Should a FMC system operate throughout the flight envelope or be used only to achieve a flutter margin beyond the dive speed? Here again these questions are only a few of those which must be answered.

Ride quality criteria need to be established in terms which are meaningful to the airplane and control system designer. Presently missing are criteria at the low frequency end of the vibration spectrum where humans are most susceptible to motion sickness.

One of the most significant difficulties in analysis is the present state-of-the-art of aeroelastic mathematical modelling. CCV control system syntheses require a complex mathematical model of the aircraft structural and aerodynamic characteristics. For example the B-52 math models used in the LAMs, ECP 1195 and CCV program analyses had as many as 30 degrees of freedom in each axis.

Approximately three months are presently required to develop an adequate math model of a large flexible aircraft for CCV system synthesis. This length of time makes assessment of potential CCV benefits very difficult during preliminary design activities. In addition the accuracy of current math modelling techniques is not well defined, and as a result industry lacks the technology and confidence necessary for committing critical CCV functions to production. Improvement in math modelling techniques will be a large factor towards incorporation of CCV techniques in airplane design. An alternate approach is to design control systems which are insensitive to math model accuracy.

In the past, wind-tunnel testing of dynamically scaled airplane models has proven economically desirable to predict airplane dynamic characteristics prior to flight testing. As aircraft become more dependent on stability augmentation systems, wind-tunnel testing of aeroelastic models to prove control concepts will become increasingly more attractive for reducing flight testing, as discussed in Reference 58.

Consequently, in 1967, AFFDL and NASA-Langley jointly initiated a program to demonstrate an active modal suppression system on a one-thirtieth scale B-52E aeroelastic model in the Langley transonic dynamics tunnel. This model includes aileron and elevator actuation systems and provisions for a cable mount system (Reference 59). Model gust responses have been obtained using the airstream oscillator system installed in the tunnel (Reference 60). Boeing-Wichita is assisting NASA in developing a ride smoothing system for the model using 50 Hz bandwidth aileron and elevator actuation systems. Subsequently, canards and flaperons were added for RC and FMC testing, which is now nearing completion. In 1974 a MLC system will be tested.

In addition wind tunnel tests are being conducted at NASA-Langley on a SST wing model which utilizes a FMC system (References 61 and 62). Wider use of such models will be of great benefit in CCV system synthesis and test.

Improved analysis techniques for hydraulic and electric power systems are required primarily to improve weight and cost estimating accuracies, for inputs to the economic tradeoff analyses. As noted earlier, improvements in reliability and safety criticality analysis techniques would be useful.

5.0 CONCLUSIONS AND RECOMMENDATIONS

The performance benefits of the CCV concepts have been demonstrated by a considerable number of studies and flight tests. To date CCV has been applied where a significant payoff was shown to result (U.S. SST, YF-16), and where benefits could be achieved with low risk (L-1011). As confidence is gained in CCV capabilities, civil and military aircraft regulations will be revised to take advantage of the benefits offered.

The acceptance of CCV technology in aircraft design will be accelerated in proportion to the research effort exerted to solve the problems of reliability development and verification, economic payoff trades, and improved preliminary design techniques, including development of design criteria and improved analysis techniques. Support of this research offers very worthwhile payoffs, and is recommended on a high priority basis.

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ACKNOWLEDGEMENT

The work reported in this paper is a result of the efforts of many people in the Air Force, the NASA and the aircraft industry. The author has summarized many studies and has attempted to properly credit their originators by referencing the data sources.

Within The Boeing Company the following individuals made particularly significant contributions:

Gerald E. Bergmann
Walter J. Rohling
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AVIONIC FLIGHT CONTROL SUBSYSTEM DESIGN AND INTEGRATION IN THE C-5 AIRPLANE

by

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SUMMARY

The preliminary design process had significant influence on the C-5 avionic flight control system development, production, and operational cost. The design decisions made during the preliminary design phase relative to the stability augmentation systems illustrate the extent of the impact on the design, test, manufacture, and installation of the avionic systems. These decisions lie mainly in the areas of mission success capability, airplane safety, reliability, survivability, and human factor characteristics and, for the illustrative stability augmentation systems, the aircraft's handling qualities. The design processes, including the subsystem integration with the airframe and with other functional subsystems, influenced the cost of the C-5 program. Experience gained from this program may lead to improvements in preliminary design decision making procedures.

DEFINITIONS

DUTCH ROLL - A lateral-directional dynamic oscillatory motion of the airplane characterized by a short period and a low damping ratio.

DYNAMIC PRESSURE - Pressure produced by the velocity of an airplane.

FAILURE - State of improper functioning of a subsystem component.

FAIL-OPERATIVE - Description of a subsystem which can provide normal functional performance following a failure of a component.

FAIL-SAFE - Description of a subsystem which cannot result in a hazardous condition following a failure of a component.

FEEDBACK - A measured signal of an airplane response, such as pitch rate, which is amplified and fed back to a control surface actuator to provide airplane guidance or to improve stability.

FLIGHT SIMULATION - A development tool which allows pilot evaluation of the airplane handling qualities performance. It normally consists of a cockpit with manual controls and appropriate flight instruments, a visual display providing pilot cue of the landing area and a computer which provides the mathematical model real-time computation of the airplane motion.

HARDWARE - Tangible form of a subsystem.

MACH TUCK - A nose down pitching moment tendency of an airplane as it accelerates in the transonic speed region.

MEAN AERODYNAMIC CHORD - Average airfoil section of a wing.

PHUGOID - A longitudinal dynamic flight oscillatory motion characterized by a long period and a low damping ratio.

SHORT PERIOD - A longitudinal dynamic oscillatory motion characterized by a short period and a high damping ratio.

SPIRAL - A lateral-directional dynamic mode of the airplane which is characterized by a slow spiraling motion following an upset from trim.

SUBSYSTEM - A group of interfacing components providing an independent function in support of the system performance.

SYSTEM - An airplane designed to perform given missions composed of numerous integrated subsystems supporting the performance.

VENDOR - A hardware manufacturing subcontractor which designs and produces one or more subsystems for installation on an airplane.

INTRODUCTION

The preliminary design process for an airplane influences the final design, and therefore cost, in a complex manner. By examining the design development and integration of a portion of the avionic flight control system of the C-5 airplane a number of the facets of this process reveal lessons which may aid in preliminary design of future systems. The factors facing airplane designers in their endeavors to create an optimum airplane have increased in number and influence in the past two decades. This increase has been caused by increased functional demands and resulting complexity of the airplane.

C-5 DESIGN PROCESS

Design decisions were made continuously throughout the various phases of the C-5 program concerning configuring the flight control subsystems. Each phase of development provided a different design environment in which decisions were made thus influencing the import of each design judgment. The total design process for the C-5 involved four distinct phases illustrated in Figure 1. The phases were:

- Long Range Heavy Logistics System Airframe Study Program (CX-HLS) (Ref. 1)
- C-5 Support System Project Definition Phase (PDP) (Ref. 2)
- C-5 Design Phase
- C-5 Production Phase

The first phase of the design process was a system definition period consisting of parametric and trade studies. These studies defined the Heavy Logistics System mission requirements for payload, range, and compartment size and created a recommended airplane conceptual design for cost effectiveness evaluations. This was the embryo phase of the ultra large transport airplane.

The second phase, the C-5 Project Definition Phase, provided a complete detailed preliminary design effort which resulted in a C-5 system proposal detailing a recommended airplane with the how, when, and where it would be developed, tested, and produced. The functional concepts for each subsystem were devised during this development period. Lockheed-Georgia Company was awarded the contract at the end of this phase to accomplish the last two phases.

The third period was the final phase in which the normal steps of airplane design and test were accomplished to meet the conceptual requirements established in the second phase.

The last phase, the Production Phase, consisted of construction and assembly of the airframe, the engines, and the multitude of subsystems into a flyable and serviceable airplane. This phase was achieved concurrently with development and test so the airplanes would be quickly added to the U. S. Air Force inventory.

AVIONIC FLIGHT CONTROL SUBSYSTEM DESIGN PROCESS

To illustrate the influence of decisions made in the development of the many and multifunctioning avionic subsystems shown in Figure 2, the discussion will be focused on the subsystems which provide a stable airplane for normal manual flight and for automatic control of flight. These are:

- Trim Compensator Subsystem
- Pitch Augmentation Subsystem
- Yaw Augmentation Subsystem
- Lateral Augmentation Subsystem

A fifth subsystem, the primary flight roll control system, had a profound influence on the development of the lateral and yaw augmentation subsystems although it does not directly provide stability to the airframe.

Trim Compensation Subsystem

The trim compensation subsystem is an artificial stability device which augments the longitudinal static speed stability of an airplane primarily in the transonic speed regions. During the CX-HLS study phase the results of wind tunnel tests indicated that the Mach tuck tendency of the study airplane did not have sufficient influence to require a trim compensation subsystem on the airplane. The history of the trim compensation subsystem operation on the Lockheed C-141 airplane whose configuration was similar to the recommended CX-HLS airplane also supported the absence of Mach tuck effects in normal operational conditions. Thus the recommended CX-HLS airplane in 1964 did not have a trim compensation subsystem.

During the Project Definition Phase of the C-5, subsequent wind tunnel tests indicated a more aggravated Mach tuck effect and supported a need for a trim compensation subsystem using Mach number at high altitudes, high speeds and dynamic pressure at very low altitudes, high speeds.

The proposed subsystem introduced pitching moments by operating inboard elevators through electrohydraulic actuators to accomplish its stabilizing functions. The product configuration requirement (Ref. 3) stated that a trim compensation subsystem was needed and was to be a fail-safe subsystem.

During the Design Phase when the functional details of the subsystems were in development it was decided, based on additional wind tunnel data to remove the dynamic pressure input, to include a Mach rate for phugoid damping and to use, in lieu of the inboard elevators, a horizontal stabilizer actuation system which was to be similar to the trim compensation actuation scheme effective on the C-141.

Later, after the trim compensation subsystem procurement was initiated and hardware development begun, the dynamic pressure input was reinstated with expanded operation in the low altitude, low speed flight regime.

During the trim compensation subsystem endurance testing of the horizontal stabilizer actuator, problems developed caused by the high frequency on-off operation of the electrohydraulic actuator. Lengthy additional development tests and design efforts were required to provide an adequate actuation interface.

As initial flight tests of the trim compensation subsystem were made to evaluate the subsystem performance it was evident that Mach rate was not an adequate flight parameter with which to improve the long period phugoid stability. Thus a corrective design effort was required involving analyses, flight simulation, and flight testing to replace the Mach rate with True Airspeed Rate signals. An acceptable subsystem was acquired and in production in time for the U. S. Air Force acceptance flight testing.

Static longitudinal stability was increased by the trim compensation subsystem as illustrated in Figure 3. With the subsystem operative the cruise neutral stability points of the C-5 were well aft of the design aft limit of center of gravity position by 32 percent mean aerodynamic chord. The basic airplane with the subsystem inoperative displayed neutral stability close to estimated data which were also safely aft of the design limit (Ref. 4).

This prompted the test team to perform a series of tests to determine if it was possible for the pilots to detect the difference in airplane speed stability as influenced by the trim compensation subsystem. The pilots performed a range of evaluation tasks unaware of whether the subsystem was engaged or not making "guesses" of the operational status of the trim compensation subsystem and denoting the airplane handling qualities. During the tests the pilots guessed the status correctly 40.5 percent of the tests, were incorrect 29.8 percent of the trials and they could not detect the difference in stability in 29.7 percent (Ref. 4).

After this thorough flight evaluation of the C-5 static longitudinal stability characteristics, the Air Force Test Team recommended that the trim compensation subsystem be deleted from the production aircraft (Ref. 4).

What were the influencing factors that would cause a decision to add this stability augmentation subsystem to the C-5 long after the longitudinal static speed stability was judged to be adequate and a trim compensation subsystem was determined not to be required, only to be deleted from the production airplane? The major cause was concern for flight safety in the critical transonic region.

The major effect of this design program was the early requirement for an unneeded subsystem which in turn caused unnecessary expenditures toward two interfacing subsystems. It is highly unlikely that this expenditure could have been prevented due to the hazards of improperly recognized transonic effects. All Mach trim compensators have a design process history similar to that of the C-5 subsystem. Wind tunnel data determines the initial requirement for these subsystems, but flight test evaluations provide the basis for the final decision. The U. S. Air Force flight test team deemed the longitudinal static speed stability to be adequate throughout the flight envelope including the high Mach, high altitude regime. Safety of flight demanded the presence of this subsystem until proven unnecessary in flight test.

Pitch Augmentation Subsystem

The recommended CX-HLS airplane was found to possess adequate longitudinal short period damping and response characteristics as specified by MIL-F-8785A (Ref. 5).

A trade study during the Project Definition Phase period evaluated the credibility of newly established short period oscillatory requirements evolving at that time from supersonic transport studies. This trade study found the C-5 damping adequate but noted that the pitch response was inadequate.

The C-5 airplane proposal recommended a fail safe pitch control modulator using pitch rate and control stick position signals to improve the pitch response. Compensating inputs were sent to the outboard elevators using an electrohydraulic servomotor. This system was developed without the benefit of piloted flight simulation evaluations.

Prior to the selection of an automatic flight control system manufacturer, flight simulation evaluations indicated that the C-5 possessed nominal pitch response. The proposal of the contracted manufacturer recommended a pitch augmentation subsystem with stick input for pitch quickening and pitch rate for damping. A fail-safe signal went to an electrohydraulic servomotor operating the outboard elevators. During the Design Phase this subsystem experienced many simulator evaluations and configuration modifications but the subsystem at first flight still used stick position and pitch rate feedback signals.

During this phase three major changes were made to the pitch augmentation. The subsystem was modified to be fail-operational requiring triple redundant electronics. The second alteration was the replacement of the electrohydraulic servomotor with an electrohydraulic valve arrangement on the primary surface servoactuator. These two changes were also incorporated on both the lateral and yaw augmentation subsystems and the decision to do so was made just prior to avionic vendor selection. The third modification came late in the design phase resulting from flight simulation evaluations. The operation was enlarged to include the cruise as well as the approach and landing configurations.

The longitudinal short period oscillatory characteristics for the C-5 are presented in Figure 4 (Ref. 7). The landing and power approach configurations do not meet the supersonic transport oriented criteria indicated but this criteria were discarded during the flight simulation evaluations. The Air Force Test Team reported the unaugmented oscillations to be essentially deadbeat throughout the flight envelope, but the damping provided by this subsystem was beneficial during the inflight refueling maneuver (Ref. 4). Thus the subsystem on the airplane today includes only minimum pitch rate damping.

The subsystem experienced a design cycle from nonexistence to a complex, sophisticated controller back to a nominally operative subsystem. The design effort required a large expenditure of manpower affecting the change of subsystem hardware well into the production stage. Could this have been prevented or at least kept to a minimum? The major factors influencing this design cycle were:

- Unknown handling qualities criteria of an ultralarge transport aircraft
- Advent of new flight simulation development technique
- Design schedule restraints

It is improbable that this subsystem could have been configured to its final form without major modifications. The introduction of unproven supersonic transport design criteria and the flight simulation development method provided a "learning curve" situation which could only peak during flight test.

Yaw Augmentation System

The CX-HLS airplane possessed adequate Dutch Roll characteristics when compared to the requirements of Specification MIL-F-8785A (Ref. 5). A simple yaw rate damper was added to improve the directional handling qualities. The stability and control trade study conducted during the Project Definition Phase further emphasized the adequacy of the basic airplane Dutch Roll. The C-5 Proposal report recommended a yaw rate damper with control wheel position included to improve the turn coordination which was degraded by the yaw rate input.

At contract award, two very demanding requirements were imposed on the airplane which caused significant changes to the yaw augmentation as well as the lateral augmentation yet to be discussed. The first was the definition of a critical 200 feet lateral offset maneuver to be performed during approach to landing. Passing through cloud cover at 200 feet altitude on an approach to landing and encountering an offset of 200 feet from the runway centerline this would prove to be the most demanding maneuver required of the aircraft. This maneuver is illustrated in Figure 5. To ensure adequate roll maneuver ability a requirement was imposed to achieve a bank angle of 8 degrees in the first second following maximum control wheel input in the power approach and landing configurations.

Extensive piloted flight simulation studies were conducted involving this critical maneuvering configuration during the Design Phase. The 8 degrees bank angle demand in the maneuver created a manual primary roll control system capable of producing extremely high roll rate capabilities. These high rates caused turn entry coordination problems which were solved by a more complex, multifeedback, yaw augmentation subsystem than originally proposed. Mission success bred a fail-operational, fail-safe configuration. The requirement of an 8 degree bank angle in the first second was not achieved as is illustrated in Figure 6 (Ref. 7). However, during flight test evaluations, the U. S. Air Force test team judged the roll performance of the aircraft to be satisfactory and stated that "from 200 feet altitude and 200 feet offset there appeared to be no problem getting the aircraft aligned with the runway" (Ref. 6).

The ability of this subsystem to coordinate airplane turn entry is provided in Figure 7. The sideslip angle resulting from a maximum manual roll control command with the augmentation operative is about one-half the sideslip with the subsystem disengaged. The damping effects are illustrated in Figure 8 (Ref. 7) and are excellent. The unaugmented airplane is only deficient in Dutch Roll damping in the high speed cruise configuration and the augmentation performs the damping task more than adequately. Decisions made during this design process were influenced principally by stringent design requirements and conventional analytical and newly developed flight simulation techniques. Subsystem configuration decisions made prior to hardware production were definitely correct judgements. This is confirmed by the orderly test process and the lack of major hardware design changes.

