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PRELIMINARY AIR CRAFT DESIGN

Clem C. Weissman

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AGARD LECTURE SERIES No. 65

on

Preliminary Aircraft Design

NORTH ATLANTIC TREATY ORGANIZATION

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NORTH ATLANTIC TREATY ORGANIZATION

ADVISORY GROUP FOR AEROSPACE RESEARCH AND DEVELOPMENT

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AGARD Lecture Series No.65

PRELIMINARY AIRCRAFT DESIGN

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PREFACE

This Lecture Series, sponsored by the Flight Mechanics Panel and the Consultant and Exchange Programme of AGARD, presents the procedure used by that small group of senior engineers who must respond to a proposed military requirement with the first estimate of a complete airplane design. The decision rationale and the initial estimation of size, weights, lift and drag, performance and cost is presented vis-à-vis the military payload for various aircraft types and classes and the proposed mission. Emphasis is placed on how this small preliminary design team must make the first decisions regarding technical feasibility and operational desirability. An experienced design team can predict with sufficient accuracy the overall weight, configuration, performance and cost to permit confident decision to proceed with advanced development of the project. The manner in which this is accomplished is the principal theme of this Lecture Series.

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Introduction to Preliminary Aircraft Design

AGARD Lecture Series No. 65

Clem C. Weissman

Gentlemen, as the announcement of AGARD Lecture Series No. 65 indicates my introduction will include a discussion of the military requirement and the relationship to the resulting aircraft size and performance. My basic premise is that to design an aircraft is relatively simple and can be accomplished by a few experienced aircraft designers but that to satisfy the ever changing requirements for special performance or special cost objectives or special desires and still produce a universally acceptable aircraft is a challenge in iterative human relations.

To begin with, I assume we are all generally familiar with the basic preliminary design procedure, i.e., beginning with a desired physical payload, be it people or electronics or weapons, and an indication of desired performance, a three-view and inboard profile drawing is made to estimate weight and balance and the non-interference of moving parts. Lift, drag, thrust and fuel flow estimates follow and calculations are made of take-off, climb, cruise, combat performance, etc., to see if the initially assumed physical size of the preliminary aircraft is too large, too small, or in the desired arena. Iteration of the above can then provide the "optimum" weight and size aircraft for the postulated performance and payload. For the moment I have glossed over the many decisions the designer must make in this process but you will hear of these in much detail from the lecture team.

If we accept for the moment that the basic preliminary design analysis is relatively easy and is an objective problem and solution process, I should like to discuss the subjective part of preliminary aircraft design.

The aircraft designer would naturally like to have in hand before he begins his work, a clear statement of the desired aircraft in all respects possible. The statement, either in the form of a military requirement or a specification should provide him with the goals and objectives of the military or commercial user of the aircraft. The degree of information has always been in contention. Either the requirement is too broad and questionable or it is too specific and binding. This results in committees, studies, and operations analyses to determine what the requirement should "really" be. This argument may be enlarged by threat analyses, economists, inventors, and politicians who all have their unique solutions. In my personal experience, I have seen all kinds of situations in which the preliminary design engineer was either a participant or a bystander trying to obtain sufficient information to begin objective studies. In the U.S.A. in recent years this has been resolved by the politicians and industrialists announcing the industry will produce "prototypes" and requirements will "naturally" evolve.

The degree and amount of information given the airplane designer can and does vary with a country's culture and personalities. Comparisons of life-styles and engineering developments are frequently made to argue that the other man's way of doing airplane development is a better way and should be adopted. Contractor participation either directly in government studies or thru the advertising media influences the requirements finally issued to an aircraft design team. The aircraft designer in this changing emotional subjective period must ascertain for himself certain basic boundaries for the new design. These are:

1. The operational concept. Is there a principal dominating mission? Does an envisioned tactic drive the problem? Are there geographic or atmospheric restrictions in operations? Are there physical boundaries in operations?

2. The aircraft performance desired. Desired aircraft speed, range, altitude ceiling, acceleration and payload for various flight profile missions are usually specified. Are the numbers rigid or is there a priority balance to the many performance items? Are the military payloads consistant with the performance desired?

3. Physical constraints. If land-based what is the surface bearing factor? What are the hanger door dimensions? If ship-based, are there catapult and arresting gear energy limitations? Is there a physical size and weight limitation?

Simply stated the preliminary airplane designer wants to begin his problem with information about the desired airplane performance when the airplane must carry a specified load and must perform within a set of physical restraints. To his confusion the economist will argue all projects must be designed to a fixed cost; the politician will argue the threat probability and how to build the aircraft; and the military operator will change the performance required.

The period of argument and change before a preliminary airplane designer can begin his work consumes more time and effort than the act itself of designing to a fixed requirement. Since the designer collected parametric and various basic data of past aircraft to compare and substantiate his new design as part of his experience data bank, it has become necessary to include him in discussions and operational analyses arguing the new requirement. This leads to what I call the "What If Period" before a requirement is fixed and issued. By that I mean the airplane designer is asked over and over again -- quote -- What does the airplane look like if we change the range? What if the speed is supersonic? What if the cost is less? What if our prime objective is maintainability? -- unquote.

Depending on the motives of the people engaged in this period of subjective argument, the project can be delayed by ordering more study, ordering more information on cost or operations or maintainability or by arguing the threat and time period that the aircraft is "really" needed. In this atmosphere the basic airplane dimensions and performance become more and more fixed in the designer's opinion and he has acquired considerable awareness of what people and what conditions may influence his final design. By the time the firm requirement is issued, he can literally press the computer button and the answer falls out.

Why the preliminary aircraft designer can do this I believe is because he has been studying the past to predict the future. To produce a paper design in which you have high confidence that it will perform as described when produced in actual hardware requires knowledge and experience of what worked and how well it worked in the past. The preliminary airplane designer collects and uses this information. You will hear from the lecture team how the designer uses it, I would like to indicate what some of the basic items are.

To begin with there are certain laws of nature and physics that must be observed. Man has a physical size and weight and needs volume to work in and lines of vision to observe pertinent outside actions or surroundings. My first step in a three-view and inboard profile drawing was to put down the average pilot and a then required fifteen degree down over-the-nose line of vision. With this starting point, the governing features of the military payload, such as diameter of nose radar, volume for a command center, or space for internal weapons carriage, are added and a first-try airframe with selected wing sweep, aspect ratio, fuselage length to beam ratio, tail size, etc., follows. Depending on mission and availability of propulsion systems, a suggested power plant arrangement is added and the three-view drawing is developed so that airplane volume can be checked for fuel storage. During this drawing the airplane designer is aware of weight and balance and the aerodynamic effect of the features of the preliminary design. Features are drawn consistant with his experience and available flight test or wind tunnel data from similiar configurations. Compromises are made between such items as good all around vision in a fighter aircraft and the high canopy drag that would induce. Attempts are made to minimize drag and weight and maximize lift and thrust.

After the three-view drawing and a weight and balance statement is prepared, the designer is ready to calculate the lift and drag at various altitude and flight speeds and compare these values to corresponding expected thrust and weight conditions. The estimated performance is then compared to the desired or required performance and the process iterated with those modifications to the first approximate design necessary to approach an acceptable final preliminary airplane design.

It is my contention that this objective design process is relatively simple, and can be used by a small group of experienced engineers who appreciate the effect of suggested point changes on the overall design, to quickly establish the principal characteristics of aircraft that when produced will perform as stated with acceptable tolerances.

There is one apparent dichotomy to discuss in closing. During the design cycle addition of equipment or adding more fuel for greater range results in the overall design weight increase by factors ranging from four to twelve depending on aircraft type. Yet during the life cycle of military aircraft, equipment and tasks are changed without too much notice or attention. In fact, changes are made within limits until they noticeably affect other items. So we observe the cycle of additional equipment weight increases finally resulting in propulsion system changes and in changes to provide strength and safety. Overall performance may be lost or sufficient changes are made to regain the original performance.

AVANT-PROJETS CIVILS ET MILITAIRES AUX AVIONS MARCEL DASSAULT-BREGUET AVIATION

par

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1. INTRODUCTION

Au cours des 25 dernières années la Société AMD/BA a étudié, réalisé et mis au point une quarantaine de prototypes d'avions civils et militaires. Cette grande famille comporte les types d'appareils les plus variés. On y trouve des chasseurs, (intercepteurs et appui tactique) y compris un VTOL supersonique, un bombardier Mach 2, des appareils embarqués, des avions-école, des chasseurs-bombardiers à géométrie variable, des avions de surveillance anti-sous-marine et des transports STOL ; ceci pour les avions militaires. Les avions civils sont représentés par des avions d'affaires, par l'avion de transport 3 ème niveau FALCON 30 et par le MERCURE, le plus grand et le plus lourd appareil que la Société ait jamais construit.

La plupart de ces prototypes ont donné lieu à des séries plus ou moins importantes (figure 1). A ce jour plus de 3 000 appareils ont été construits ; de plus, plusieurs séries sont en cours de démarrage. Fait particulièrement intéressant, la participation de la Société à l'ensemble de la production est inférieure à 50 % en heures de travail. Les séries sont produites en sous-traitance, en coopération nationale ou internationale, ou fabriquées sous licence.

La figure 2, représentant le domaine balayé en flèche et en allongement par l'ensemble des appareils réalisés, en illustre la grande variété. La figure 3 met en évidence une autre caractéristique importante de conception, le progrès dans la continuité. Les enseignements tirés de l'étude et de la mise au point de ces appareils constituent évidemment une source inépuisable d'informations.

Associés à cette double qualité de progrès et de continuité, d'autres facteurs sont à la base de la réussite de la plupart des programmes entrepris :

- la qualité des équipes tant dans les bureaux d'études que dans les ateliers
- le souci de l'efficacité et de la rapidité. Décisions de base prises très rapidement et faibles délais de réponse pour les ajustements nécessaires.
- la recherche permanente des possibilités d'amélioration
- la liasse prototype relativement réduite, la liasse série très détaillée en vue de permettre l'éclatement de la fabrication.

Cette politique dynamique où le temps constitue un facteur si important est grandement facilitée par la coopération fructueuse entre les Services Officiels et le constructeur. Le système pratiqué en FRANCE, qui consiste à désigner un ingénieur de marque dans le cadre du STAé et de plus un officier de marque dans le cadre de l'EMAA pour les avions militaires, n'a cessé de se perfectionner au cours des années. L'ingénieur de marque, de par sa formation, parle aussi bien le langage du constructeur que celui des militaires, assure les liaisons indispensables et accélère les communications entre les parties concernées. Dans ces conditions, le constructeur est à même de participer à l'élaboration des programmes, d'effectuer les études de faisabilité et de contribuer ainsi à l'orientation et à l'évolution des spécifications. Ainsi le programme définitif sera le fruit d'une coopération, à l'occasion de laquelle les problèmes critiques auront été abordés et leur solution examinée. Il va sans dire que dès ce stade toutes les contraintes telles que :

- les prévisions budgétaires
- les délais d'exécution du programme
- le niveau de technologie à appliquer

ainsi que les besoins du marché intérieur et extérieur sont pris en compte.

Après ces généralités nous nous proposons de décrire l'organisation de l'équipe d'avant-projet (qui par la suite constituera le noyau du groupe chargé du projet).

Sans être totalement rigides, les principes directeurs en sont les suivants :

- pour ce qui concerne les options fondamentales et les décisions finales, elles sont prises au niveau le plus élevé, Monsieur Marcel DASSAULT intervenant personnellement aux différents stades de l'étude et de la mise au point
- la responsabilité du programme est confiée à un chef d'avion, ou ingénieur de marque. Pour développer le projet celui-ci dirige une équipe et fait appel aux Divisions spécialisées avec lesquelles il est en liaison permanente. Il est à souligner que toutes les décisions importantes sont supervisées et sont suivies dans le détail par le Directeur Technique
- la tâche principale de l'équipe d'avant-projet, formée d'un nombre réduit d'ingénieurs et de techniciens, consiste principalement en un travail de synthèse et de dessin
- l'analyse est généralement effectuée par les Divisions spécialisées, plus particulièrement par la Division des Etudes Avancées qui dès ce stade entreprend des études aérodynamiques et des essais en soufflerie relativement détaillés.

Nous arrivons maintenant au problème délicat du choix de configurations initiales. A des études paramétriques systématiques, en particulier pour les programmes militaires, on préfère l'évaluation aussi complète que possible d'un certain nombre d'avant-projets de conceptions différentes, chacun étant adapté au programme, sinon optimisé.

Un premier tri permettra d'éliminer les solutions s'avérant nettement inférieures à la moyenne.

Au cours de l'itération suivante sur les solutions restantes l'étude sera poussée plus loin dans les détails, conduisant ainsi à une réévaluation plus précise et à un nouveau tri. Lors de ces itérations, des données subjectives, telles que l'expérience acquise sur telle formule particulière, la disponibilité de certains types d'outillage etc... sont prises en considération.

Après quelques itérations le nombre de solutions satisfaisantes se réduit régulièrement pour aboutir finalement soit à une solution unique, cas relativement rare, soit à 2 ou 3 solutions concurrentes, qui sans être équivalentes, constituent des compromis très difficiles à départager.

A ce moment crucial il revient à l'Etat-Major de la Société d'exercer son jugement et son expérience, pour effectuer le choix le plus adéquat parmi les quelques solutions restantes. C'est à ce stade que l'architecture générale du projet se dessinera et les options adoptées conditionneront la suite du programme.

Ces considérations montrent l'importance des études préliminaires comparatives, et en particulier la nécessité de disposer d'informations précises pour ces évaluations. Toute erreur d'appréciation pourrait déformer le jugement et conduire à un choix incorrect.

Pour être en mesure d'effectuer ces études avec la rapidité souhaitable, l'équipe d'avant-projet doit avoir a sa disposition :

- des méthodes d'évaluation de devis de masse suffisamment précises (mises à jour en permanence)
- des résultats expérimentaux définissant les caractéristiques aérodynamiques des principales formules envisagées
- des programmes de calcul d'aérodynamique théorique et des méthodes de corrélation permettant la généralisation des résultats à l'ensemble des formules étudiées (variation des formes en plan, épaisseurs relatives, des dimensions et formes du fuselage ; évaluation des interactions entre différents éléments, influence de la propulsion etc ...).
- des données relatives à la propulsion, c'est-à-dire suivant le cas, des programmes de calcul pour le ou les moteurs donnés, ou des performances généralisées si la propulsion est également à définir.

Quant au système d'armes, au cours de l'établissement de la fiche programme il aura fait l'objet d'une première série d'études :

. de recherche opérationnelle
. de compatibilité et d'optimisation
. de définition des charges militaires

Par suite de la forte interaction entre l'électronique, l'armement et la cellule, de nombreuses retouches seront nécessaires sur tous ces éléments au cours de l'élaboration de l'avant-projet.
Aussi toute une équipe spécialisée de systèmes travaillera dès ce stade sous l'autorité directe du Chef d'avion.