Lateral Augmentation Subsystem

The CX-HLS airplane was thought to possess adequate roll damping and spiral stability thus no lateral augmentation was recommended. The trade study of lateral directional characteristics mentioned the possible use of such subsystems but did not recommend the use of any artificial stabilization. The C-5 proposal report did not include lateral augmentation. With the addition of the critical runway offset maneuver requirement, a subsystem was developed by simulation methods using control wheel position for roll quickening, roll rate for damping, and roll attitude for improved spiral stability. Since this subsystem functioned in unison with the yaw augmentation subsystem while improving the C-5 handling qualities, both units were incorporated into the same computer. This allowed sharing common feedback sensors, electrical power circuitry, and built in test equipment. More importantly it allowed a reduction in airplane weight by approximately 15 pounds.

During initial flight tests the roll rate input was deleted since it significantly degraded roll response to primary control commands. The final configuration found to be acceptable by the test team possessed a minimally amplified roll attitude signal to the aileron for a slight improvement of spiral stability.

The effect of this subsystem on spiral stability for three pertinent flight conditions is illustrated by the spiral convergence times given in Figure 9 (Ref. 7). A positive value which denotes positive stability is the time required for the spiral mode to decrease in amplitude to one-half of the peak value following an upset disturbance. A negative value is an unstable spiral and is the time required to double the amplitude. The augmentation subsystem improves the spiral stability in

the landing and power approach configurations but is slightly destabilizing in the cruise modes. This instability is acceptable since the pilot workload is relatively light in cruise allowing time to correct minor deviations from the intended flight path.

This subsystem experienced the same cyclic process as the pitch augmentation, progressing from nonexistence during the initial CX-HLS study program to a complex multifeedback subsystem and returning to a single feedback fail-operative, fail-safe unit. Why did this subsystem not experience the same relatively smooth design process as the yaw augmentation since the yaw and the lateral subsystems were developed concurrently using the same design techniques and test facilities? The only plausible explanation is a tendency toward conservative design influenced by the involved interaction of the lateral-directional dynamic flight characteristics. Only extensive flight test evaluations provided a realistic understanding of these modes of airplane motion resulting in a simple lateral augmentation subsystem.

DESIGN DECISIONS

A perusal of the key decisions made during the design process of the C-5 stability augmentation subsystems, summarized in Figure 10, indicates a definite pattern of subsystem design which can be reasonably extended to the majority of the C-5 avionic flight control subsystems. The trend began with nonexistent or at the minimum, simple, single feedback subsystems, and progressed to four complex, multifeedback units operative for flight test evaluation. During the flight test period the design trend was back to simple, single feedback subsystems and in one case to removal from the production airplane.

During the system definition phase, the large size of the CX-HLS airplane had not yet caused an impact on the airplane handling qualities criteria. Also the flight simulation technique was not available for handling qualities evaluation of this ultra large transport. Thus augmentation subsystems were not required or were simple units.

Concern for the effects of the large sized airplane during the Project Definition Phase caused detailed parametric and trade studies which illuminated the unknown handling qualities and created new design criteria. This concern caused the creation of additional stability augmentation on the proposed C-5 airplane.

After the contract award for the design and production program, the following factors influenced the initial development decisions related to the augmentation subsystems.

- o New critical mission requirements
- o New unproven design criteria
- o Design conservatism
- o Piloted flight simulation development
- o Program schedule

The critical runway offset maneuver and transonic flight requirements coupled with the new unproven large transport design criteria caused the flight control subsystem design to be very conservative in configuring the augmentation functions. Also, for the first time in the design process, the flight simulator played a significant role in the initial development of handling qualities and the subsystems which augmented stability deficiencies. This allowed the pilot to provide his knowledge and experience at an earlier phase of the subsystem design process. The design engineer, in partnership with the pilot, possessed in the simulator a design tool capable of evaluating an almost unlimited number of subsystems configurations. Naturally the pilot had doubts about the performance of the simulated airplane; thus, he provided conservative contributions to the subsystem development.

All these effects combined to define complex, multifeedback, redundant stability augmentation subsystems which were fabricated for flight test evaluations in production form, but with minor configuration adjustments. This situation caused decisions made during the design phase of the C-5 to have significant and compounding effects on the final production avionic components.

During the flight test program, the engineering test pilot was back in his normal environment with complete trust in his evaluations of the airplane response characteristics. His evaluations of the handling qualities and the augmentation subsystem characteristics were free of the over-conservatism present during the flight simulation. Now the configurations of the subsystems were evaluated for possible simplifications and the use of production hardware did not facilitate the changes required for these configuration evaluations. However, many components were removed and one entire subsystem was deleted allowing an increase in system reliability, a decrease in airplane weight, and increase in airplane safety of flight, while the airplane still possessed adequate flying characteristics.

HINDSIGHT

Many lessons were learned from the C-5 avionic flight control subsystem development experience. Lessons learned can serve both as warnings of pitfalls and also as guidelines for a future event; thus, the C-5 experience should be incorporated into the design process of future aircraft programs. The first lesson was the requirement for design criteria. Meaningful, understandable, and usable design criteria must be available at the beginning of the preliminary design phase if future program schedules are to be as demanding as the C-5 program. Criteria, to be usable, must be validated prior to conceptual development. Design requirements cannot be verified concurrent with system development without unfavorable effects on costs.

The C-5 program experience also taught that design techniques such as mathematical analyses, piloted flight simulation, and hardware testing procedures in the laboratory and in flight, effected the attainment of final optimum subsystem configurations. The need for design methods to keep abreast of involved subsystem requirements was a double edged sword causing use of insufficient but immediately available techniques, or creating the development of adequate techniques concurrent with the actual requirement for their use.

The last major lesson learned from the C-5 design program was the requirement to recognize all the ramifications of a very short program schedule. Figure 11 illustrates the impact of allowed design time on the overall cost of a typical augmentation subsystem design and production program. This curve indicates a given optimum design time for a minimum cost subsystem program. The figure also indicates that there is a minimum time below which infinite available funding could not procure the subsystem. To meet the optimum time, design requirements, data, techniques, personnel, and interfacing information must become available with precise timing. Should a slippage occur in the availability of any of these, the design time will increase accompanied with a significant increase in cost.

EPILOGUE

Decisions made during the early design phase to place minimally required subsystems on an airplane can have far-reaching effects on the airplane long after it has entered service. Naturally these effects include maintenance and reliability, but also include functional growth. An operative subsystem on an airplane allows the possible incorporation of another function to that airplane. This can be simply another operation of the autopilot or a function almost totally independent of the subsystem to which it is added. Two of the stability augmentation subsystems previously discussed are now serving as outlets to a subsystem whose function is to improve structural characteristics of the C-5 wing. This subsystem defined as the Active Lift Distribution Control System (ALDCS) is intended to redistribute the wing spanwise lift during maneuvers and the response produced by atmospheric gusts to reduce the bending moment at the wing root and to increase the wing fatigue life. This redistribution is achieved through symmetrical deflection of the ailerons by inclusion of a compensated signal into the lateral augmentation subsystem. This load relief function decreases the C-5 short period damping characteristics; thus a corrective signal must be forwarded to the pitch augmentation subsystem. This subsystem is presently in development at Lockheed-Georgia Company and its addition to the C-5 airplane fleet is planned.

Decisions to include these two minimally required flight control subsystems on the C-5A facilitate immediate placement of a required function after the last production airplane was delivered to the U. S. Air Force. A Gypsy with her crystal ball could never have predicted the advancement of these augmentation subsystems into active roles of improving airplane service life.

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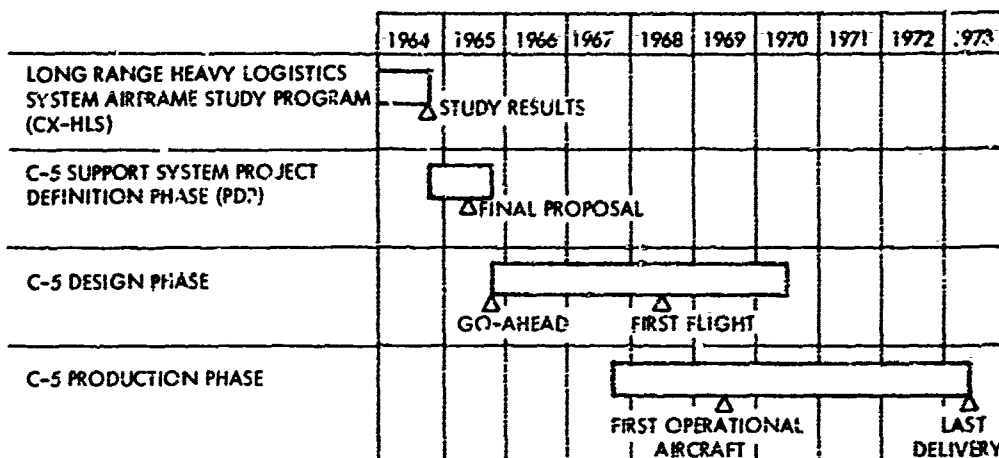


FIGURE 1. LOCKHEED C-5A AIRPLANE TOTAL AIRPLANE DESIGN & PRODUCTION PROCESS

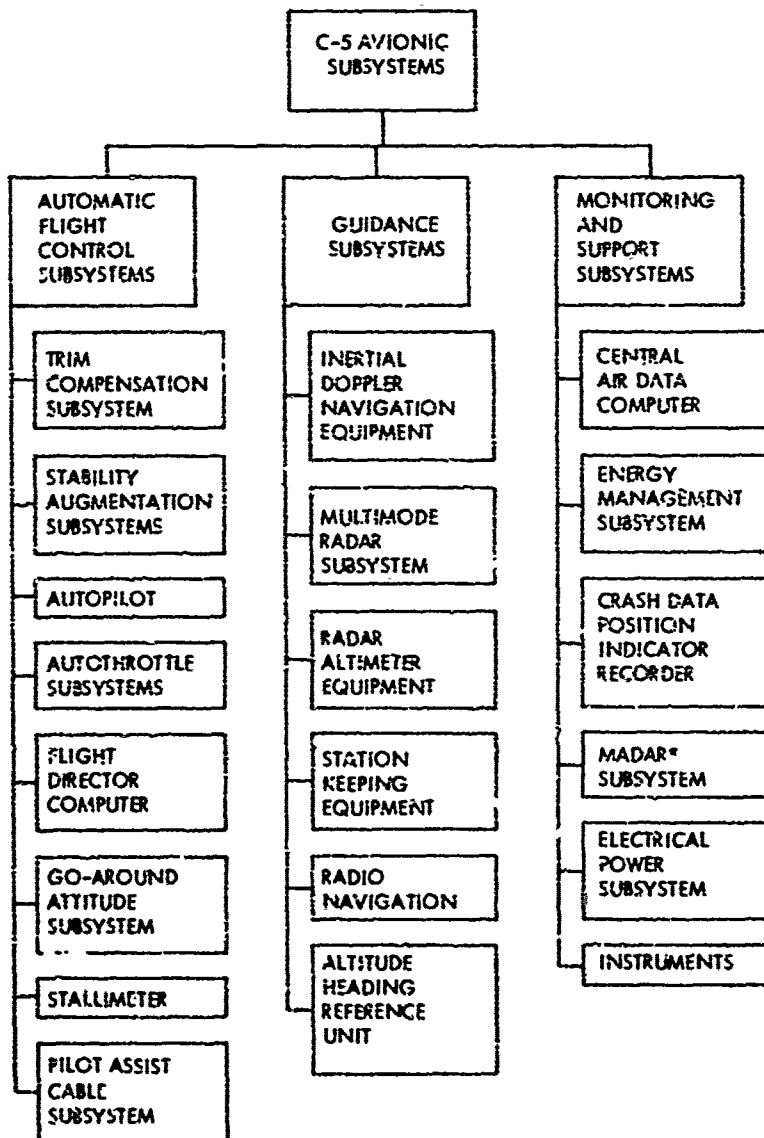


FIGURE 2.

* MALFUNCTION DETECTION ANALYSIS AND RECORDING

1. 1973-1974-1975-1976-1977-1978-1979-1980-1981-1982-1983-1984-1985-1986-1987-1988-1989-1990-1991-1992-1993-1994-1995-1996-1997-1998-1999-2000-2001-2002-2003-2004-2005-2006-2007-2008-2009-2010-2011-2012-2013-2014-2015-2016-2017-2018-2019-2020-2021-2022-2023-2024-2025