Complétant l'équipe d'ingénieurs, un groupe de quelques dessinateurs sera chargé de certaines études de détail, de dimensionnement, d'aménagement ou de conception structurale, de manière à amener les différentes versions de l'avant-projet au même stade d'avancement.

Pendant cette phase de l'étude une liaison permanente est maintenue avec les Divisions spécialisées pour que les solutions techniques et technologiques adoptées soient poussées aux limites, sinon au-delà, de "l'état de l'art". Cette façon de procéder doit permettre de puiser au maximum dans l'expérience acquise, sans être cependant trop attaché au passé, et de tirer le meilleur parti des possibilités du progrès technique.

2. LES AVIONS DE COMBAT AMD/BA.

2.1. OURAGAN, MYSTERES et MIRAGES.

Examinons maintenant comment ces principes ont été appliqués dans divers domaines et tout d'abord dans le domaine d'activité longtemps prépondérant, celui des avions de combat.

Le premier avion de combat à réaction, conçu par la Société, l'OURAGAN, a effectué son premier vol le 28 Février 1949. Par suite de la politique de continuité pratiquée par la Société on peut considérer qu'il était pour de longues années le seul véritable prototype. Sans vouloir nous attarder sur cet appareil, disons seulement qu'à partir de l'OURAGAN, équipé du réacteur à compresseur centrifuge Nene, et construit en 360 exemplaires, les Bureaux d'Etudes par le procédé d'extrapolation continu illustré sur la figure 3 ont dérivé successivement le MYSTERE II, le MYSTERE IV et le SUPERMYSTERE. Dans cette évolution, qui a duré 6 ans, l'épaisseur relative est passée de 13 % à 6 %, la flèche pratiquement de 0 à 45°, la poussée des réacteurs de 2 300 à 4 500 kg.
Au total 800 MYSTERES ont été construits et nombre d'entre eux sont encore en service dans plusieurs pays.

Un tournant a été pris vers les années 1952-1954 lorsqu'il est apparu que l'évolution ne pouvait pas se poursuivre dans la voie précédente. Dans leur conception de l'époque ces appareils étaient pratiquement en butée de vitesse, leur charge militaire et leur rayon d'action insuffisants, et de plus ils avaient besoin de pistes de plus en plus longues, donc vulnérables.

On se trouvait également à la croisée des chemins du point de vue propulsion. D'une part, il n'existait pas de réacteur suffisamment puissant qui permettait d'obtenir les performances de vitesse (M ≈ 2) jugées indispensables. D'autre part pour définir la politique à suivre en matière de propulsion et développer le réacteur de taille adéquate, il fallait également se prononcer sur les mérites respectifs du mono et du biréacteur, problème qui à l'époque se posait en termes encore plus complexes qu'aujourd'hui.
Les aléas de développement des réacteurs de faible puissance n'ayant pas été sous-estimés, les Services Officiels ont opté pour la poursuite simultanée de leur développement et de celui du réacteur ATAR de la SNECMA, équipé de postcombustion.
Pour atteindre les nombres de Mach et altitudes élevés jugés nécessaires à l'interception l'emploi de fusées était en outre indispensable. Le programme de l'intercepteur léger a fait l'objet d'une compétition serrée ; la Société DASSAULT a proposé deux projets, l'un biréacteur, propulseurs provisoires 2 Vipers à postcombustion + fusée, l'autre monoréacteur + fusée. Elle a obtenu la commande de la version biréacteur, le MIRAGE I, dont le premier vol a eu lieu en Juin 1955 et moins d'un an après il volait équipé de postcombustion et de fusée. Son successeur, le MIRAGE II, équipé des réacteurs définitifs se trouvait à un stade de construction avancée, lorsque pour des raisons diverses qui seront développées ci-dessous, la Société décida de changer de politique, d'en arrêter la construction et de revenir avec son propre financement à un monoréacteur, en utilisant l'aile déjà construite du MIRAGE II.

Cette opération a été menée avec une extrême rapidité : l'appareil MIRAGE III-001 équipé de l'ATAR 9B + postcombustion + fusée, et dont le fuselage comportait déjà la loi des aires a volé en Novembre 1956, 9 mois seulement après cette décision. Très rapidement il a atteint des vitesses supersoniques en palier, M = 1,5 au 5 ème vol et Mach 2 à la date du 24 Octobre 1958. Finalement cet appareil emporta la compétition et ouvrit la voie au chasseur polyvalent MIRAGE III. Plusieurs versions ont été développées par la suite et actuellement une cinquantaine de variantes volent dans 18 pays différents. Le nombre d'appareils livrés ou en commande est de l'ordre de 1 500.

Analysons maintenant les raisons du changement de politique.

- A la suite d'études comparatives, qui ont continué pendant la construction du MIRAGE II, il est apparu qu'avec un monoréacteur + fusée il était possible d'obtenir un rapport poussée/poids plus élevé, à charge alaire égale, qu'avec un biréacteur, avantage fondamental pour un intercepteur
- Des difficultés se sont présentées dans la mise au point du moteur de faible puissance, en particulier dans le développement de la réchauffe. Par contre l'ATAR, y compris sa version avec postcombustion manifestait des progrès prometteurs.
- La crainte des limitations opérationnelles de la fusée (qui s'est avérée d'ailleurs injustifiée) incita le Bureau d'Etudes à étudier la possibilité de la supprimer par la suite, compte tenu du développement prévisible des réacteurs et grâce à des améliorations aérodynamiques que l'on commençait à entrevoir.
- Les résultats d'essais en vol acquis sur le MIRAGE I ont confirmé la validité du concept de l'aile delta sans queue et renforcé la confiance dans la formule aérodynamique adoptée.

Toutes ces conditions réunies ont donc logiquement conduit la Société à conserver l'aile delta, et à adopter la formule monoréacteur + fusée avec pour objectif plus lointain de revenir au réacteur pur.

Ajoutons pour terminer cette partie de notre exposé que la fiche-programme comportait quelques clauses qui plus ou moins directement devaient avoir une influence décisive sur l'évolution ultérieure vers la polyvalence. Ainsi, on n'a pas succombé en FRANCE à la mode de l'époque, qui était l'avion-engin. Aussi dès ses premières versions de série le MIRAGE III était équipé de deux canons DEFA de 30 mm.

En outre les exigences de piste relativement sévères ont conduit à une surface de voilure importante. La faible charge alaire qui en a résulté, associée au développement du moteur a produit la grande manoeuvrabilité si brillamment démontrée en combats aériens. (Figures 4 à 11).

Cependant il existe dans le domaine de vol quelques zones, où l'avion sans queue ne peut pas rivaliser avec l'avion empenné dans le cadre de la technologie courante. Dans les cas particuliers où ces zones seraient jugées opérationnellement importantes, il faudra faire appel à des avions empennés ou à une technologie avancée.

2.2. Appui tactique léger.

Avant de continuer l'histoire des MIRAGES qui est loin d'être terminée, il faut parler du programme d'appui tactique léger, qui lancé un an après celui de l'intercepteur n'a pas obtenu le succès de son prédécesseur. Là encore il y a eu compétition entre plusieurs Sociétés, dont DASSAULT et BREGUET aujourd'hui fusionnées. Ce sont elles qui ont été désignées comme gagnantes, avec respectivement l'ETENDARD II et le BREGUET 1100, (2 réacteurs Gabizo, formules à aile en flèche hypersustentée, décollage et atterrissage sur pistes courtes en herbe). Un an après le programme d'appui tactique léger français le NATO a lancé un programme analogue dans le cadre d'une compétition internationale. BREGUET et DASSAULT se classaient parmi les gagnants avec FIAT. Cependant, malgré la construction et la mise au point réussie de plusieurs prototypes, les modifications intervenues dans la politique aéronautique française ont conduit à l'abandon des programmes NATO et français d'appui tactique en faveur d'autres options. Les excellentes performances subsoniques acquises entre-temps par le MIRAGE III ne sont pas totalement étrangères à ces décisions. Le seul rescapé de la famille fut l'ETENDARD IV, avion d'attaque embarqué ; les porte-avions pouvaient difficilement s'accommoder des avions sans queue. Une centaine d'ETENDARD IV ont été construits et l'appareil modernisé, le SUPER ETENDARD, est appelé à acquérir une seconde jeunesse pour constituer la deuxième génération française d'avions d'attaque embarqués (figure 12).

2.3. MIRAGE (suite)

Le MIRAGE III a servi de maquette volante à deux extrapolations très différentes, le MIRAGE IV et le MIRAGE III-V. Le premier biréacteur Mach 2 est l'avion de la force de dissuasion française construit en 62 exemplaires, en service depuis 1963. Le MIRAGE III-V, équipé d'un réacteur de propulsion P et W-SNECMA TF 306 et de 8 réacteurs de sustentation ROLLS ROYCE RB 162 est actuellement le seul avion VTOL à avoir atteint Mach 2. Mais l'intérêt militaire manifesté au début des années 60 pour les VTOL ayant fortement diminué le programme a été arrêté en 1967.
Parallèlement à la construction en série des MIRAGE III et IV la Société, en coopération avec les Services Officiels, a entrepris l'étude de nouvelles formules d'avions de combat, aux performances améliorées dans tout le domaine de vol. Le développement de systèmes d'armes et de navigation de plus en plus sophistiqués, les progrès de la propulsion, en particulier la diminution de consommation spécifique obtenue avec les double-flux militaires, ont permis d'envisager la réalisation d'un chasseur-bombardier tout-temps de rayon d'action sensiblement accru et capable de voler à grande vitesse et à basse altitude sans être repéré. Ce type d'avion doit avoir une forte charge alaire incompatible avec l'aile delta. Pour ne pas être tributaire de pistes trop longues, l'appareil doit être équipé d'une hypersustentation efficace. De plus, ses qualités de vol doivent lui permettre de voler à grande incidence en toute sécurité. Ces conditions ont conduit au prototype MIRAGE F2, à aile en flèche hypersustentée au bord d'attaque et au bord de fuite, aile haute et empennage bas. L'appareil, pesant 18 tonnes à pleine charge était équipé d'un réacteur TF 306 de 9 tonnes. Il devait donner suite à une version opérationnelle de chasseur tout-temps.

Après étude plus détaillée, ce programme a été abandonné, par suite de l'absence de propulseur adéquat. La politique en matière d'avion d'armes a été alors réorientée dans trois directions :

- intercepteur, successeur du MIRAGE III
- avion école et d'appui tactique
- chasseur-bombardier à géométrie variable

2.4. Une nouvelle génération d'avions de combat : le MIRAGE F1.

Du point de vue de la conception aérodynamique, cet appareil monoplace, monomoteur ATAR 9K 50 de 7 T 2 de poussée a été dérivé du MIRAGE F2 (figure 13). Par rapport au MIRAGE III (figure 14) son domaine de vol a été élargi dans les deux directions, grandes vitesses et basses vitesses ; performances de décollage et d'atterrissage nettement améliorées ainsi que la manoeuvrabilité à certaines altitudes et Mach. Grâce à une capacité de carburant importante, l'appareil dispose d'un temps d'interception accru et est capable de missions prolongées à toutes les altitudes. Outre sa mission principale qui est l'interception, la polyvalence est assurée par une capacité d'emport de charges variées (7 points d'accrochage). (Figure 15 à 22). L'armée de l'Air française en a actuellement commandé une centaine d'exemplaires et d'autres ont été ou sont sur le point d'être commandés par divers pays étrangers. Les premiers avions de série sont entrés en service en 1973.

Parallèlement au développement en cours du réacteur SNECMA M 53, réacteur à double flux pour Machs élevés, une version MIRAGE F1 équipé de ce moteur est en cours de construction. Les améliorations de performance de ce réacteur par rapport à son prédécesseur tant en ce qui concerne la poussée, le rapport poussée/poids que la consommation spécifique doivent se répercuter sur celles du F1, et plus particulièrement sur la manoeuvrabilité, l'accélération supersonique et le rayon d'action.

2.5. L'avion école et d'appui tactique JAGUAR.

Pour respecter une certaine chronologie, quittons une fois de plus les MIRAGES, et examinons l'histoire du JAGUAR, résultat de la coopération franco-britannique tant dans le domaine de la cellule que dans celui des moteurs. La fiche programme française de l'avion Ecole et d'Appui Tactique (E.C.A.T.) est sortie en 1964 et faisait l'objet d'une compétition qui a été gagnée par BREGUET en ce qui concerne la cellule. Un accord de coopération franco-britannique signé peu après prévoyait la commande de 150 appareils par chacun des deux pays, portée ultérieurement à 200. La coopération s'établit entre BREGUET et la BRITISH AIRCRAFT CORPORATION pour la cellule, entre TURBOMECA et ROLLS ROYCE pour les moteurs en 1965.
Les points les plus importants du programme définitif sont :

- utilisation de pistes courtes (800 à 1000 m) en appui tactique
- rayon d'action en mission Lo-Lo-Lo 450 n.m
- rayon d'action en mission Hi-Lo-Hi 750 n.m
- capacité supersonique
- charge militaire maximale 4500 kg

La conception aérodynamique de l'appareil dérive de l'avion d'appui tactique léger du NATO, le BREGUET 1001, avec des perfectionnements concernant principalement le vol à grande incidence. Aile haute en flèche de 40°, empennage relativement bas grâce à un dièdre négatif. Volets hypersustentateurs sur toute l'envergure, le contrôle latéral étant assuré par des spoilers et le braquage différentiel de l'empennage horizontal. L'appareil est équipé de deux réacteurs à double-flux ROLL ROYCE/TURBOMECA Adour de 3 350 kg de poussée. Cinq versions ont été prévues :

- Appui tactique (A) monoplace, français
- Ecole (E) biplace , français
- Attaque (S) monoplace, britannique
- Entraînement (B) biplace , britannique
- Marine (M) monoplace, français

Cette dernière a été abandonnée en faveur du SUPER ETENDARD.

Après une mise au point rendue plus difficile du fait que la cellule, le moteur et une partie importante des équipements étaient des prototypes le JAGUAR est entré en service en 1973 et à ce jour près de soixante quinze appareils ont été livrés aux deux armées de l'Air. Des négociations sont en cours avec plusieurs pays en vue de son exportation. (Figures 23 à 27).

2.6. MIRAGE G, G8 et G8-A.

Le troisième volet de la nouvelle orientation politique aéronautique a été l'avion à géométrie variable. L'extension de la notion de polyvalence à des missions de plus en plus variées, et souvent contradictoires a conduit les USA au F 111. Malgré un environnement opérationnel différent, la souplesse offerte par la géométrie variable a incité les Etats-Majors européens à définir des programmes pour lesquels la géométrie variable devait constituer la meilleure solution. Une coopération franco-anglaise s'est ébauchée sur ce programme vers les années 1965-67 mais elle n'a pas abouti... Plutôt que de se limiter à des études partielles la Société DASSAULT a décidé le lancement de l'étude et de la réalisation d'un prototype expérimental qui a effectué son premier vol en novembre 1967.