LONGITUDINAL STATIC STABILITY

CRUISE CONFIGURATION

SYMBOL	ALTITUDE - FEET
△	80,000
◇	26,000
○	24,000
⊙	10,000
□	35,000

OPEN SYMBOLS - TCS OPERATIVE
 SHADED SYMBOLS - TCS INOPERATIVE

DATA BASE: LGIT 19-1-10

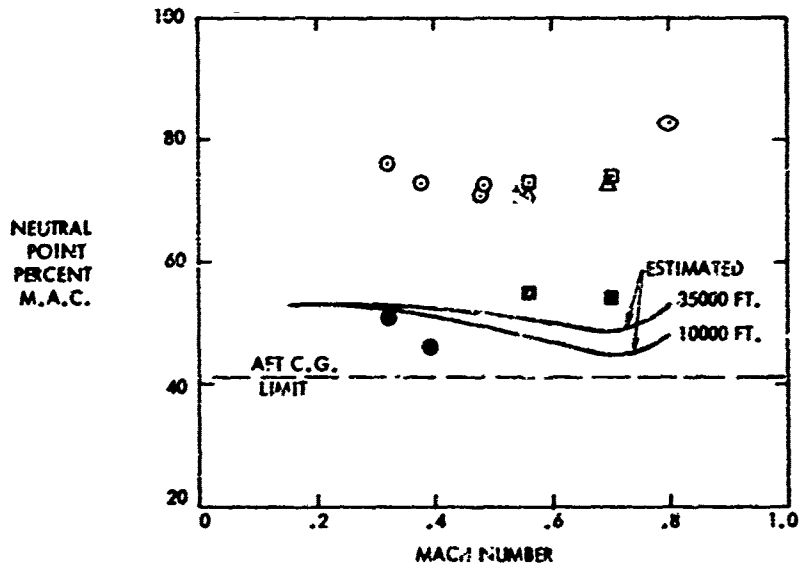


FIGURE 3. LOCKHEED C-5 AIRPLANE TRIM COMPENSATION SUBSYSTEM PERFORMANCE

SHORT PERIOD OSCILLATORY CHARACTERISTICS

SYMBOL	CONF	ALT FT	SPEED FPS
⊙	CR	20000	622
□	CR	40000	743
△	PWR	0	670
◇	PWR	20000	622
◇	LDG	0	218
△	APPR	0	246

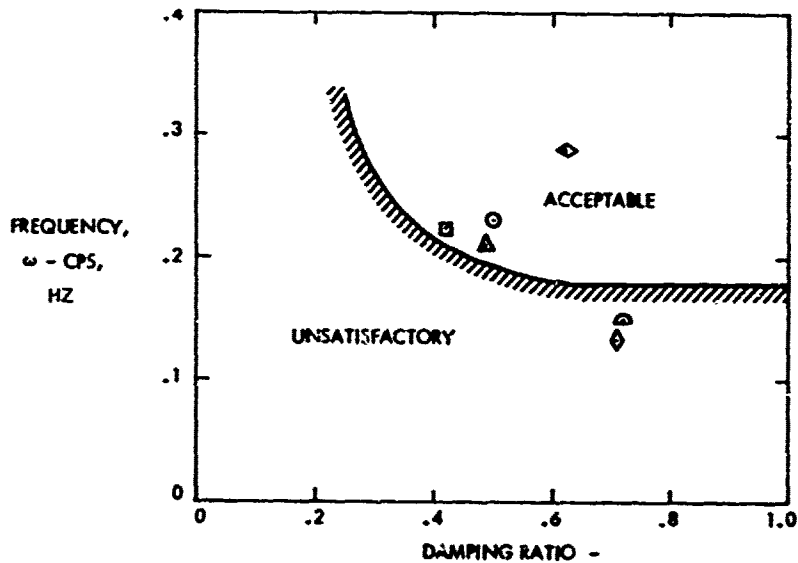


FIGURE 4. LOCKHEED C-5 AIRPLANE PITCH AUGMENTATION SUBSYSTEM PERFORMANCE

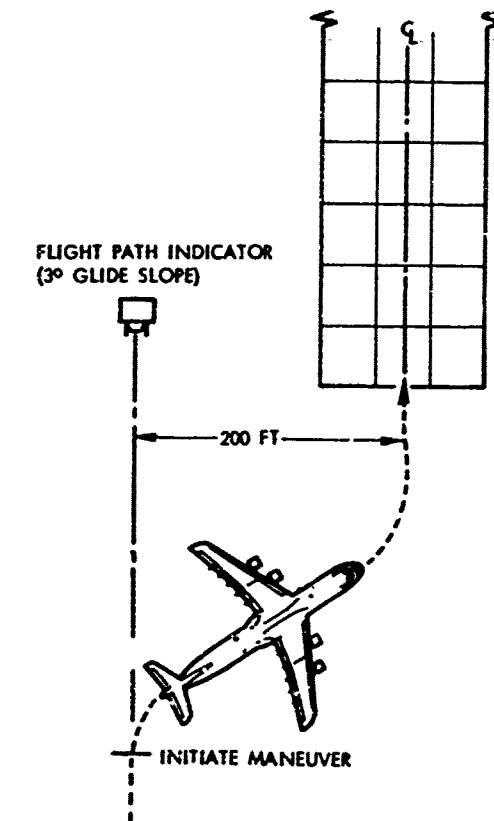


FIGURE 5. LOCKHEED C-5A AIRPLANE RUNWAY OFFSET MANEUVERS

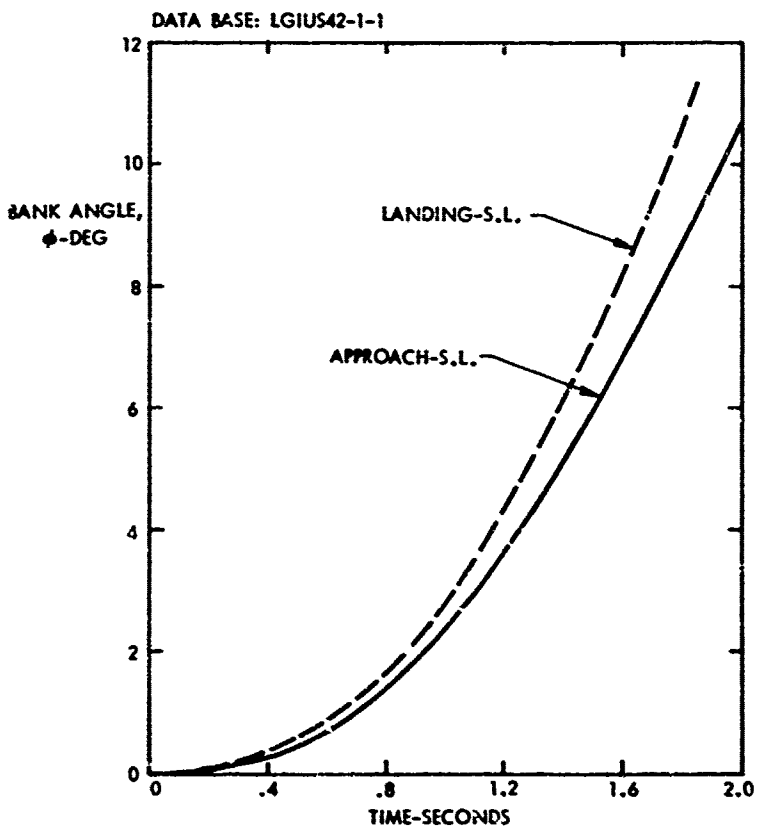


FIGURE 6. LOCKHEED C-5 AIRPLANE MAXIMUM ROLL RESPONSE

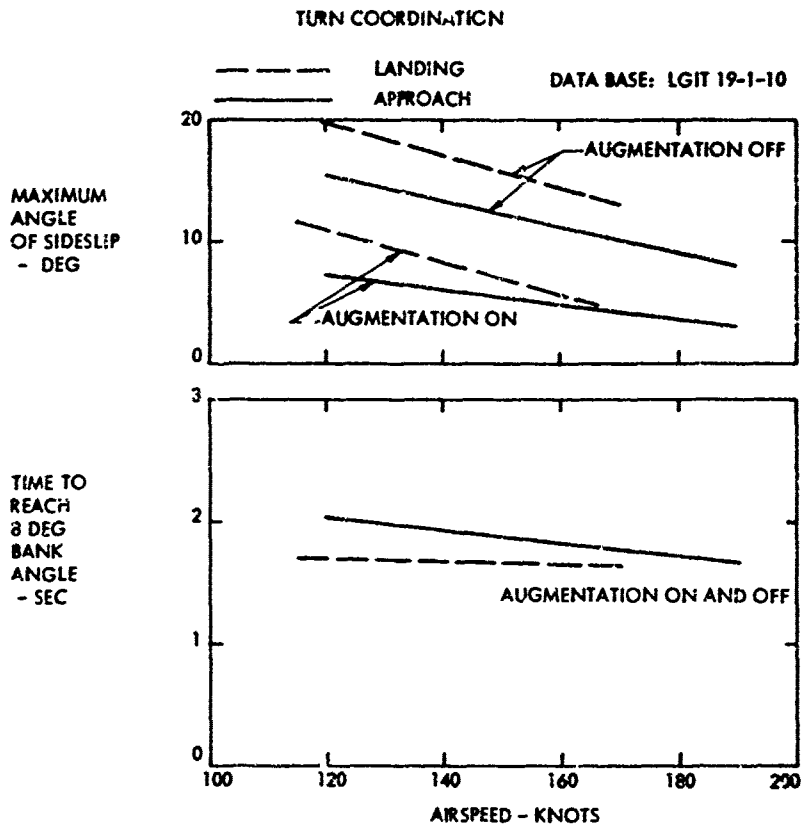


FIGURE 7. LOCKHEED C-5 AIRPLANE YAW AUGMENTATION PERFORMANCE

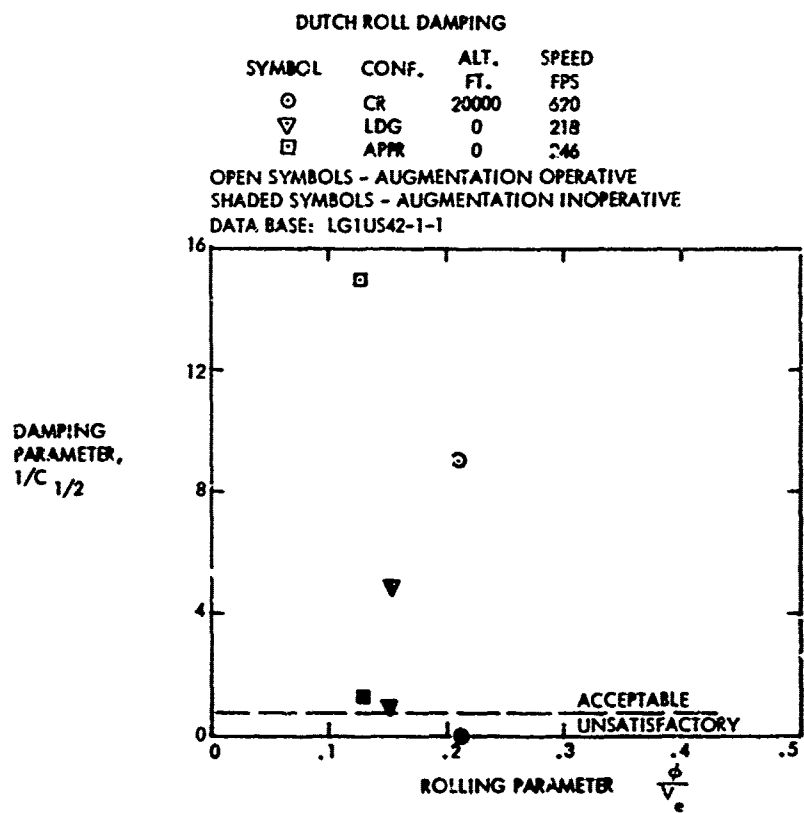


FIGURE 8. LOCKHEED C-5 AIRPLANE YAW AUGMENTATION PERFORMANCE

SPIRAL STABILITY

CONFIGURATION	CRUISE		LANDING		APPROACH	
SPEED - KNS	440		218		246	
WEIGHT - LB	450000		579000		579000	
ALTITUDE - FT	40000		0		0	
AUGMENTATION	OFF	ON	OFF	ON	OFF	ON
DUTCH ROLL DAMPING RATIO	.011	.709	.148	.854	.089	.455
PERIOD - SEC	10.1	7.8	11.2	4.0	11.1	11.8
SPIRAL TIME TO HALF AMPLITUDE T 1/2 - SEC	+136	-6.9	-23.5	+2.7	-40.1	+95.0

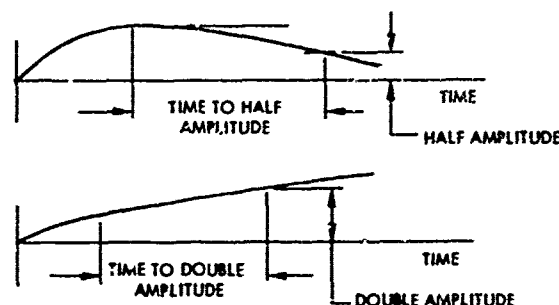


FIGURE 9. LOCKHEED C-5 AIRPLANE
LATERAL - DIRECTIONAL
AUGMENTATION PERFORMANCE

Decisions	Influencing Factors	Hindsight
<p>Trim Compensation Subsystem</p> <p>CX-HLS-No requirement PDP - Subsystem required for Mach tuck tendency DP - Subsystem required for Mach tuck and low speed static instability PROD - Subsystem deleted</p>	<p>System requirements</p> <ul style="list-style-type: none"> - Airplane safety - Handling qualities <p>Design experience Design techniques</p>	<p>Recognized that safety of flight dictates this trend because of</p> <ul style="list-style-type: none"> - hazards of transonic flight. - experience level of average pilot
<p>Pitch Augmentation Subsystem</p> <p>CX-HLS-No requirement PDP-Pitch response quickening DP-Pitch response quickening and short period damping PROD-Minimum short period damping during inflight refueling maneuver</p>	<p>System requirements</p> <ul style="list-style-type: none"> - Design criteria - Human factors <p>Design techniques Program schedule Design experience</p>	<p>Recognize requirement for adequate definition of mission requirements, design criteria, design and test time.</p>
<p>Yaw Augmentation Subsystem</p> <p>CX-HLS-Simple subsystem for dutch roll dumping PDP - Simple subsystem for dutch roll damping and turn coordination DP - Complex subsystem with multiple feedbacks for damping and turn coordination PROD - Some subsystem as during PD</p>	<p>System requirements</p> <ul style="list-style-type: none"> - Airplane mission - Design criteria - Human factors <p>Design techniques Program schedule Design experience</p>	<p>Recognize requirement for adequate definition of mission requirements, design criteria, design and test time.</p>
<p>Lateral Augmentation Subsystem</p> <p>CX-HLS-No requirement PDP - No requirement DP - Complex subsystem with multiple feedbacks for dutch roll damping and spiral stability PROD - Single feedback subsystem for spiral stability</p>	<p>System requirements</p> <ul style="list-style-type: none"> - Airplane mission - Design criteria - Human factors <p>Design techniques Program schedule Design experience</p>	<p>Recognize requirement for adequate definition of mission requirements, design criteria, design and test time.</p>

FIGURE 10. SUMMARY - EFFECT
OF DESIGN DECISIONS

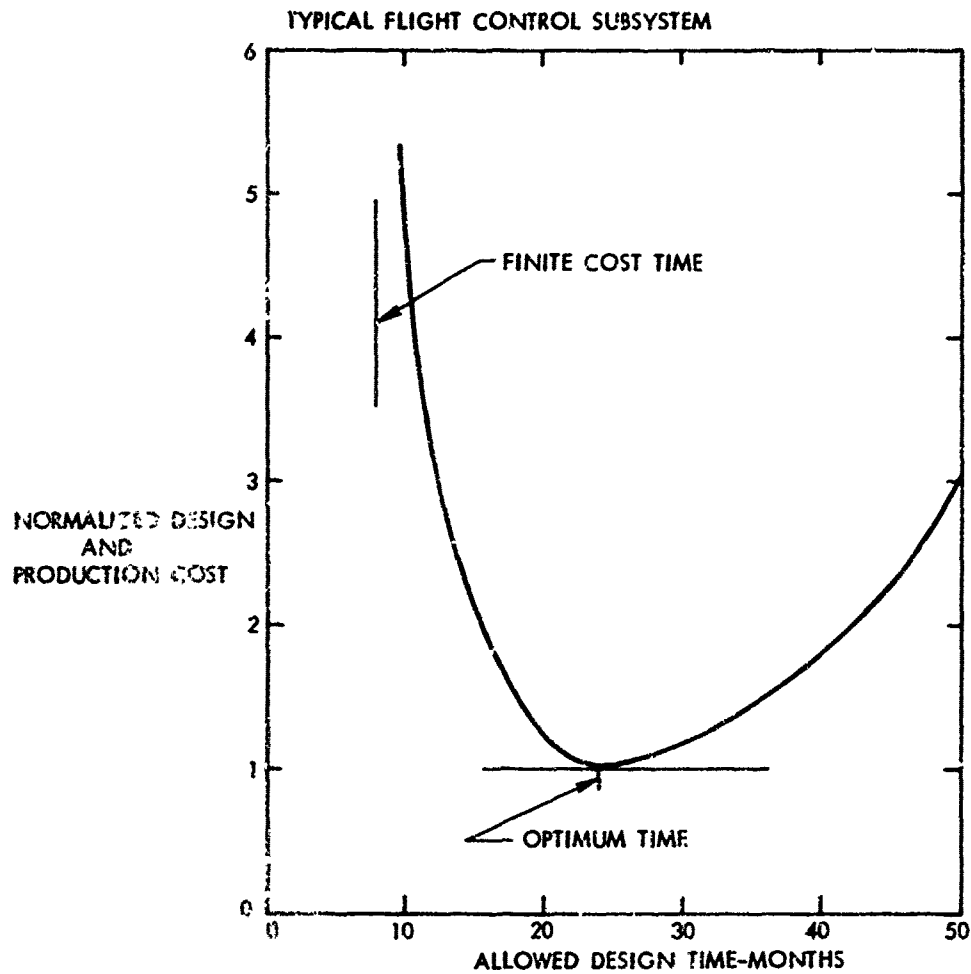


FIGURE 11. LOCKHEED C-5 AIRPLANE
EFFECT OF ALLOWED DESIGN TIME
ON COST

ADVANCEMENTS IN FUTURE FIGHTER AIRCRAFT

by

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SUMMARY

1. A new aircraft development can be justified if the performance of that new aircraft exceeds that of an old aircraft by at least 15 % to 20 %.
2. Foreseeable technological air frame advancements such as CCV and composites do not justify the development of a new weapon system, per se.
3. Recent engine technology advancements allow to achieve a level performance, which does justify the development of a new generation of air superiority fighters.
4. Foreseeable technological air frame advancements do, however, pay off if applied to a new aircraft development.

CLASSIFICATION OF ADVANCEMENTS

Forecasting technological advancements means forecasting future. History of such forecasting shows:

a) That the attempt to assess technological advancements far into the future, say 50 or 100 years, tends to underestimate realities. This is because the only technique of forecasting without falling into dreaming is extrapolation. Extrapolation, however, does not take into account innovations. Innovations which exceeded extrapolating forecasts by orders of magnitude, for example, have been the jet engine or the transistor. By definitions, advancements based upon innovations cannot be forecasted and thus remain the subject of dreams. Fig. (1) lists such advancements (1) which are not against the laws of physics and therefore might very well develop, however, we can not say when and how and whether such advancements pay off.

b) That the attempt to forecast short-term advancements, say 5 or 10 years, tends to overestimate realities. This is because such advancements are based upon existing knowledge and technology, however, they take money and the acceptance of a certain amount of risk to materialize. See right side of fig. (4).

- | | |
|--|--|
| • Silent aeroplane - to prevent disturbance to population | |
| • "Boomless" SST - to prevent disturbance to population | |
| • 50% reduced overall operating costs | |
| • Better safety - orders of magnitude improvement | |
| • Better reliability - orders of magnitude improvement | |
| • Aircraft built from far fewer parts | |
| • Autocontrol and computing devices using cheap plastics in stead of rare metals | |
| • Totally different way of moving through the air - gravity control | |
| | • Utilisation of thrust vectoring for STOL or VTOL high speed aircraft |
| | • Wake propulsion |
| | • Powered lift for high speed aircraft |
| | • Exotic fuels, high energy fuels |
| | • Laminar flow control |

Fig. (1) DREAMS

Fig. (2) POTENTIAL ADVANCEMENTS, NOT YET SATISFYING SOLUTIONS

There is a third group of advancements, which seems to fit into the group of short-term type advancements on the first view or for people who are not intimately familiar with them. Some of them, applicable to fighter aircraft, are listed in fig. (2). However, a careful investigation or even an attempt to build, for example, a wake propelled aircraft or a VTOL supersonic fighter would indicate that any known solution is not quite satisfying in order to really break through. Any forecasting of advancements in this area would be based on certain assumptions, for example the existence of a new thrust vectoring device for afterburned gases, light heat resistant materials, facilitated handling of toxic fuels, etc. It needs innovations on a lower level to introduce such advancements and therefore they still remain dreams even with a good chance of becoming reality.

Another group of advancements for fighter Aircraft is highly desirable from an operational point of view, as listed on fig. (3). For each of those items not even a large break-through can be expected under any assumption. Their record shows a gradual improvement with small steps, such as the use of BIT. Unfortunately, saturation characterizes the trend of these items, even deterioration in some cases. Again, it would need innovations to achieve improvements of any significant size which puts even this group into the dream classification.

- Electronic systems integration
- Reliability
- Safety
- Maintainability
- Sensors
- Low cost production, value engineering

Fig.(3) GENERAL AREAS OF DESIRABLE IMPROVEMENTS

ADVANCEMENTS

To improve existing aircraft	To justify the development of new aircraft
<ul style="list-style-type: none"> • Improved engine performance • Improved avionics • Installation of new weapons • Limited use of advanced materials • Improved flap system • Improved production methods 	<div style="border: 1px dashed black; padding: 5px; margin-bottom: 5px;"> Existence of new engine Extensive use of adv materials CCV </div> <ul style="list-style-type: none"> • Supercritical airfoils • Revolutionary new conf concepts

Fig.(4) CANDIDATE AREAS OF ADVANCEMENTS

Let us therefore turn back to the (b)-type short-term group of advancements, e. g. those advancements, which today could be realized with some risk provided the funds are available. Now we have to distinguish between such advancements, which are significant enough to justify a new development and those, which are just worthwhile to be applied to an existing aircraft according to fig. (4). Since this paper deals with future fighter aircraft only, we shall concentrate on the items of the right hand side of fig. (4) only asking whether these techniques really would be attractive enough to justify a new aircraft development.

Without going into details: supercritical airfoils do yield a performance advantage at a particular point of the flight envelope, however, at the expense of flight conditions off that design point. Overall, there may be some advantage for a fighter aircraft for which off-design mission performance is important.

Revolutionary new configuration concepts could very well lead to new developments as did, for example, the application of a variable sweep. Other candidates could be strake wing configurations or some of the new VTOL ideas which are under consideration. Very little, however, can be said about the use of ejector wings for VTOL operation before significant values of thrust augmentation have been demonstrated at realistic air frame weights.

The field of composite materials and control configured vehicle concepts lend themselves to a fairly reliable generation of data. Concentrating further on air frame technology the analysis will use these two areas as examples to answer the question posed above. In addition, recent advancements in engine technology are investigated with regard to their sensitivity to the analysis.

CONTROL CONFIGURED VEHICLES (CCV)

The term CCV concerns a wide range of design features as listed in fig. (5). In general, we are talking about CCV whenever an electronic system is used to improve the performance, flight characteristics or even some operational characteristics of an aircraft.

Biggest performance improvements can be demonstrated for relaxed longitudinal aerodynamic stability, which is to be replaced by artificial stability provided by an artificial stability system. In fig. (6) induced drag characteristic is plotted as a function of parametrically varied wing-tail configuration of a hypothetical fighter aircraft. It happens that the minimum induced drag applies to unstable configurations. Also, maximum achievable lift of various wing-tail combinations is limited by natural aerodynamic stability.

- Relaxation of static margin
- Relaxation of lateral stability
- Direct lift control, side force control
- Active flutter suppression
- Maneuver load control
- Gust alleviation
- Flight path control
- Push button flight management

Fig.(5) SCOPE OF CCV

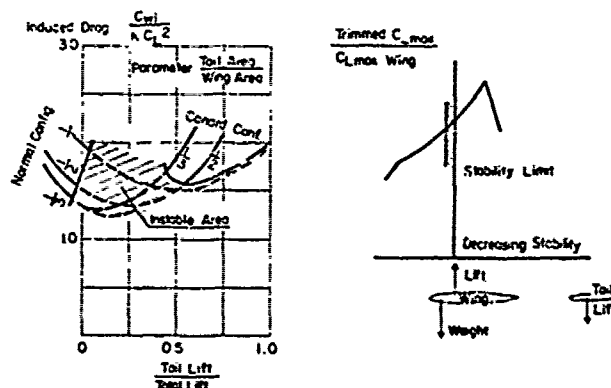


Fig.(6) CCV LIFT AND DRAG IMPROVEMENTS

The feasibility of drag and lift polar improvement is easy to demonstrate by proper combination of measurement points of wind tunnel tests (fig. 7). Improvements of such magnitude are not achievable by any other known means without weight penalties. Artificial stabilization of aircraft allows to reduce the area of control surface resulting in even greater performance improvements (fig. 8). For a typical fighter aircraft designed around a fixed engine some performance improvements are shown in fig. (9). The results indicate that the largest pay off is expected for negative stability margins in the order of 10% to 20% depending upon the particular configuration and the driving performance requirements. Generally, the high load factor requirements benefit the most from relaxed stability. In fig. (10) the potential performance improvement is expressed in terms of a reduction of overall aircraft weight at constant performance. Again, an instability of more than 10% margin is desirable to take full advantage of the CCV feature.

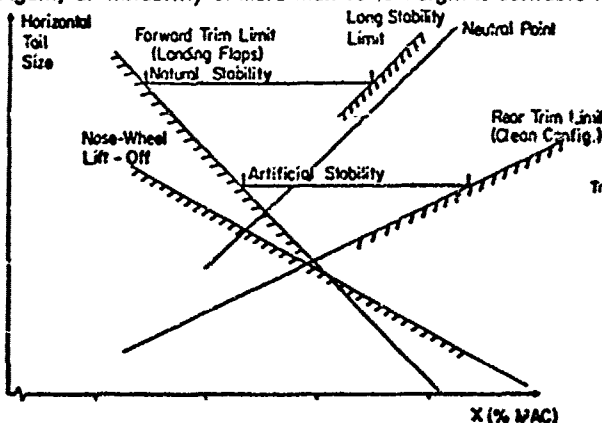


Fig. (8) CCV REDUCTION OF TAIL SIZE

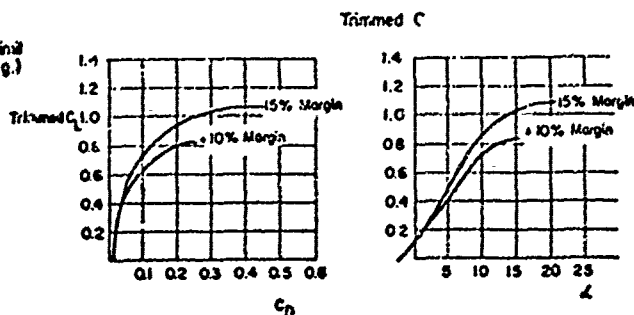


Fig. (7) WIND TUNNEL TEST RESULTS

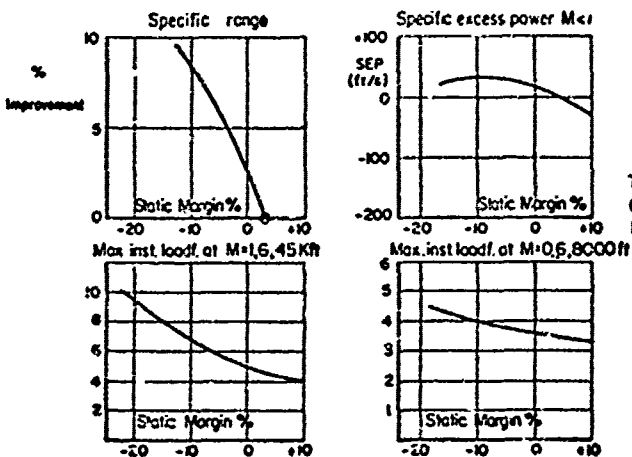


Fig. (9) CCV PERFORMANCE IMPROVEMENTS (FIXED ENGINE)

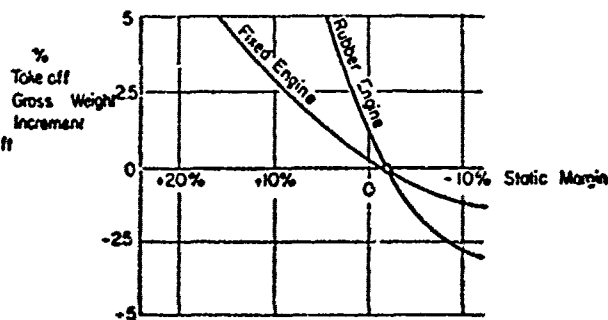


Fig. (10) WEIGHT REDUCTION BY CCV (CONST. PERFORMANCE)

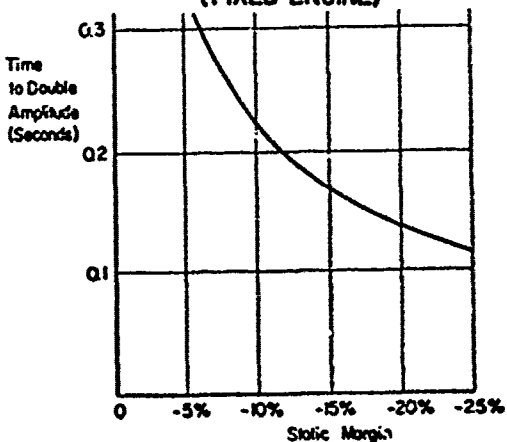


Fig. (11) NEED TO RELY UPON THE CONTROL SYSTEM

Such measures of instability create new demands in designing the control augmentation system. The reaction time of the aircraft movements following a control system failure is extremely short. Time to double the flight path amplitude as a function of negative static margin is shown in fig. (11) indicating that the human pilot could not react fast enough and the aircraft would be lost within a very short time. Thus, a new level of control systems reliability must be achieved if artificial stability is to come into being. Fig. (12) shows the criticality of the main subsystems relative to their failure rates and the redundancy level necessary to satisfy safety requirements equivalent to those of mechanical flight control systems.

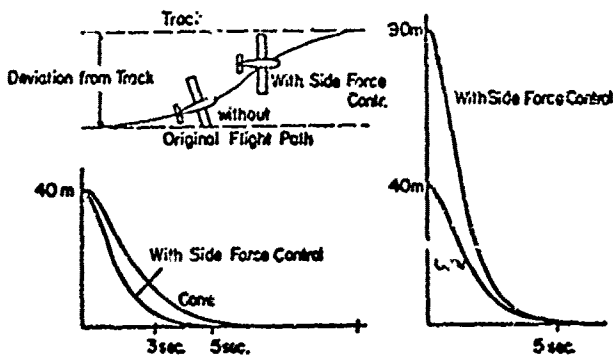
	Criticality Relative to Failure Rates	Safety Req. Accidents per Flight Hour	Necessary Redundancy
Sensors	1%	6 · 10 ⁻⁸	4 LINES
Signaling/Computing	4%		
Actuators/Hydraulics	35%		

Fig. (12) ALLOCATION OF CONTR. SYSTEM CRITICALITY CCV SAFETY AND REDUNDANCY REQU.

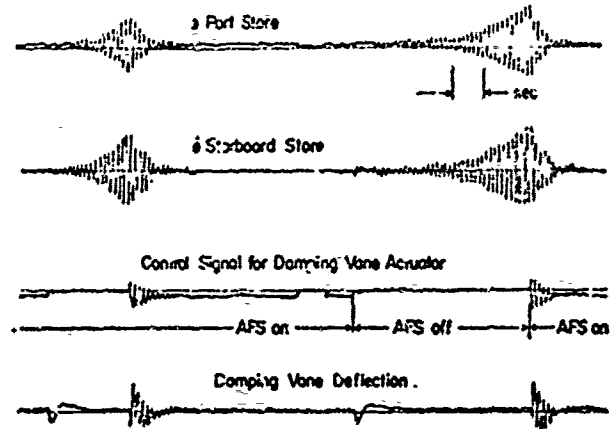
Fig. (13) illustrates the advantage of side force control, which is a CCV feature since effective side force control needs at least the assistance of an electronic control system. Using such capability a deviation from a desired flight path can be corrected without banking and rolling. That maneuver would shorten the time to achieve the flight path adjustment or would enable to adjust a larger deviation, within the same time respectively. Fig. (13) represents results of simulations of a typical fighter aircraft maneuver. Side force maneuvers are most likely to improve the accuracy of ground or air target attacks.

Electronic control devices also have been used to suppress flutter of transport wings. Significant structural weight savings could be achieved if the control devices could be relied upon. In fighter aircraft design flutter very often is a difficult problem for external loads. Artificial flutter suppression lends itself to tackle that problem. Fig. (14) shows results of wind tunnel testing using a wing-pylon-store combination of the MRCA aircraft. Beyond a particular speed and without the suppression system being switched on the pylon builds up large amplitudes. A simple circuit using the signals of an accelerometer and activating a small vane mounted at the nose of the store is able to suppress the oscillation within a few cycles. Because of the elastic interaction between wing and pylon the suppression system must also be installed at the wing in order to avoid its loss whenever the store is jettisoned. Expected weight savings by proper application of flutter suppression ranges in the order of 2% of take-off weight.

Another attractive CCV feature is maneuver load control, e. g. the unloading of outer wing areas by proper activation of flaps under maneuvering conditions or gust encounters. The immediate gain is an improvement of structural fatigue up to 100% and subsequently considerable structural weight savings at a given fatigue requirement.



TIME AND DISTANCE TO CORRECT
FLIGHT PATH
(Fig. (13))



WIND-TUNNEL DEMONSTRATION
OF ACTIVE FLUTTER SUPPRESSION
(Fig.14)

COMPOSITES

The application of composites to fighter air frames is simply a question of weight saving versus cost increase. Besides that there are other advantages of composites, such as stiffness and less corrosion. In some cases the use of composites even is a question of feasibility of the configuration. For example, rigid rotors for helicopter would not have been feasible without glass fiber or carbon fiber blades. High thrust for weight ratio fighter aircraft always suffer off c. g. problems which may call for composite tails.

However, the driving requirement remains the economy of composites. For fighter aircraft the higher cost of composites has to be recovered almost entirely by a benefit in procurement cost because fighter aircraft in peace time do not carry paying payload.

Fig. (15) shows the percentage breakdown of a typical fighter aircraft structural weight for various stages of maturity of composite application. Obviously, the weight reductions are significant.

	% of struct weight	Potential Weight Break down		
		Replacement	Redesign ^{*)}	Future ^{*)} Potential
Wing	26	12	13	8
Empenage	9	6	3	2
Basic fuselage	44	24	28	18
Second fuselage		10		
Control surfaces	5	3	*) according to (2)	
Other structural comp.	16	14		
	100	76		

POTENTIAL WEIGHT SAVINGS
BY COMPOSITES
(Fig.15)

Weight Saving	30%	
Weight Sensitivity	3	
Cost Sensitivity	200 \$/lb Procurement	
	300 \$/lb Life cycle	
Cost per lb of Structure	70 \$/lb AL	500 \$/lb Composite

$$\frac{\text{Expense}}{\text{Gain}} = \frac{0.7 \cdot 500 - 1 \cdot 70}{3 \cdot 0.3 \cdot 200 (300)} = -1.5 (1.0)$$

COMPOSITES ECONOMICS ASSESSMENT
(Fig.16)

Fig. (16) tries to emphasize the economic aspect of composite utilization in fighter aircraft: one pound of conventional aluminium structure, assuming a weight saving of 30%, could be replaced by 0.7 lb. of composite structure. A very optimistic cost figure for composite structure is 500 \$ per lb. as compared to 70 \$ per lb. for aluminium structure. Thus, there is a cost penalty of 0.7.500 = 1.70 \$/lb. associated with that replacement. On the other hand, assuming a weight growth factor of 3, there would be an overall weight saving throughout the entire aircraft of 0.3 (original weight saving) \times 3 = 0.9 lb. if the use of that composite component is planned from the very beginning. Each pound saved results in a reduction of procurement cost of about 200 \$ and 300 \$ of life cycle cost respectively. Thus the overall gain amounts to 0.9.200 = 180 and 0.9.300 = 270 respectively. This simple comparison indicates that the use of composites would pay off only under certain most favorable conditions. Such conditions apply to a percentage of structural weight much smaller than indicated in fig. (15). As a very approximate figure, a maximum of 10% composite utilization seems to be economically justifiable.

ENGINE TECHNOLOGY

The purpose of this paper is to analyse advancements in air frame technology. Engine technology is to be discussed only in terms of the effect on overall aircraft weight and performance. An interesting way of viewing the economy of engine technology is to compare the J 79 technology with current technology demonstrated (fig. 17). Such comparison comprises 20 years of engine technology development and a resulting improvement of engine thrust to weight ratio from 4.5 to about 8. At the same time cost per pound of engine thrust has increased from 20 \$/lb. to more than 40 \$/lb. assuming a comparable buy level and 1972 economic conditions. An aircraft utilizing the RB 199/F 100 technology would be 20% lighter but its propulsion system would be about 60% more expensive as compared to a fighter of similar performance which is designed around a J 79 technology engine. This comparison is based upon an aircraft overall thrust to weight ratio of 0.7 which is typical for today's fighters in operation. Finally, the procurement cost of both aircraft would be about equal.

	J 79 techn	Current Engines
T/W ratio	4.5	8.0
Cost per engine thrust	20 \$/lb	40 \$/lb

Constant T/TGW ratio of aircraft

Take off gross weight decrease 20%
Cost of propulsion system increase 60%

Cost of old engine
Technology
Aircraft

Air Frame	Air Frame
Propulsion	Propulsion
Equipment	Equipment

Cost of new engine
Technology
Aircraft

Fig. (17) ENGINE TECHNOLOGY ECONOMIC ASSESSMENT

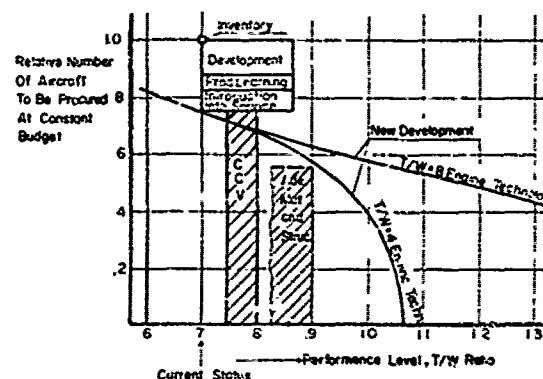


Fig. (18) a/c NUMBER PENALTY OF NEW DEVELOPMENT AND HIGHER PERFORM

The question arises whether any of the candidate technology advancements is likely to justify a new weapon system development. The foregoing comparison have been made for constant performance level, however, the history of fighter aircraft exhibits increasing or even escalating performance. In fig. (18) the cost of increasing aircraft thrust to weight ratio at equal technology is translated into numbers of aircraft to be procured at constant budget relative to the number of current performance status aircraft in the inventory. Even the introduction of a new weapon system of the same performance level would decrease that number because of the development and the putting-into-service cost of the new system. In addition, buying just more old aircraft would benefit from production learning which, in order to be fair, must be added to the cost of a new development as a penalty. Consequently, the curve in fig. (18) starts at about 0.75. At aircraft thrust to weight ratios significantly beyond 0.7 engine technology starts to become important.