Les premiers vols de cet appareil équipé d'un réacteur TF 306 de 9 T de poussée, d'une masse de 15 tonnes n'ont pas révélé de difficultés fondamentales, en particulier celles qui auraient pu résulter de la flèche variable. Ce résultat remarquable est le fruit de plusieurs facteurs :

- application du principe de continuité partout où c'était possible de façon à éviter les innovations inutiles (position aile-empennage, forme et aménagement du fuselage, système de propulsion (figure 28).
- commandes de vol classiques, excepté le spoiler tout électrique (figure 29).
- conception et mise au point mécanique du pivot et de la commande de rotation particulièrement soignée (figure 30).
- application à grande échelle des méthodes numériques d'aérodynamique théorique au dessin de l'avion.
- expérimentation en soufflerie très détaillée (figure 31).

L'appareil était supersonique dès le 5 ème vol ; la variation complète de flèche a été réalisée au bout du 7 ème vol et Mach 2 au 11 ème (figures 32 et 33).

Les performances basses vitesses ont démontré la réussite du système hypersustentateur. Malgré un rapport poussée/poids relativement faible et une forte charge alaire l'appareil décollait en 600 m et atterrissait en 500 m. La manoeuvrabilité s'est avérée excellente à toutes les vitesses, et des virages remarquablement serrés ont pu être démontrés en décélération combinée avec la diminution de flèche.

Les essais en vol du MIRAGE G ayant confirmé les promesses de la géométrie variable, des versions opérationnelles ont été mises à l'étude. L'une des conclusions auxquelles on a été conduit à cette occasion est que la formule n'est pas valable au-dessous d'une certaine taille. Il s'ensuivait qu'avec les réacteurs militaires disponibles l'appareil devait être obligatoirement biréacteur, ce qui était par ailleurs conforme aux désirs de l'Etat-Major.En conclusion, deux prototypes de biréacteurs (ATAR 9K 50) ont été commandés en 1968 et le premier de ces appareils baptisés G8, effectua son premier vol en Mai 1971, le deuxième en 1972. Là encore la mise au point a été très rapide et l'opportunité d'avoir deux prototypes a été utilisée pour l'étude de nouvelles formes d'entrées d'air et de tuyères. Accessoirement on a découvert que les avions à géométrie variable constituent de remarquables maquettes volantes, à très grande échelle, pour des avions à géométrie fixe. Les résultats acquis au cours des vols à différentes flèches de voilure ont pu être exploités et pourront donner lieu à une nouvelle génération d'appareils.

Nous arrêtons là l'histoire de la famille des MIRAGES, pour parler maintenant d'autres réalisations.

2.7. Le BREGUET ATLANTIC.

Cet avion de lutte anti-sous-marine est issu d'un concours NATO lancé en 1957 et gagné par BREGUET sur une trentaine de concurrents. Le programme établi par le Comité Directeur International, comportait des missions de surveillance à longue distance et de longue durée. Pour la phase de surveillance une maniabilité exceptionnelle était requise aux basses vitesses, imposant une charge alaire relativement réduite. Une grande variété de charges largables, transportées dans une soute de grandes dimensions, devait servir à la recherche et à l'attaque des sous-marins à très basse altitude. Pour ce type de missions le propulseur le mieux adapté à l'époque (et même actuellement) était le turbopropulseur et parmi le petit nombre de types existant le Rolls-Royce Tyne. Cette adaptation était valable aussi bien en vol de croisière à 300 kts à 30 000 ft qu'à la recherche basse vitesse à faible altitude. La formule aérodynamique, très classique, l'emploi de servocommandes et des essais de maquette motorisée très complets ont permis d'éviter pratiquement toute mise au point aérodynamique en vol. Par contre la conception structurale basée sur l'emploi généralisé du nid d'abeille dans les structures primaires, relevait d'une technique nouvelle et a donné lieu à quelques difficultés aujourd'hui résolues (figures 34 à 36).

L'appareil a été réalisé en coopération, dès le stade de l'étude, entre BREGUET, SUD-AVIATION, FOKKER et DORNIER. C'était le premier appareil important réalisé en coopération internationale. L'appareil est entré en service en 1965 dans les Marines française et allemande , en 1971 dans la Marine néerlandaise et en 1972 dans la Marine italienne. Différentes versions modernisées sont envisagées, qui amélioreront aussi bien les performances que les capacités opérationnelles de l'appareil.

2.8. L'avion-école franco-allemand ALPHA JET.

La dernière production AMD/BA, étudiée et réalisée en coopération avec la Société DORNIER, l'avion école de début, prévu également pour des missions appui-feu, a bénéficié des méthodes d'études avancées dont il sera question par la suite. Equipé de deux réacteurs à double-flux SNECMA-TURBOMECA Larzac 04 de 1 350 kg de poussée, pesant suivant la configuration de 4 500 à 7 000 kg, cet appareil est appelé à être construit en un grand nombre d'exemplaires en coopération franco-allemande et éventuellement en association avec d'autres pays.

Le prototype 01 a effectué son premier vol le 26 Octobre 1973 à ISTRES, le 02 le 9 Janvier 1974 à OBERPFAFFENHOFEN et à la date du 15 Février les deux appareils totalisaient plus de 60 heures de vol et avaient couvert leur domaine de vol complet. Après cette première phase satisfaisante aussi bien en ce qui concerne la cellule que la propulsion, on abordera maintenant les essais de vrille et les essais avec charges extérieures (figures 37 à 39).

3. LES PRODUCTIONS CIVILES DES AMD/BA

3.1. FALCON 20 et FALCON 30/40.

Au début de la précédente décennie, au moment où l'avenir de l'avion de combat piloté a été remis en question, nombre de constructeurs se sont tournés vers le marché civil. Pour pénétrer dans ce marché difficile, fidèle à sa philosophie de continuité, la Société a recherché le domaine où son expérience des avions militaires pouvait être la plus profitable et les risques techniques et financiers suffisamment réduits. C'est dans ces conditions que fut entreprise l'étude de l'avion d'affaires MYSTERE 20, qui par la suite devint le FALCON 20 ou FAN JET FALCON (1962).

L'un des objectifs de l'étude était l'utilisation du maximum d'éléments communs avec des avions existants (déjà la commonalité) et la mise en oeuvre d'une technologie aussi voisine que possible. C'est ainsi que le projet s'est naturellement orienté vers les vitesses de croisière élevées par filiation avec les chasseurs de l'époque. Dans la même lignée, l'expérience acquise dans le domaine des servocommandes, et les avantages qu'elles procurent en qualités de vol et en rapidité de mise au point sont à l'origine de la décision de les adopter de préférence à la compensation aérodynamique. Les économies réalisées par l'utilisation d'éléments communs ont pu être reportées sur les postes particuliers aux avions civils, comme l'aménagement du fuselage, les équipements, l'installation des moteurs, etc. Après un premier vol effectué en 1963, le prototype a été rapidement arrêté pour remplacer les réacteurs P et W par des réacteurs double-flux à ventilateur arrière GE CF 700 pour tenir compte de l'intérêt manifesté par la Pan Am pour une telle version. Les premiers résultats d essais en vol ont été suffisamment encourageants pour donner lieu à une commande de la Pan American (160 appareils). Les certifications française et américaine ont été acquises en 1965 et depuis 300 appareils ont été livrés et le total actuel des commandes et des options est de l'ordre de 400. En 1970, la certification d'origine a été complétée par une certification acoustique suivant la FAR Part 36, le FALCON ayant été ainsi le premier au monde à acquérir ce titre.