As explained in fig. (19) aircraft gross weight and cost increases progressively with increasing aircraft thrust to weight ratio even approaching infinite values. The asymptotic aircraft thrust to weight ratio depends on engine thrust to weight ratio, e. g. engine technology. As a result the economic trade off outlined in fig. (17) would be unfavorable to the J 79 technology at aircraft thrust to weight ratios of, for example, 1.0. This establishes a good case for new engine technology for future fighter aircraft.

CCV and composites are also allocated in fig. (18). Cost penalty of CCV can be neglected, whereas cost penalties are pacing in case of composites. Obviously, extensive use of composites can only be justified if the existence of a fixed engine requires a decrease of structural weight regardless of cost in order to achieve a required performance level. Even in case of CCV one would have to demonstrate that a performance improvement of about 10% would be worth an overall reduction of fleet size of about 25%.

FUTURE MISSION REQUIREMENTS

The question about the value of performance improvement needs a definition and evaluation of mission requirements. Fig. (20) distinguishes between 4 basic missions:

- The bomber type mission requiring moderate T/W, high wing loading for optimum cruise and on the deck ride qualities and on increasing sophistication of avionics for survival and successful target attack. There is no question about the usefulness of that mission. Variable sweep has been a significant advancement combining the required mission capability with short field and loiter as well as self-defence capability. The trend for advancements lies in the area of avionics and weapons.
- The point intercept type mission asking for high acceleration (high T/W), moderate wing loading and advancements in air-to-air armament and guidance as well as fire control systems. The classic interceptor competes against the surface to air missile.
- Close air support can be satisfied with moderate T/W but needs low wing loading for self-defence. There is an unresolved question about the general lay-out (AX question) and the competition by the armed helicopter idea.
- Air superiority over the battle field. This role requires wing loadings much lower and thrust to weight ratios much higher than designed into current aircraft. Due to the progressive trend of weight and cost vs. thrust to weight ratio indicated in fig. (19), such a combination would require advancements in air frame design and would very much benefit from recent advancements in engine technology.

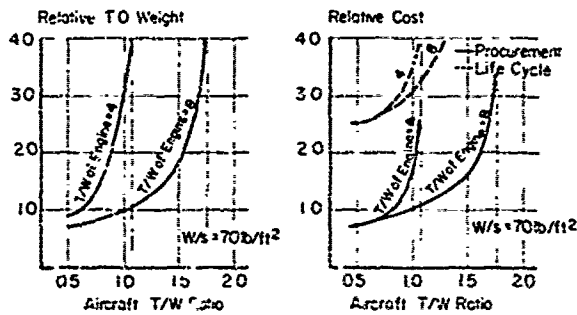


Fig.(19) ASYMPTOTIC PERFORMANCE LIMITS VS.ENGINE TECHNOLOGY

Mission	Techn Characteristics	Questionality
Deep Interdiction	Moderate T/W High Wing Loading Sophisticated Avionics	None
Counter Air	High T/W Moderate Wing Loading ADV Guidance System ADV A-A Missile	Aircraft VS Missile
Close Air Support	Moderate T/W Low Wing Loading	AX Problem Armed Helicopter
Battle Field Air Sup	High T/W Low Wing Loading	ser fig. 21
Battle Field Interdiction	Mod Soph Avionics	

Fig.(20) MISSION REQUIREMENTS

The importance of air-to-air capability is illustrated in fig. (21). According to the results of simulations, a large percentage of air-to-ground payload is lost due to air intercept by enemy fighter aircraft. This situation could be significantly improved by either sufficient self-defence capability built into the air-to-ground aircraft or by protecting the air-to-ground aircraft by fighter aircraft. Simulations indicate that the use of the same aircraft for both ground attack and air cover would be the most economical solution. Highly maneuverable future fighter aircraft would be able to develop an air umbrella over a limited battle area much more effectively as compared to current fighter which have not been designed for that purpose. Figures on the right hand side of fig. (21) are also the result of large scale simulations within the European scenario expected for 1985, indicating almost an order of magnitude difference in mission success. Conclusion: fighter aircraft could play an important role in future military conflicts and lend themselves to advancements in air frame and engine technology due to the high level of performance required to do their job efficiently.

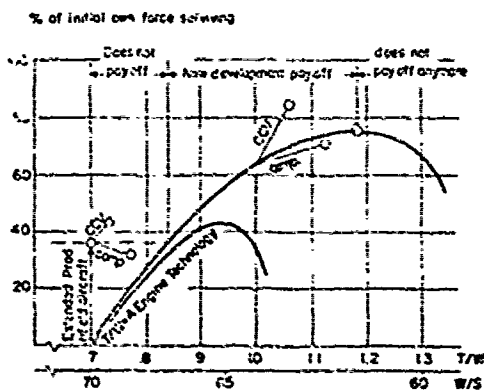


Fig.(22)

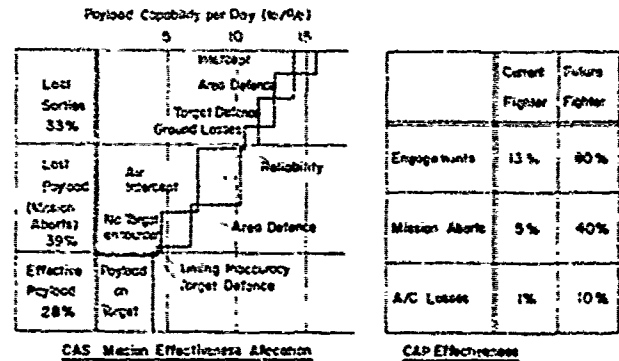


Fig.(21) IMPORTANCE OF AIR TO AIR CAPABILITY

Fig. (22) concludes the analysis addressing the question about the cost-effectiveness of air superiority fighter aircraft performance increase and about the performance margin which would justify the development of a new aircraft using current technology or advancements in air frame/engine technology. At a constant budget this means a trade of number of aircraft vs. individual aircraft performance to be evaluated in terms of fleet effectiveness. Fleet effectiveness is defined as the percentage of aircraft surviving an air battle after all enemy aircraft have been destroyed. If both sides operate aircraft of same performance having available the same budget, e. g. the same number of aircraft, such a battle would result into a complete deterioration of both fleets indicated by an effectiveness measure of zero in fig. (22). Performance increases along the horizontal axis of fig. (22) expressed by the T/W ratio-W/S combination: yielding a maximum of maneuverability. Consequently, the number of own aircraft decreases with increasing performance according to the constant budget assumption. Increased performance, even at less aircraft, pay off up to a maximum beyond which, because of the progressive trend of cost vs. performance, the low number of very high performance fighters start to lose their superiority against the large number of current performance enemy aircraft.

If an airforce decides to buy a fleet of old aircraft of the 0.7 T/W ratio type instead of developing a new aircraft, it could spend the development funds and the money to introduce a new weapon system to buy more old aircraft and it could benefit also from production learning. Thus, according to fig. (18), the size of the fighting fleet could be increased by 25 % which, in fig. (22) would result in a better fleet superiority equivalent to 35 % of surviving aircraft. A horizontal line from that point in fig. (22) cuts the fleet effectiveness curve at a point (T/W = 0.85 for new engine technology) which can be interpreted as a break-even point. Beyond that point a new fighter aircraft development starts to pay off.

A repetition of the calculation using a higher T/W for the enemy fighter fleet is not changing the maximum location of the fig. (22) very much. At T/W values of about 1.2 being equal for both sides it would be more cost efficient to build more aircraft rather than increasing the performance level.

How do air frame advancements, such as CCV and composites, fit into that scheme? (See small dashes)

- Improvement of the old aircraft: small improvement by CCV due to the fact that the additional cost is small compared to the potential performance increase. Deterioration of fleet effectiveness for composite structural component replacement.
- Application to a new development: significant improvement by CCV. Slight disadvantage by composites which might very well turn into an advantage in the near future as the cost of raw materials decrease.

The result of this analysis must not be generalized in specific terms. It is limited to the assumptions about the importance of air-to-air combat of the dog fight type. However, in general the question will always exist about how much quality improvement would justify a new development and how much it would pay off at all as compared to a quantity concept. In this sense the analysis is to be viewed as an example.

SUMMARY

1. A new air superiority fighter aircraft development can be justified if the performance of that new aircraft exceeds that of an old aircraft by at least 15 % to 20 %.
2. Foreseeable technological air frame advancements (such as CCV and composites) do not justify the development of a new aircraft, per se.
3. Recent engine technology advancements allow to achieve a level of performance which does justify a new development⁺⁾
4. Foreseeable technology air frame advancements do, however, pay off if applied to a new aircraft development.

+) Battle Field Air Superiority

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ESTIMATION OF PROGRAMMES AND COSTS FOR MILITARY AIRCRAFT

by

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SUMMARY

The paper sets out to provide a brief and general presentation of the purposes, history and methods of Budgetary Estimation for UK military aircraft programmes. The derivation of the methods will be discussed using the airframe as the main example. Finally, the use of the timescale, resources and cost estimating techniques to provide cost trade-offs for different aircraft operational capabilities will be demonstrated.

1 INTRODUCTION

Budgetary estimates of the resources (manpower, facilities, time and money) likely to be needed for the development and production of new aircraft are required within the Air Systems Controllerate (ASC) Procurement Executive, Ministry of Defence (MOD(PE)), for three main purposes.

- 1 Planning and funding (Long Term Costing (LTC)) purposes
- 2 Project selection, appraisal and approval, and
- 3 Assessment of the future loading on the aircraft industry for Aerospace policy decisions.

The aim of this paper is therefore to present briefly what is involved in, and the methods used for, making budgetary estimates and then give an indication, because of the special interest of this symposium, of how they can be used for cost and performance/effectiveness trade-off assessment. Although estimates are made for all aspects of an aircraft's procurement, because of the time available, the examples given concentrate on fixed-wing airframe development.

In order to make comprehensive cost appraisals, life cycle costs, rather than procurement costs, are required but this is a major area for future work.

2 HISTORY

In the 1950s and early 1960s, there was a succession of 'cost' escalation and 'overruns' in UK military, and for that matter the relatively more simple civil, aircraft procurement programmes.

The reasons for the past failures of estimates to predict the total expenditure reasonably accurately have been identified as:-

- i Changes in details of the Requirement, causing changes to specification
- ii Changes in technical possibilities giving greater potential "requirements"
- iii Basic underestimation of the time and cost needed to achieve identified task
- iv Incomplete identification of all the tasks involved in achieving the specified project, and
- v The effects of inflation.

A "factor of three" was commonly quoted in the case of development for the ratio actual cost to the initial estimate, but because of the inflation effect (v) acting over the development timescales of 5-10 years, the inflation factor alone averaged about 1.5, so that in constant prices the cost escalation factor was in effect two not three. The timescale overrun factor was about 1.3 and failure to estimate the timescale accurately contributed significantly to the shortcoming of the initial estimate.

Points (iii) and (iv) were areas which could be corrected by proper "resource" analysis of the necessary and appropriate content of the programmes taking account of the influence of the project's technical and operational characteristics.

Responsibility for points (i) and (ii) does not lie with the estimator, but with the control applied by the operational requirement and management function of the project development.

As a result of the growing realisation during the early 1960s that the estimation "system" was inadequate, points (iii) and (iv), a new unit called "Project Time and Cost Analysis" (PTCA) was set up in early 1964 to collect as much data as possible on aircraft projects and their programmes and to analyse these to provide methods of predicting future development costs and timescales. As these data would be "actuals" they would contain also the effects of (i) and (ii). The initial work was to be concentrated on airframe development. In 1965, the then Director General Engines (DG Eng) initiated similar work on aero-

engine development and this activity was absorbed into PFCAN in 1969. Also during 1965, the Director General Aircraft Production (DGAP) set up two groups to deal with budgetary production estimation, one on airframes (DAP Costs) and one on engines (Eng P Costs). PFCAN has a co-ordinating role to bring together airframe, engine and equipment production estimated costs for project selection, appraisal and approval purposes. During the last year or so work within PFCAN has been widened to devise estimating methods for aircraft equipment which it is hoped, will ultimately include aircraft avionics and flight control system equipment. The three groups; PFCAN, DAP Costs and Eng P Costs work in close harmony and currently cover fixed and rotary wing aircraft for both military and civil projects.

3 DEVELOPMENT COST ESTIMATION

3.1 General

In order to improve the budgetary estimating methods which had been in use prior to 1964, it was believed that the "cost" of future projects could only be predicted with reasonable assurance by reference to the actual "cost" of a wide spectrum of previous projects. The first important step which had to be taken at that stage was to initiate the collection from past records, if available, of the actual expenditures in terms of time, resources and money etc. on past aircraft projects. This was, and still is, a most difficult task because of the wide variations in the quantity, quality and definition of the data which had been recorded. However in spite of the difficulties, many useful data have been, and still are being, slowly accumulated by the diligence, hardwork and enthusiasm of the "investigators".

The first method adopted was to establish the total cost of the airframe development and production for previous projects. In the case of the development costs extreme care had to be applied in the analysis and all cases had to be adjusted to a common standard. This was because the basic data covered a variation of such factors as the number of aircraft used in the development programme, whether they were prototype, development batch, or production aircraft, the number of flying hours required to clear the aircraft depending on the number of roles or range of flight envelope over which they were designed, and the degree of R & D which had been completed before the go-ahead was given for the operational aircraft. When the "actual" development costs were adjusted to a common standard, it was possible to relate the adjusted costs to the characteristics of the aircraft. (Figure 1) However, it would be very fortunate if a future aircraft project aligned itself with the "norm" so the data of Figure 1 can only be used as a "basis" from which the new estimate can be made and so this method has shortcomings.

To a lesser, but still important degree, the same considerations applied to production costs. (Figure 2) For the same aircraft characteristics, weight speed or Mach. No. there may be differences in complexity and therefore production cost, for example, a small highly manoeuvrable combat aircraft with a complex nav/attack system compared with say a two-seat trainer with a minimal equipment fit.

The analysis technique which was developed and is in current use is to break down the total activity into a number of smaller elements selected on the basis that they were clearly identifiable and definable and that data in terms of costs, resources and timescales could be readily collected and linked to characteristics of the aircraft which are available from the very early stages of the project's conception. Typical percentage breakdowns for airframe and engine development are given in Figure 3, and for production in Figure 4.

3.2 Resumé of Current Methods for Development Cost Estimation

Time does not allow description of the method of estimating each item in detail, even if only airframe development is considered, so it is proposed to illustrate the estimating procedure by considering two of the major aspects which account for approximately 60% of the development costs, i.e. the design effort involved in the development and the manufacture of the development aircraft.

3.2.1 The Development Programme

The estimating process starts by defining the development plan. An example is given in Figure 5. This is important for two reasons, it decides the framework against which the estimates of the sub-groups are made, and decides the timescale over which the costs are spread. Development is a succession of the "basic" three component cycles, design, make and test; then re-design, modify and re-test, etc. The two main features required therefore are first the resources to design and make the required test aircraft, and second, adequate overall flight test capability (aircraft, flight test instrumentation, technical personnel, etc) to develop and prove the aircraft. During the first phase, the design, make and test cycle proceeds in parallel with the main aircraft stream in certain of the sub-groups, Figure 5, for example, wind tunnel tests, rig tests and structural tests. The overall timescale for the programme can be initially derived from data such as those given in Figure 6. This suitably adjusted for the project under review, allows an initial appreciation to be made as to whether the timescale stated in the Air Staff Requirement (ASR) is realistic, assuming the "normal" aircraft development process is used. It also provides an indication of a realistic period for the flight development programme.

Historical data have been collected and analysed which give the following information:-

- 1 Flying rates achieved by broad aircraft types, i.e. Bombers, Fighters/Combat aircraft, Transports. Flying rates (hrs/aircraft/month) both for the development flying and the official flying at the proving establishment (A & AEE).
- 2 The flying hours by both the firm and the proving establishment to achieve both the initial CA Release and the full CA Release.
- 3 For certain of the aircraft (and the list is growing) the number of hours required for the major test packages by discipline eg. handling, performance, engine/intake behaviour, engine handling, nav/attack, communications, weapon release, autopilot and automatic flight control system.

- 4 The number of aircraft used in the programme and whether they are prototype, development batch or production.
- 5 The rate at which the test aircraft can be introduced into the programme.

From the above data it is possible to construct a realistic flight test programme and this, combined with using other data available on for example, tunnel testing, allows an initial overall development programme adequate for budgetary estimating purposes to be defined. Obviously as the project becomes more defined and the project is assessed in much greater detail, this initial programme may be changed, but it is believed that providing the aircraft requirement is not altered these adjustments have a relatively small effect on the costs and can be allowed for by a contingency addition in the budgetary estimate. At the present time the aircraft system procurement process is calling for major changes to existing aircraft in order to lengthen their Service life by updating their operational capability. From the budgetary estimators' point of view, this is a more difficult task than estimating for a brand new aircraft but data such as that obtained against point 3 above, ie. flying hours required for different disciplines, hopefully allows realistic estimates to be made.