Le succès commercial du FALCON 20 a encouragé la Société dans la voie civile et presque simultanément elle a lancé trois programmes, intéressant trois marchés différents :

- le FALCON 10, avion d'affaires ultra-rapide, 2 ème génération
- le MERCURE, avion de transport court-moyen courrier de grande capacité
- le FALCON 30/40, avion de transport 3 ème niveau

Les quatre appareils sont visibles sur la photo prise au BOURGET au moment du Salon 1973 (figure 40).

Dans le cadre d'une conférence AGARD il n'est pas indiqué de s'étendre longuement sur les caractéristiques de ces appareils. Il n'aurait pas été logique, cependant, de les passer sous silence pour plusieurs raisons :

- ils bénéficient, ainsi qu'il a été rappelé précédemment de l'expérience et de la technologie des avions militaires
- aux AMD/BA les mêmes Divisions Spécialisées étudient tous les avions, qu'ils soient civils ou militaires (pour le compte des équipes attachées à chaque avion et en liaison avec elles)
- dans certains programmes la "commonalité" peut aller encore plus loin, au bénéfice des deux types d'appareils.

Nous nous limitons donc à quelques indications d'intérêt général.

Le dernier des trois appareils du point de vue chronologique est le FALCON 30/40, dérivé direct du FALCON 20. On a conservé l'aile extrême en augmentant le plan central et le diamètre du fuselage et en ajustant les empennages en conséquence. La propulsion, constituée par des réacteurs Lycoming ALF 502, double-flux à forte dilution, est particulièrement intéressante, aussi bien du point de vue consommation que bruit. Avec des vitesses de croisière relativement élevées, le coût d'exploitation de l'appareil se compare favorablement à ses concurrents éventuels.

3.2. FALCON 10 et MERCURE et les nouvelles méthodes de dessin aérodynamique.

Simultanément à l'étude du MIRAGE G, ces deux appareils ont bénéficié des programmes ordinateur développés pour le dessin des ailes modernes, pour l'interaction aile-fuselage, pour l'hypersustentation, pour les entrées d'air, etc...
Tout un département s'est consacré à ce problème pendant de nombreuses années. Pour illustrer la puissance de ces méthodes et pour montrer le progrès qu'elles ont permis de réaliser, nous donnons ci-dessous quelques exemples. La figure 41 montre l'évolution des Cx du FALCON 20 d'une part, du MERCURE d'autre part. L'accroissement de nombre de Mach limite de $\Delta M = 0{,}03$ est d'autant plus remarquable, que la flèche du MERCURE n'est que de 25° contre 30° pour le FALCON 20, et que l'épaisseur à l'emplanture est de 12,5 % contre 10 %. Ces différences permettent par ailleurs d'alléger la masse de l'aile et d'augmenter la capacité de carburant d'une façon sensible et d'améliorer en même temps l'efficacité de l'hypersustentation. Cette dernière a été dessinée au moyen de programmes ordinateur adaptés aux corps multiples, compte-tenu de la viscosité et de certaines non-linéarités. Ainsi sur le MERCURE, il a été possible d'obtenir une très bonne finesse au décollage avec des Cz_{max} supérieurs à 2, ce qui est particulièrement important pour un biréacteur, et à l'atterrissage un coefficient de portance maximale voisin de 3. Un autre programme ordinateur permet de calculer la répartition de pression sur avion complet, compte tenu de toutes les interactions. Les figures 42 à 50 donnent des exemples de tels calculs. Les résultats ainsi obtenus ont de nombreuses applications :

- l'examen de la répartition de pression permet de détecter des zones de décollements éventuelles, et en effectuant des modifications locales ces zones peuvent être réduites sinon éliminées.
- dès le stade avant-projet des calculs de charges aérodynamiques relativement précis peuvent être faits et permettent d'améliorer l'estimation de la masse de structure
- les programmes permettent d'inclure dans le calcul des déformations élastiques ; la rigidité de la structure peut être évaluée et contrôlée
- toutes les dérivées aérodynamiques, statiques et dynamiques nécessaires à l'étude des qualités de vol ou aux études sur simulateur peuvent être déterminées et ces études entreprises dès ce stade.

Pour illustrer l'efficacité de ces méthodes par un exemple concret, on peut comparer le Mach maximal de croisière du MERCURE avec ceux de deux avions de transport de même catégorie, équipés des mêmes moteurs (JT 8D 15). Alors que le MERCURE, pour emporter selon le cas 15 à 25 % de plus de passagers a une taille, exprimée soit en surface de voilure, soit en surface mouillée, plus grande dans les mêmes proportions, son Mach maximum de croisière reste néanmoins supérieur de 0,01 à 0,03 à ceux de ces avions qui ont par ailleurs la même flèche de voilure.

L'utilisation à grande échelle de l'ordinateur s'étend maintenant également aux autres disciplines. L'intégration des programmes correspondants avec ceux de l'aérodynamique tend à devenir l'outil de base des avant-projets. Ce rôle accru de l'informatique sera brièvement exposé au chapitre suivant.

4. AVANT-PROJETS D'AVIONS MILITAIRES, PRESENTS ET FUTURS.

4.1. Fiche-programme.

Ayant rappelé les méthodes utilisées par le passé et leur évolution récente, nous examinerons maintenant les orientations qu'elles sont susceptibles de prendre parallèlement au développement de l'informatique.

On se limitera aux avions de combat, mais après adaptation convenable l'exposé sera applicable aussi bien aux avions militaires que civils.

Au stade initial envisagé, l'avionneur disposera d'une fiche-programme qui sans être totalement figée, définit cependant un certain nombre de données de base, obtenues à la suite d'études préliminaires. Elles concernent :

- les différentes charges militaires à transporter
- les profils des missions (principales et secondaires)
- l'environnement dans lequel l'avion sera appelé à évoluer (en territoire ami et ennemi)
- le système d'armes souhaité (portée, précision, degré d'automatisation, etc ...)

A ces conditions s'ajouteront des contraintes de coûts et de délais et souvent d'autres relatives aux performances, à la manoeuvrabilité, la structure, les servitudes d'emploi, l'aptitude à la maintenance, etc. Dans certains cas la propulsion sera imposée, sinon par le programme, par la disponibilité limitée de réacteurs, réduisant sensiblement les options envisageables.

Dans le cas fréquent où il ne sera pas possible de trouver une réponse satisfaisante à l'ensemble des conditions imposées, le programme devra être revu et modifié dans le sens d'un compromis acceptable.

Quelle que soit la manière dont le problème est posé, la première tâche consistera à déterminer en première approximation les principales dimensions, les masses et les performances des différentes formules envisagées.

Ce travail effectué, des itérations successives amélioreront la précision des résultats par l'étude de plus en plus détaillée des problèmes.

4.2. Etude préliminaire des différentes configurations (approximation "O").

Après avoir choisi les configurations que l'on se propose d'étudier dans le cadre du programme, un premier dégrossissage est effectué d'après le diagramme de la figure 51.

Les différents éléments du diagramme seront déterminés par analogie avec des avions existants ou d'autres moyens simples. A ce stade les cases structure et qualités de vol ne jouent pas encore de rôle important. L'analyse des masses sera faite par des considérations statistiques simples. Par exemple pour un avion de combat (supériorité aérienne ou appui) on peut adopter les pourcentages suivants pour le devis de masse (% de la masse de décollage) :

Structure	35 à 38 %
Propulsion	15 à 18 % (dépend de la technologie du moteur et du rapport T/W à obtenir)
Equipement	12 à 14 %
Masse à vide équipée	62 à 70 %

D'où une charge utile de 38 à 30 %. Pour la mission de base le pourcentage de carburant est pratiquement imposé. La charge militaire connue de cette mission s'obtiendra en pourcentage par différence entre le pourcentage de charge utile et le pourcentage de carburant. D'où la masse de décollage et le devis de masse. (Configuration correspondant au domaine complet de vol).

Cet exemple n'a pas d'autre prétention que de montrer que pour ce type d'avion la charge militaire ne peut représenter qu'une très faible fraction de la masse de décollage et que toute erreur sur le devis de masse se répercute très rapidement sur la charge militaire ou le rayon d'action. On voit également que toute augmentation de la charge militaire, conduit à performance égale, à une augmentation de masse de décollage de l'ordre de 3 à 5 fois l'augmentation de la charge.

La poussée du réacteur sera ensuite déterminée par des considérations de technologie, en particulier par son rapport poussée/poids et la masse disponible pour la propulsion.

On pourra ensuite dessiner le fuselage avec le réacteur mis en place, ainsi que son aménagement complet. Il restera à déterminer les caractéristiques géométriques de l'aile. On considérera différentes formes en plan et pour la surface on adoptera une valeur basée sur la charge alaire, ou plusieurs valeurs. Dans cette phase les empennages seront définis par des considérations statistiques.

L'avion étant ainsi géométriquement déterminé on calculera ses caractéristiques de traînée et de portance par des méthodes simplifiées. La première ligne du bloc-diagramme étant ainsi entièrement évaluée, on pourra passer au calcul des performances et des manoeuvrabilités (pas de coûts à ce stade) et au calcul des sensibilités aux paramètres principaux.

En général les performances et manoeuvrabilités requises par le programme ne seront pas obtenues. Au moyen des sensibilités du bloc-diagramme il sera possible de corriger les paramètres principaux de manière à se rapprocher raisonnablement des caractéristiques exigées.

Il y a lieu d'insister sur le fait que l'approximation qui vient d'être définie n'est nécessaire que lorsque le type d'avion à étudier s'éloigne sensiblement des formules sur lesquelles on a une bonne expérience. Dans le cas contraire cette approximation pourra déjà être relativement précise et le nombre d'itérations à effectuer sera alors notablement réduit.

4.3. Itérations et finalisation de l'avant-projet.

Partant de l'approximation définie au paragraphe précédent, nous pouvons maintenant réévaluer les éléments du bloc-diagramme de la figure 51 en prenant en considération cette fois les blocs structure et qualités de vol.

La géométrie des différentes versions considérées étant définie, certains paramètres pouvant rester variables (l'épaisseur relative et la surface, etc...) on évaluera les caractéristiques de portance et de traînée par des méthodes plus ou moins sophistiquées. Celles-ci sont basées sur des résultats théoriques et expérimentaux soit disponibles soit établis pour ces avant-projets particuliers. (Voir par exemple l'Annexe 1).

Les performances de propulsion seront en général disponibles pour les divers types de moteurs considérés.

Parallèlement aux calculs, des études d'aménagement de fuselage et de voilure sont faites pour préciser les formes, les dimensions et les volumes pour le carburant, l'armement, le train, etc.

Le devis de masse sera établi :

- au moyen de formules éprouvées pour ce qui concerne la structure (références 2,4, 5)
- par les données fournies par le motoriste et l'étude détaillée de l'installation propulsive
- par la définition aussi précise que possible de la liste d'équipements et du système d'armes.

Des boucles internes d'itérations permettront de tenir compte du domaine de vol et des masses caractéristiques. Des recoupements seront effectués au moyen de dessins de détail pour contrôler les masses des principaux éléments de structure.

Au cours de ces évaluations on examinera les gains qui pourraient résulter des progrès techniques et technologiques, de l'emploi de nouveaux matériaux, etc ...

Simultanément aux études de configurations et de leurs performances, on abordera les problèmes de qualités de vol.

On examinera :

- différentes configurations d'empennages et leur dimensionnement en vue d'obtenir des qualités de vol excellentes en mode normal et acceptables dans les modes dégradés
- l'influence des charges extérieures dans tout le domaine de vol et de centrage opérationnel
- différentes solutions de réalisation des commandes, d'efforts artificiels et de stabilisation artificielle
- le comportement de l'appareil :
  . dans les manoeuvres rapides
  . dans la turbulence
  . au moment du largage des charges
  . aux incidences élevées
  . éventuellement en vrille

Ainsi qu'il a été dit précédemment une partie de l'étude pourra se faire par voie théorique. Pour le reste, il faudra disposer de maquettes, sinon entièrement conformes, à ce stade préliminaire, du moins représentatives.

Les résultats obtenus pourront guider la sélection entre plusieurs configurations par ailleurs sensiblement équivalentes. Ainsi la configuration de l'avion-école franco-allemand ALPHA JET a été adoptée principalement à cause de son comportement favorable en vrille, mis en évidence à l'occasion d'essais comparatifs effectués au stade de l'avant-projet.

Ajoutons que ces considérations sont valables pour des commandes de vol classiques, ou à la rigueur pour la période transitoire actuelle. Lorsqu'avec le développement des commandes de vol électriques (fly-by-wire) les CCV (Control Configured Vehicles) deviendront opérationnels, elles devront être adaptées à la nouvelle situation.

Revenons maintenant à notre objectif principal qui est la finalisation de l'avant-projet. Avec les données de base ainsi améliorées nous sommes en mesure d'effectuer maintenant des calculs de performances et de manoeuvrabilité réalistes sur l'ensemble des configurations envisagées.

Parmi le grand nombre de solutions obtenues, résultant des différentes combinaisons de paramètres variables, il y aura lieu de retenir celles qui se rapprochent au mieux des conditions imposées par le programme.

Au moyen des coefficients de sensibilité déduits des résultats précédents il sera alors possible de réajuster certains paramètres pour que les objectifs fixés soient satisfaits au mieux, dans les limites des possibilités technologiques. Des itérations supplémentaires seront effectuées au cas où la précision obtenue serait jugée insuffisante.

La sélection finale entre les différentes solutions ainsi élaborées sera basée, outre les considérations de qualité de vol précédemment mentionnées, sur des considérations complexes de côut/efficacité, de potentiel de développement, etc.

Nous avons ainsi mis en évidence le rôle toujours croissant des ordinateurs dans l'établissement des avant-projets. Sans leur emploi intensif il n'aurait pas été possible de mener de front les études indispensables, ni quantitativement ni qualitativement. Grâce aux développements que l'on peut entrevoir dès maintenant, des améliorations importantes pourront être obtenues sur le plan de la précision et d'une meilleure intégration des différentes disciplines.

Les méthodes analytiques qui viennent d'être rappelées brièvement sont compatibles avec une politique dynamique de développement de prototypes. De leur association judicieuse doivent sortir des productions sensiblement améliorées.

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ANNEXE 1

DETERMINATION DE LA TRAINEE ET DE LA PORTANCE AU STADE AVANT-PROJET

1. Nous disposons maintenant de plusieurs méthodes sur ordinateur permettant d'aboutir à partir d'un dessin d'avion à un bilan de trainée. Ces méthodes peuvent nécessiter un temps de calcul qui peut aller de quelques secondes d'ordinateur à plusieurs heures.