3.2.2 Manufacture of Test Aircraft

The cost of manufacturing the development airframes is the item which dominates the airframe development programme. (Figure 3) This cost is based on an assessment of manhours required for either the 100th unit or the average over the first 100 units. Production manhour historical data are more reliably established down the learning curve, Figure 7, than for the early units because batch manufacture and modification effects on these early units make their actual manhours deviate more from the average line. Throughout an aircraft production programme the unit labour manhours continuously decrease, in some cases down to a basic unit value, and the variation of unit labour manhours as the aircraft are produced defines the learning curve. Initially, such data were simply plotted against the product of empty weight and speed, or on a carpet showing the generation effect. (Figure 8) There is evidence that for a given type of aircraft, interceptor fighter, successive developments over the years call for an increase in the manhours per lb. This reflects the increase in aircraft capability and complexity which has taken place and which is known as the generation effect. Recent work, using regression analysis techniques, has aimed at providing a prediction formula, the parameters of which give the least dispersion of actual data from prediction. (Figure 9) These two examples, Figures 8 and 9 are good demonstrations of the main techniques appropriate to the budgetary phase, namely the "near neighbour" and "parametric". The other important input into the estimating process is the exercise of technical judgement, and this is required to adjust the results of the analyses of the historical data, to allow for the characteristics of the new projects, for example, "technical" complexity. This means that it is essential, in my view, that the programme assessment is made by engineers/scientists with development or production experience. We are not as once described "Financial Wizards".

The manhours for the first aircraft are calculated by using the slope of the learning curve to obtain a factor (see Figure 10) by which to multiply the estimated manufacturing manhours appropriate to the 100th unit or to the average of the first 100 units. The manhours for the subsequent development aircraft are calculated assuming reduced learning by 5-10% to allow for the installation of instrumentation and experimental equipment. The manhours can be converted to cost by multiplying by the known or predicted wage and overhead rates. To derive the aircraft cost the cost of raw materials, bought out equipment and the equipment developed specifically for the new aircraft must be added. Obviously these methods are equally applicable at the budgetary phase for production aircraft cost estimation.

Fairly simple "rules of thumb" are used to obtain the cost of such things as airframe spares and modification costs as these are in general a relatively small percentage of the total and even significant errors in their estimation only have a second order effect on the total estimated development costs.

3.2.3 Design Costs

The next major item in the airframe development cost is "design" and this embraces not only the draughtsmen and loftsmen, but all the other technical effort involved in the design and development of an aircraft, stressmen, aerodynamicists, electronics engineers, structural and armament specialists etc. For many years prior to the formation of PICAn, ID Plans, the group on which PICAn was formed, had gathered comprehensive data of the direct and indirect design effort allocated to projects. This has allowed analyses of the total effort involved and also the "spread" during the life of the project up to full CA Clearance, see Figure 11. The analysts of these design manhours have allowed prediction formulae to be derived, an example is given in Figure 12 where it is compared with an earlier attempt using simply the product weight and speed as the parameter. It is interesting to note, as would be expected, the reduction in scatter from using the regression analysis technique. These design manhours are turned into design cost by multiplying by the known or predicted design wage rates. The manhours are for aircraft, all of which, when the design was initiated, required advances in the "technical state of the art", over that then currently known and the prediction is therefore essentially for an "average" increase in the "state of the art". The estimator has therefore to exercise his engineering judgement, after discussion with his "technical expert" colleagues, to adjust the estimated design manhours if the new aircraft is considered to reflect a substantially greater or smaller than 'average' step forward in the "state of the art". The importance of obtaining a good assessment of the technical development involved cannot be overemphasised and calls for "teamwork" between the estimating groups, the project and weight analysts in Headquarters and the specialists in the Procurement Executive research establishments.

3.2.4 Summary and the Future

Our method for breaking down the overall task into elements and the availability of the historical data and the method by which it can be used for cost assessment have been given in the previous sections. In addition to the historical data for aircraft manufacture, flight programmes and design, data are available on instrumentation, wind tunnel tests, structural tests, mock-ups and technical publications. In the case of other items, for example rig tests, ground test equipment, in which significant estimation errors

would affect only about 10% of the total cost, further data are still needed to improve the estimation of these elements.

The result of the analysis described above provides the "expected" or "most likely" development cost. Variations of technical difficulty, requirement changes, management proficiency and firms' capability have caused the actual data to be spread about the average. This enables a "statistical uncertainty" to be derived for new projects which are in the same category as those in the data sample. This "statistical uncertainty" can be expressed as a cumulative probability as suggested diagrammatically by Curve 'A' in Figure 13, which shows how the chance of success increases within a given cost as that cost increases. Curve 'B' demonstrates the risk which would be associated with a new project outside the previous range of experience, for example, the first supersonic or V/STOL civil transport. This is shown further to the right because of the higher likely cost and has a lower slope, even in percentage terms, because of the inherently greater uncertainties. It is in this difficult area of "risk" or "uncertainty" analyses that further work is required and which if successful will provide management with valuable extra guidance.

3.3 Engine Development

Similar methods and techniques to those described for the airframe are used for predicting engine development costs. The data and methods for estimating the bench engine test hours are still being developed, being related to various parameters such as turbine entry temperature, pressure ratio, by-pass ratio and thrust to weight ratio. Figure 14 illustrates the trend. It is believed that the derivation of these relationships has made a marked improvement in the estimating capability because they provide a factual basis to which the effect of new technology can be added. The need to establish a realistic estimate of the bench hours is important as this largely determines the timescale of the engine development programme, and as Figure 3 indicates approximately half the cost is incurred during bench testing.

3.4 "Proof of the Pudding"

It is early days yet to obtain a true measure of the accuracy of the methods which have been developed in the last four years and described briefly in this report, but it is encouraging to note that the estimates made using the methods of the mid-60s which formed the basis from which the current methods have been derived are indicating accuracies within 10-15%, but it must be appreciated the sample is still small.

4 "COST TRADE-OFFS"

4.1 General

Initial attempts have been made recently to use the budgetary estimating techniques to indicate cost trade-offs when aircraft with varying operational capability and broad specification requirements are compared. There are in the author's view three main ingredients required to make cost trade-offs, see Figure 15.

- 1 Comprehensive analysis of the operational situation by the "operators" and their analysts to provide the different aircraft or aircraft capability which should be considered.
- 2 Detailed project analysis to formulate these operational proposals into aircraft which can be defined in terms of performance and design standards taking into account the technological "state of the art" which could be applicable within the development timescale; and
- 3 The cost estimation of the different proposed technical and operational solutions.

It is believed that on many occasions this whole process will be re-iterative and call for teamwork involving close and constant liaison between the three major groups involved. I must emphasise that the important part of the process lies with the operational and project groups to define the aircraft parameters, standard of equipment, roles etc. which must be costed, although the cost estimating group from its general experience of development will have an input into this process.

4.2 Demonstrations of Cost Trade-offs

Comprehensive cost trade-off studies should be based on Life Cycle Costs, but this approach as far as we are concerned must be in the future. Therefore, in order to demonstrate the current cost trade-off capability Figures 16 and 17 have been prepared to show the trends in total development and unit production costs respectively. The development costs include airframe, engine and avionics and the unit production cost is for the complete aircraft averaged over the intended production numbers.

Because the aircraft type used in this example is a battlefield support aircraft, the trends are shown plotted against, what is believed to be an important operational parameter, namely take-off distance. In addition to the effect of take-off distance the variations in costs shown in Figures 16 and 17 stem from the effect of:-

- 1 Reducing the design diving speed
- 2 Decreasing the avionics systems fit
- 3 Decreasing the weapon load
- 4 Increasing the range
- 5 Increasing Specific Excess Power (SEP)
- 6 Developing the aircraft using an engine which requires no further or very little development, and

7 Using an aircraft with STOL capability in the CTOL regime.

The analysis of the particular aircraft concept taken as the example and given in Figures 16 and 17, shows the expected results as follows:-

- 1 Reducing take-off distance, increasing range, increasing design diving speed, increasing SEP or using a V/STOL configuration into the CTOL regime all increase the development and production costs.
- 2 Using an engine already developed or needing only minor development, reducing the avionics standard or weapon load reduces the development and production costs.

In order to demonstrate the results of the analysis for the example aircraft used, total procurement cost, ie. development and production combined, was considered and for the assumptions made the following very approximate total procurement/trade-off was deduced:-

- 1 For take-off distance between 4,500 and 1,000 ft the total cost increases about 5-10% for every 500 ft reduction, with an extra step of about 20% at the threshold of STOL at about 2,000-2,500 ft where it is necessary to fit lift or vectored thrust engines.
- 2 A reduction in design diving speed from 750 to 650 knots decreases the total cost by about 20%.
- 3 A reduction of SEP by one third decreases the total cost by about 20%.
- 4 For CTOL aircraft, doubling the radius of action increases the total cost by about 10%, and the increase for this is likely to be greater for V/STOL aircraft.

Obviously the trade-offs given above apply only to the particular aircraft cases considered and for the total programme assumptions used. Each trade-off analysis will provide its own answer.

5 CONCLUSIONS

The methods for the budgetary estimation of the resources (manpower, facilities, time and money) required to develop and produce aircraft in the UK have been developing since the mid-1960s. It is still a growing field and the scope is such that in this paper it has only been possible to show briefly the methods currently in use, using fixed wing airframe development as the example. The technique is to estimate the 'resources' needed not for the development or production taken as a whole, but by summing realistic major individual packages into which the task can be divided. Historical data have been and are still being acquired on which to base the estimating methods. This gathering process is the difficult time-consuming part of the work.

The aim of the budgetary estimator is obviously to achieve accurate estimates and this, where there is a significant degree of technical innovation in the development process, depends to a large extent on thorough and realistic technical appraisals of the difficulties which will be involved in turning attractive new ideas into Service hardware. This means that there is required in the estimation process an important contribution from the people involved in the preliminary design process.

One of the main uses in the future to which budgetary estimating methods can be put is project selection. The ability to make cost trade-offs for varying technical and operational requirements should provide the Services and Procurement system, with a valuable 'indicator' for defining future aircraft standards which can be afforded. In order to fulfil this capability the estimation of cost trade-offs is heavily dependent on first, the Services defining, as a result of their operational studies, the operational trade-offs, and second, the project analysts (aerodynamic, structural, system, avionic) defining technically the aircraft systems etc. which should be coated.

6 ACKNOWLEDGMENTS

The Author wishes to acknowledge the valuable assistance and co-operation given to him by his many colleagues in the preparation of this paper.

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7 NOTE

The views expressed in this paper are those of the Author and do not necessarily represent those of the Department.

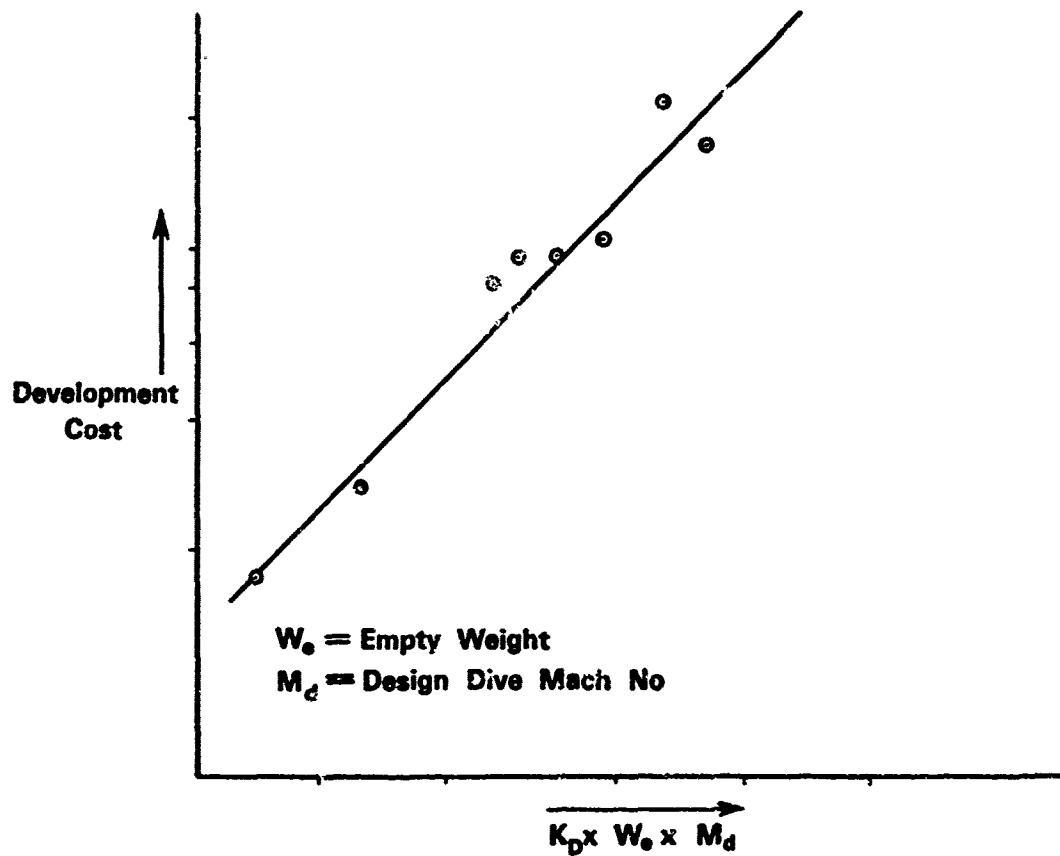


Fig.1 Military airframe development cost

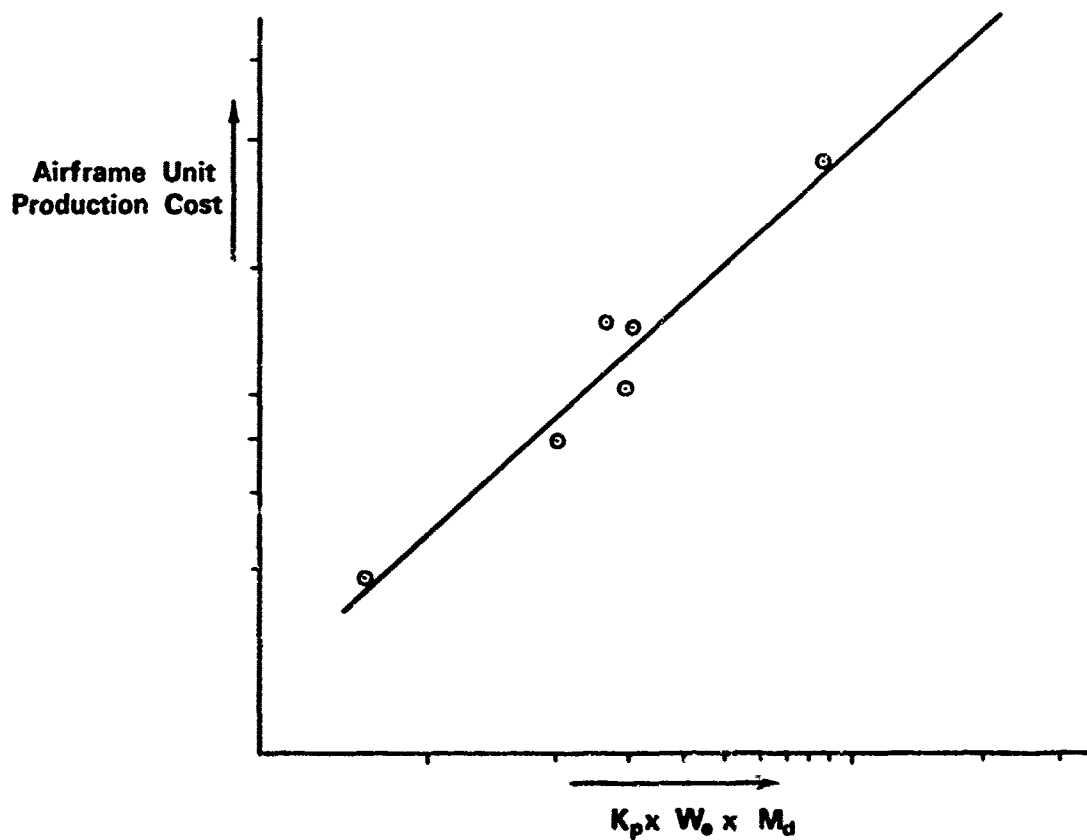


Fig.2 Military airframe production cost

**Airframe development
(To Stage 1 C.A. release)**

	%
Design	16
Jig and tool provision	10
Wind tunnel tests	3
Rig Tests	3
Structural tests	3
Structural test airframe	5
Mock-ups	2
Development airframes	40
Instrumentation	1
Flight trials and escort aircraft	2
Airframe modifications	5
Airframe spares	5
Ground and test equipment	2½
Airframe parts for engines etc.	1
Technical publications	1½
Total	100

**Engine bench development
(To type approval)**

	%
Engineering	25
Tool manufacture	10
Rig supply and test	10
Bench engine supply	20
Fit, test and fuel	10
Bench engine test parts	<u>25</u>
Total	100

N.B. In addition to the above there will be the costs of engine supply and support for the aircraft flight test programme and the costs of further bench development beyond type approval; typically for these items, 2½ x bench development cost needs to be added to arrive at total engine development cost.

Fig.3 Typical cost breakdowns

Airframe production

	%
Labour	41
Raw material	10
Bought out parts	21
Weight growth	7
Contingency/rounding	5
Productionising drawings*	3
Production tooling*	<u>13</u>
Total	100

Engine production

	%
Basic cost	60
Learner	16
Modifications growth	4
Contingency	4
Production tooling*	<u>16</u>
Total	100

**The cost of these items per unit produced will vary inversely almost directly with the numbers produced.*

Fig.4 Typical cost breakdowns

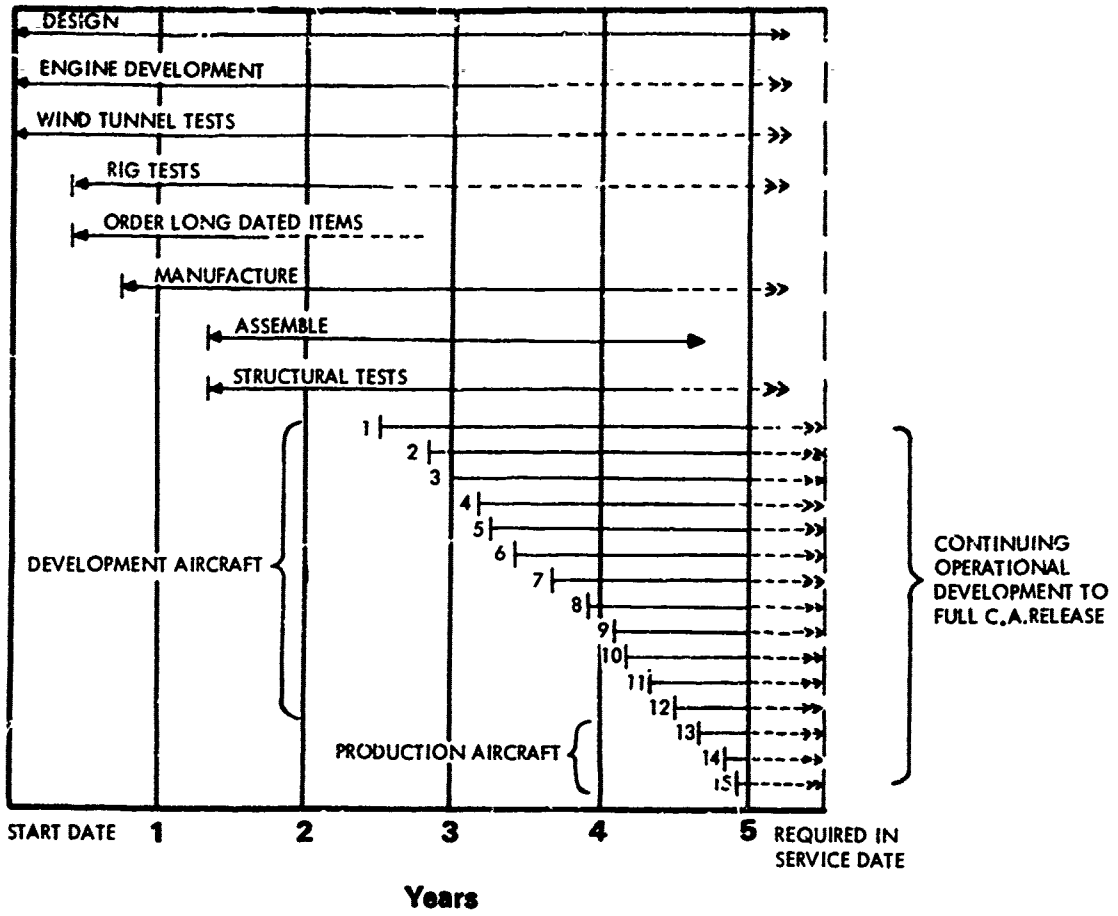


Fig.5 Aircraft development plan

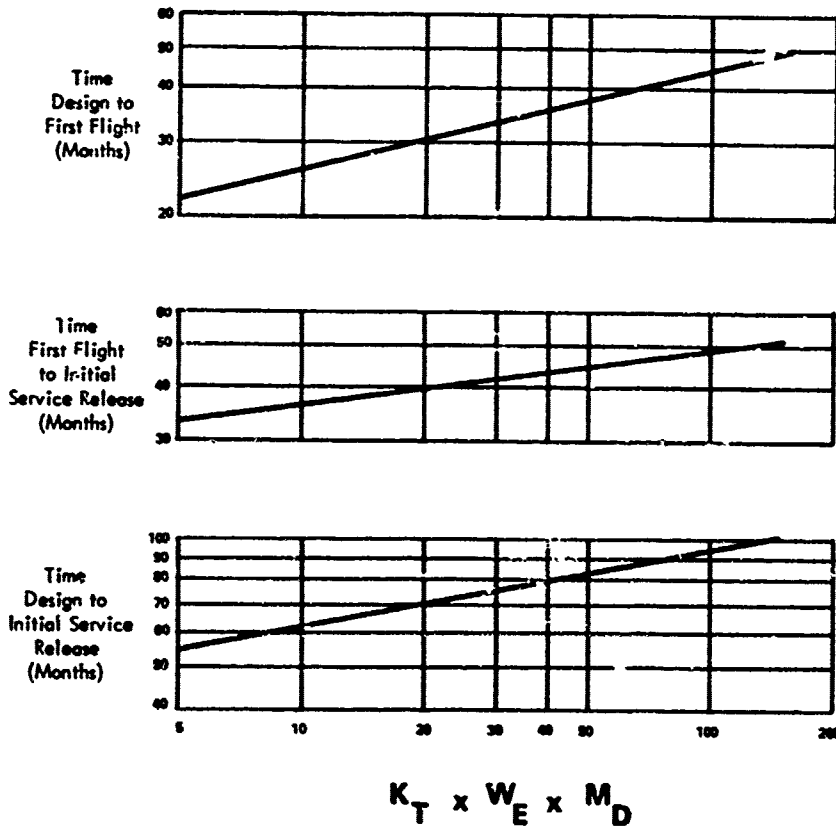


Fig.6 Development timescale military aircraft

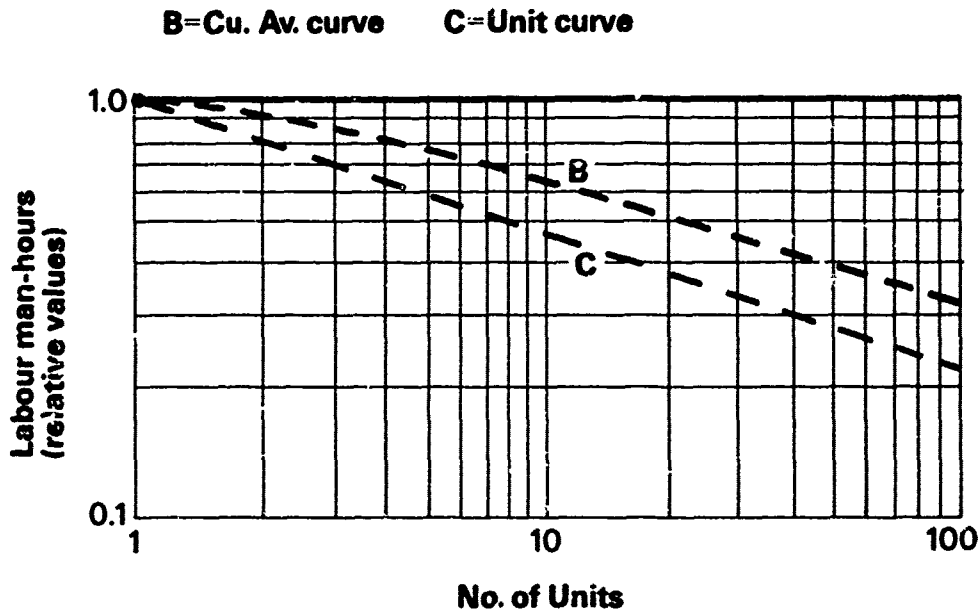


Fig.7 Manufacturing labour learning curve

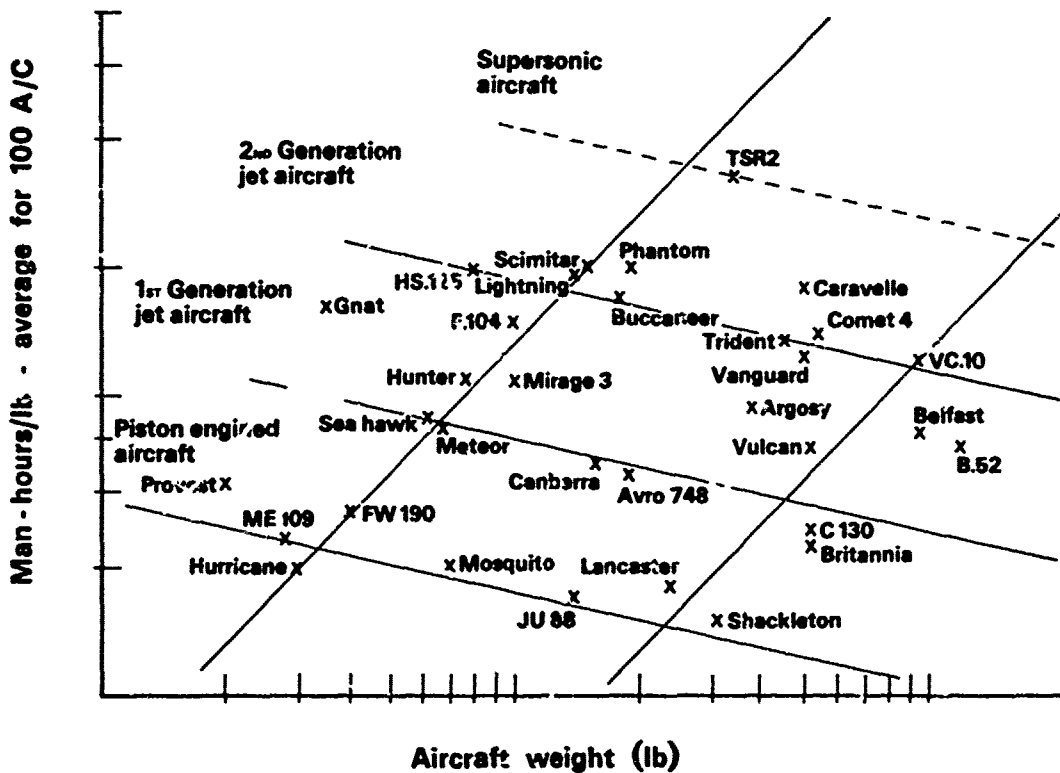


Fig.8 Manufacturing data

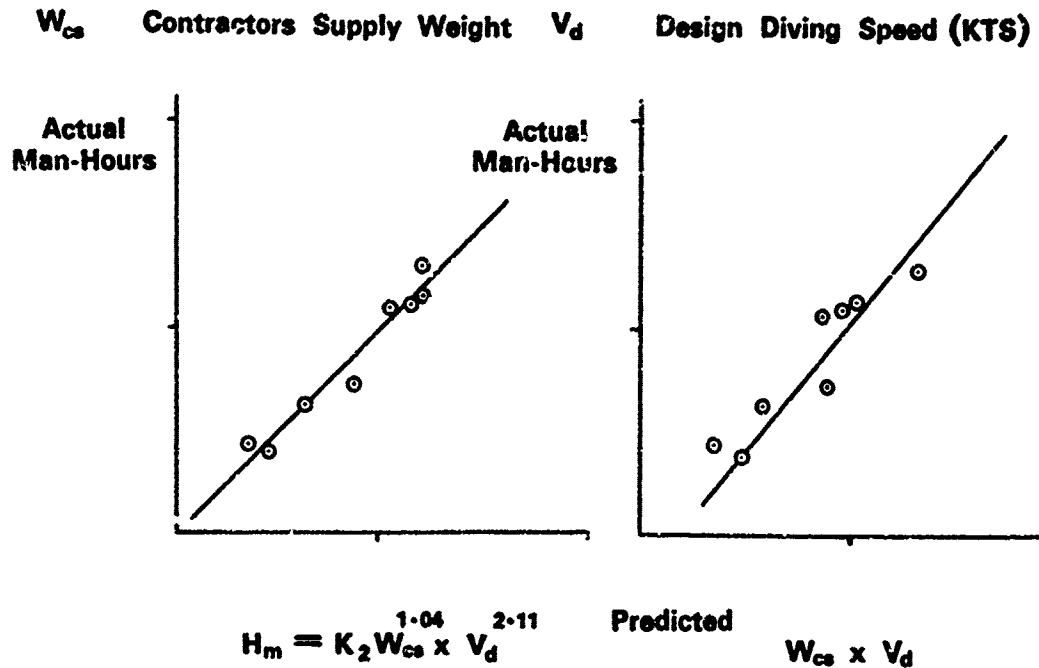


Fig.9 Airframe manufacture man-hours (H_m)

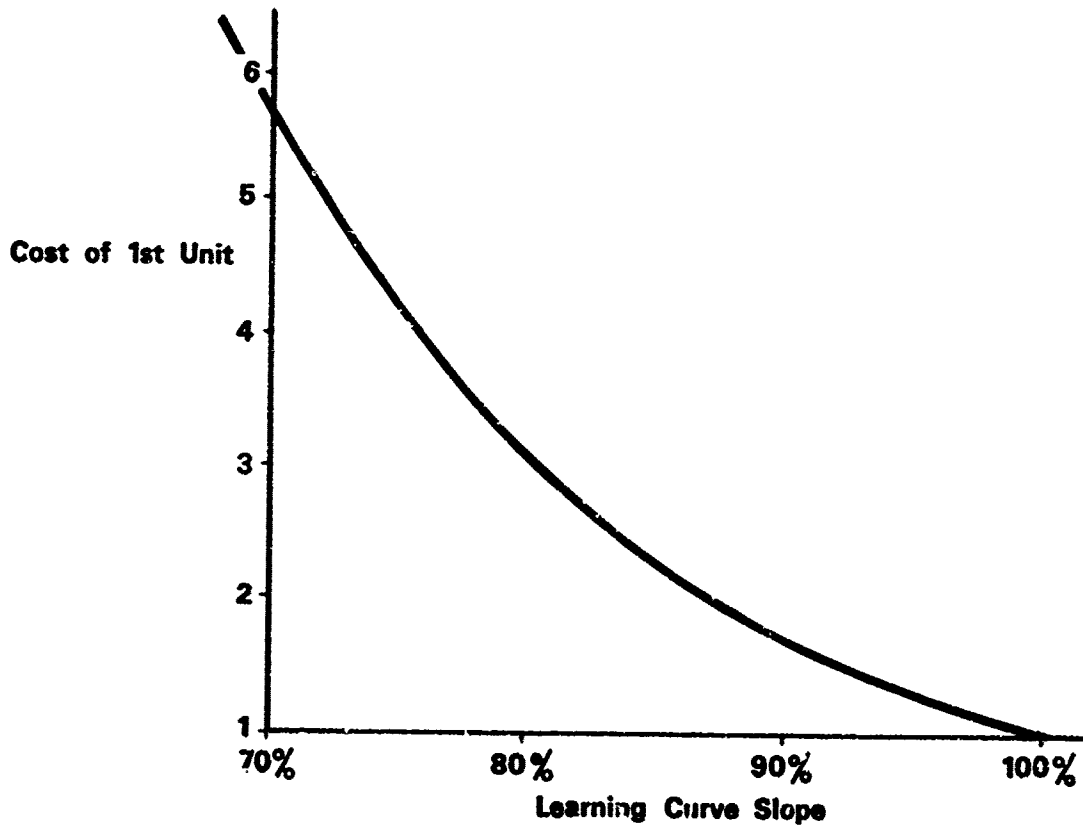


Fig.10 Cost of first unit derived from cost of average of 100 units

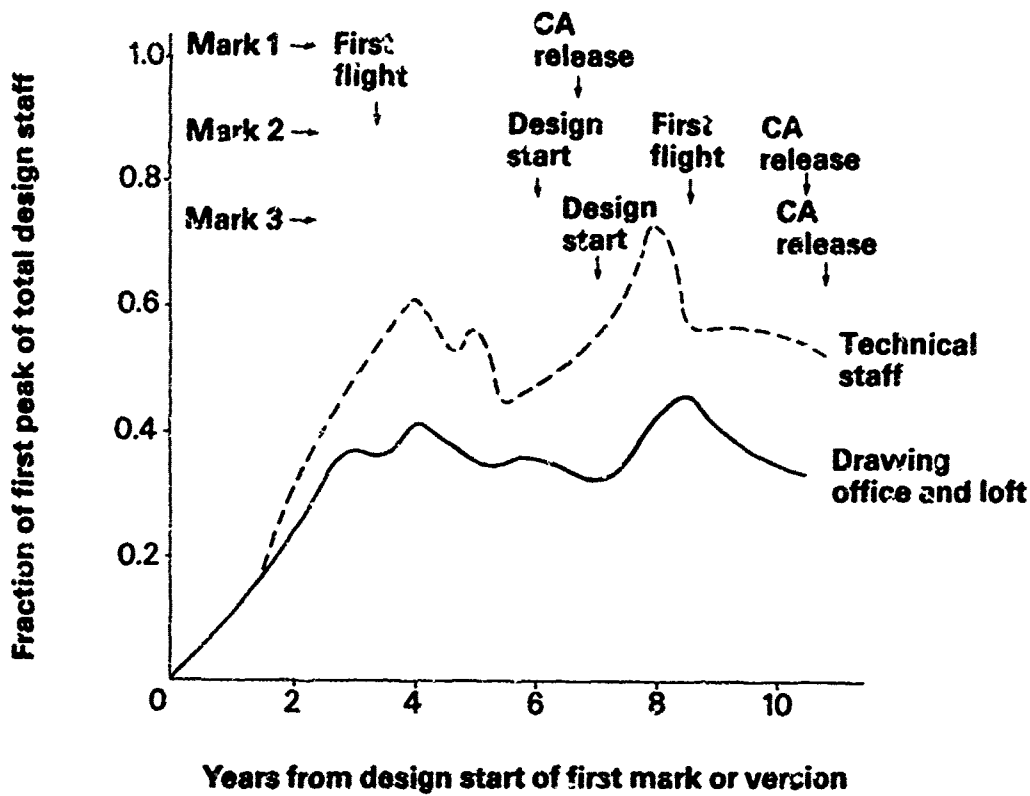


Fig.11 Design manpower – typical military aircraft

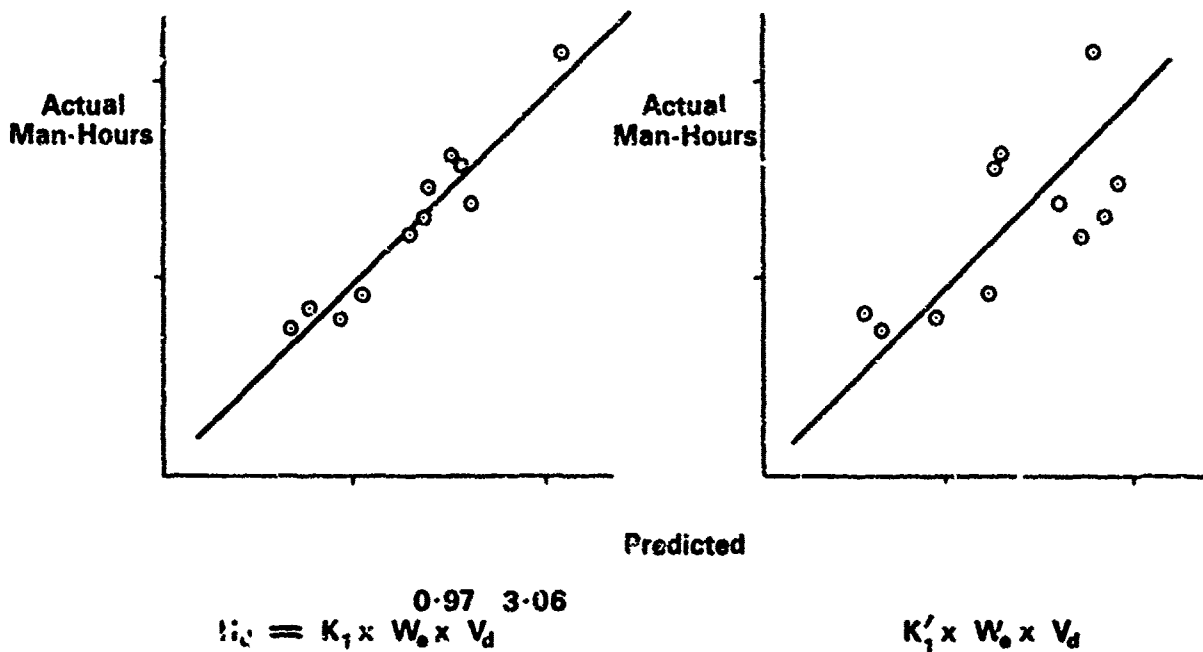


Fig.12 Airframe design man-hours (H_d)

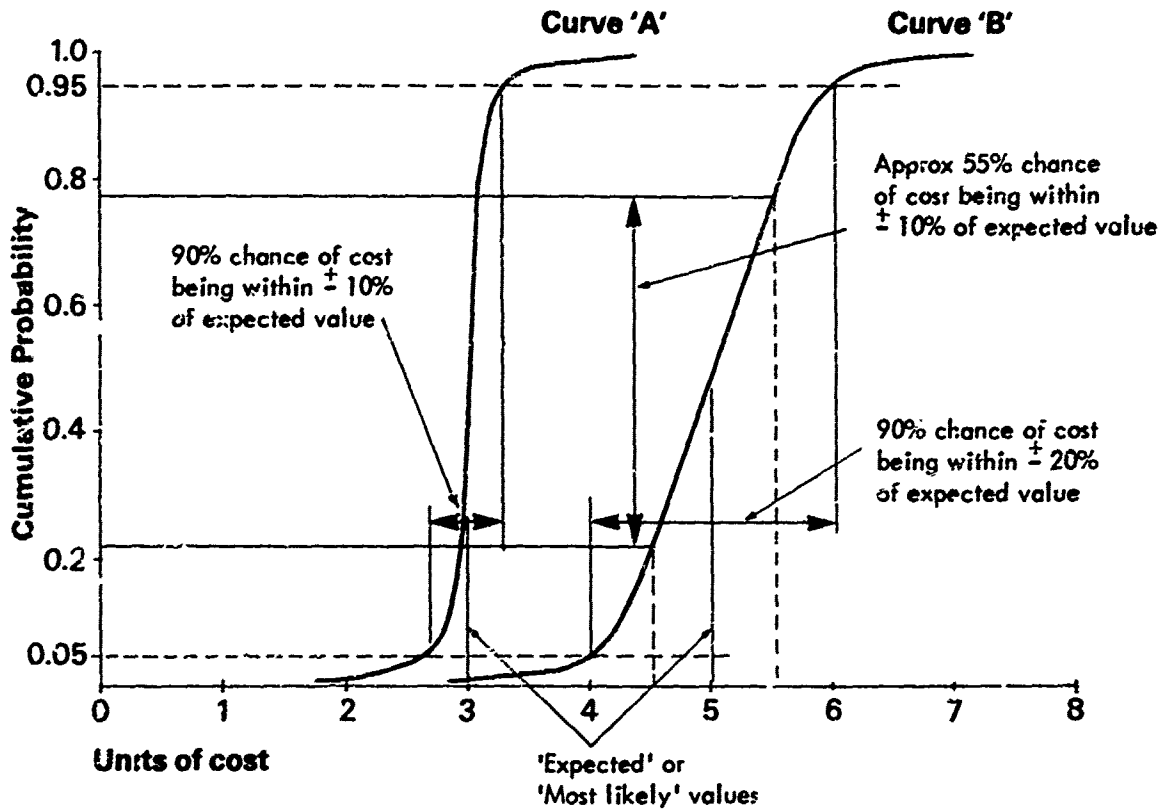
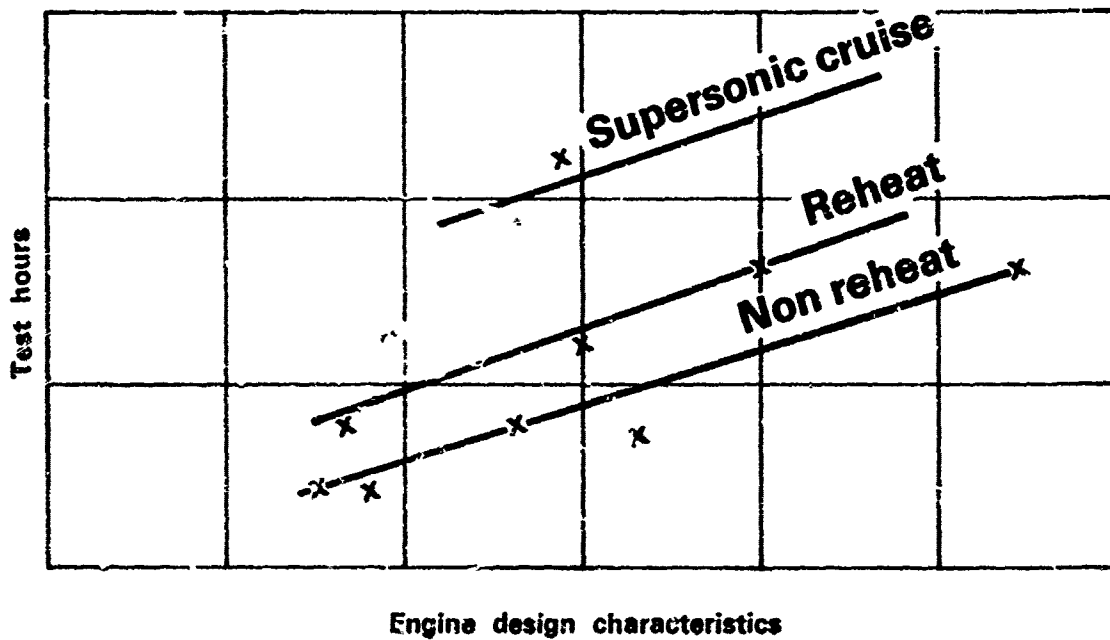


Fig.13 Statistical uncertainty



Engine design characteristics

Fig.14 Engine bench test hours

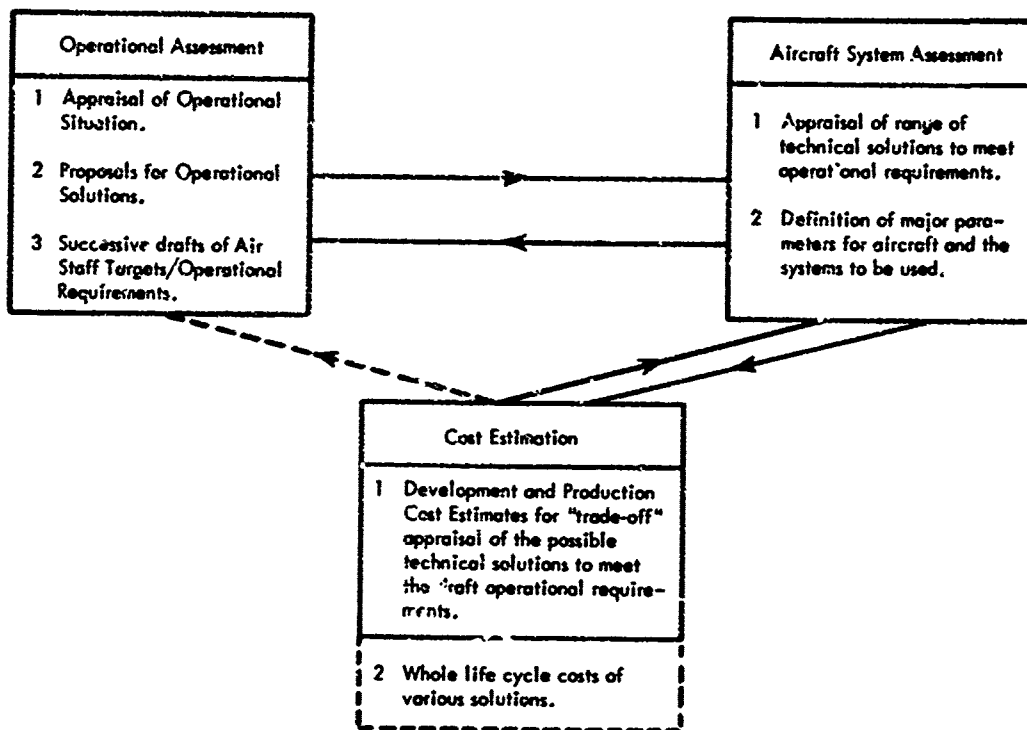


Fig.15 Cost trade-off process

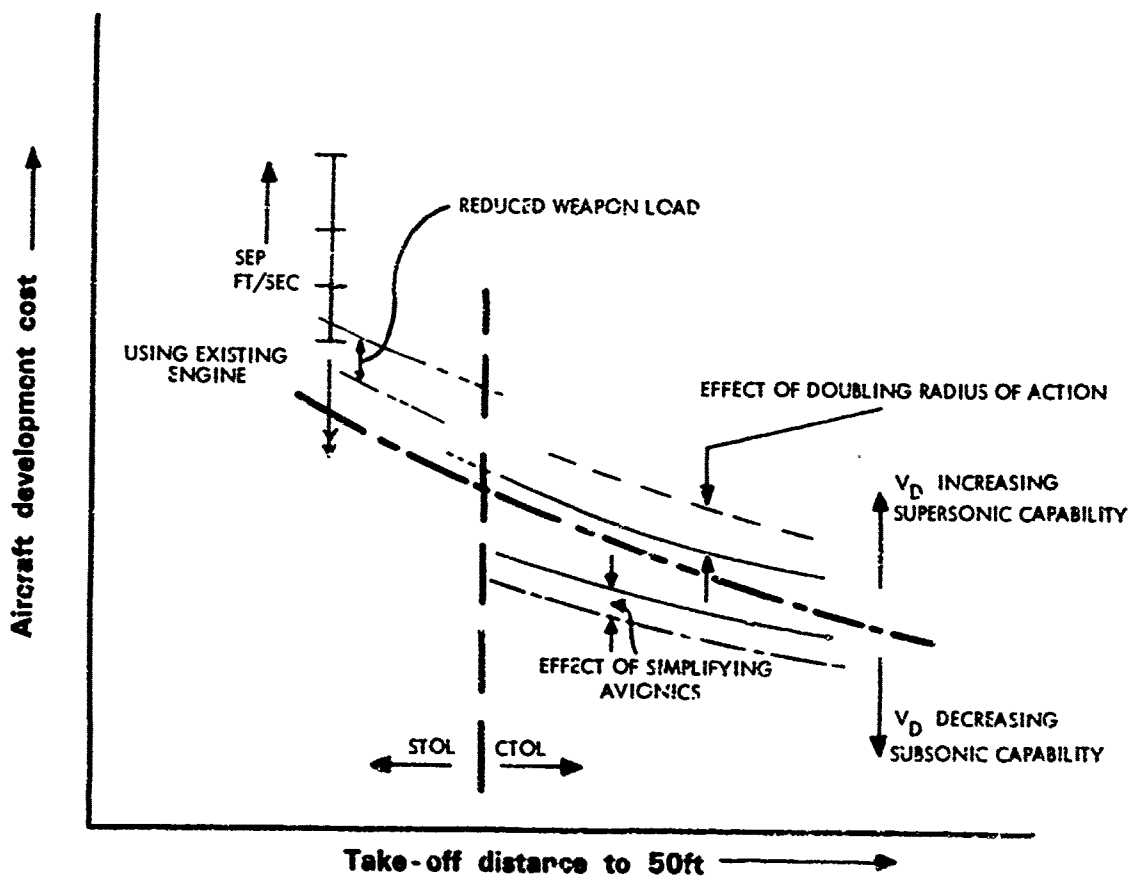


Fig.16 Aircraft development cost

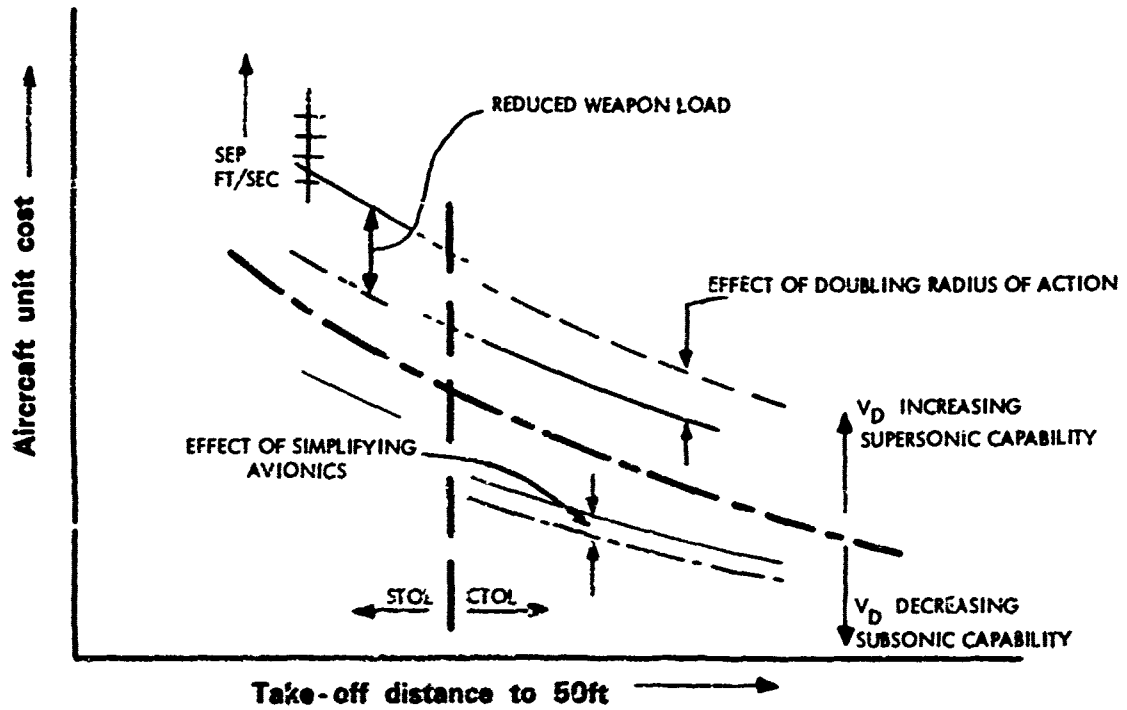


Fig.17 Average unit cost