Même au stade avant-projets, toutes ces méthodes peuvent être utilisées, le choix ne dépend en gros que du temps que l'on veut consacrer aux "entrées" dans l'ordinateur.

2. Néanmoins quelles que soient les précautions prises et le degré de sophistication des outils utilisés, il subsiste à ce stade une large plage d'incertitude, ne serait-ce que sur la définition de l'avion lui-même.

A titre d'exemple, le fuselage avant d'un avion militaire supersonique, c'est-à-dire en avant des entrées d'air dans le cas de nos avions, peut mettre plusieurs mois à se définir. En effet, plusieurs itérations peuvent être nécessaires pour aboutir à des formes donnant une répartition de vitesse satisfaisante dans le plan des entrées d'air et adaptée au domaine de Mach et d'incidence. On pourrait croire qu'une règle de trois par rapport à un fuselage précédent puisse suffire. Ceci serait vraiment trop grossier car il suffit que le nouvel avion doive emmener un radar d'un diamètre différent ou disposer de banquettes latérales plus larges pour que tout soit remis en cause et notre expérience a montré que le résultat pouvait amener des écarts de traînée non négligeables.

Le dessin de l'arrière-corps de l'avion et son mariage avec la ou les tuyères des réacteurs sont rarement bien connus au départ du projet. Il peut être en particulier fortement influencé par la position des empennages horizontaux et les dimensions des cadres d'attaches de ces empennages. Or il est rare qu'au stade avant-projet, on puisse considérer les empennages comme étant dans leur position définitive et il est fréquent que des calculs d'aérodynamique théorique tridimensionnelle avec empennage ou des essais en soufflerie viennent remettre en cause les choix initiaux.

Les différentes prises d'air utilisées pour les ventilations des propulseurs et des équipements peuvent être influencées par le choix de leur position dans l'avion et ce choix est souvent modifié lorsque le centrage ou la structure de l'avion sont mieux connus.

On peut dire également que le poste baptisé "traînée additive" des entrées d'air dépend du compromis définitif retenu pour l'adaptation de l'entrée d'air dans tout son domaine de fonctionnement.

Des compromis avec des considérations de réduction de coût de fabrication interviendront lors de la définition plus précise de l'avion. Une forme locale peut être modifiée pour permettre d'éviter l'usinage d'un cadre sur une machine-outil à commande numérique 5 axes mais cette modification est en général faite au détriment de la continuité de la courbure chère aux aérodynamiciens.

3. Enfin, une foule de petits détails peuvent intervenir dans le bilan de traînée, forme des carénages, joints de becs de bords d'attaque, joints d'étanchéité de portes, tolérances de jonctions entre panneaux, etc...

On peut dire qu'il suffit de partir de l'expérience d'un avion précédent pour estimer tous ces détails de formes ou de réalisation. Notre Société ayant réalisé de nombreux avions fort différents comme il a été montré dans le texte précédent, notre tâche est largement facilitée.

Néanmoins, cette expérience a montré qu'il peut être sage pour le responsable de la détermination des traînées de faire intervenir des coefficients assez subjectifs qui peuvent tenir compte des habitudes de l'équipe qui sera responsable de la réalisation de l'avion. Il sera utile également pour ce responsable, surtout lorsqu'il s'agira de fournir les performances garanties associées à un contrat, de connaître les orientations probables des compromis qualité/coûts et qualité/délais.

4. La liste des termes pouvant enrober d'une large tache de flou la détermination précise d'un bilan de traînée est suffisamment longue pour ne justifier au stade avant-projet que l'utilisation de méthodes grossières. Procéder ainsi serait néanmoins une erreur pour plusieurs raisons :

- tout d'abord il est important de connaître très tôt les zones critiques d'un avion, par exemple les zones susceptibles de décollements dans certaines conditions de Mach, d'incidence ou de dérapage.
- ensuite, il est important d'établir des dérivées partielles aussi précises que possible pour aboutir au meilleur compromis pour la définition aérodynamique de l'avion.

(Les paramètres les plus importants étant la surface de voilure, les dimensions du fuselage, des entrées d'air, etc ...).

5. En ce qui concerne la détermination de la portance, nous sommes à peu près dans la même situation.

D'une part, nous disposons d'un arsenal d'outils théoriques sur ordinateur. Ces outils faisant internvenir des écoulements visqueux, il est possible de faire des calculs au nombre de Reynolds de vol et à celui de la soufflerie pour les recoupements avec les essais expérimentaux effectués le plus tôt possible sur des modifications locales de maquettes existantes d'autres avions. L'expérience nous a montré que ces outils pouvaient nous donner dans le domaine très délicat de l'hypersustentation non seulement les profils de becs et de volets à adapter mais aussi une bonne estimation de leur performance.

D'autre part, il y a dans ce cas également de nombreux détails à considérer. Une jonction entre deux becs ou deux volets, le sillage d'un rail peuvent compromettre très facilement une bonne stabilité longitudinale ou un supplément de portance escompté de la part d'un "bon" volet.

Les performances d'une hypersustentation seront donc également entourées d'un certain flou, mais les conclusions générales proposées dans le cas des bilans de traînée restent valables dans le cas de l'estimation des coefficients de portance des avant-projets.

ANNEXE 2

ROLE DES AMD/BA DANS LA CONCEPTION ET LE DEVELOPPEMENT DES SYSTEMES D'ARMES

1. INTRODUCTION

La valeur d'un nouveau type d'avion ne peut plus se séparer de celle des matériels d'avionique et d'armement qui l'équipent. L'efficacité d'un Système d'Armes complet dépend beaucoup des études et essais qui ont été faits pour réaliser la meilleure intégration possible des armements et des matériels d'électronique entre eux et, avec la cellule.

AMD/BA a compris l'importance de ce problème, dès l'apparition des premiers systèmes électroniques intégrés sur les avions militaires et il existe donc au sein de l'entreprise, un Département "Système d'Armes" qui a en charge l'étude, la définition, les essais d'intégration des systèmes d'avionique et d'armement des avions.

2. PHASE D'AVANT PROJET

Chaque fois qu'un nouvel avant projet d'avion militaire est réalisé, ce Département étudie en parallèle les caractéristiques de l'armement et des matériels électroniques qui doivent l'équiper en fonction des objectifs qui ont été fixés par l'utilisateur.

L'attention se porte d'abord sur les éléments de définition du système qui ont une répercussion directe sur les performances de la cellule :

. masse des équipements

. définition des chargements externes et en particulier, les missiles

. diamètre d'antenne du radar

. protubérances extérieures : antennes - détecteurs infra-rouge - LASER - etc...

. bilans electriques et conditionnement d'air

Les princiapux éléments qui peuvent avoir un effet sur l'efficacité globale du système d'armes sont modélisés et paramétrés de façon à faire une comparaison complète des solutions envisagées.

Lorsqu'il s'agit, par exemple, d'un avant projet d'intercepteur la comparaison des solutions est effectuée en tenant compte :

- de la taille et de la trainée des missiles air-air et des diverses caractéristiques d'installation sous avion
- la taille de l'antenne radar et les caractéristiques du radar
- les moyens de détection, de navigation et de guidage de l'intercepteur
- les moyens de détection et de guidage au sol

La modèlisation complète du système d'armes permet d'orienter le choix des solutions et la définition des performances demandées aux matériels électroniques.

Ces études sont effectuées avec la participation active des constructeurs du radar et du missile.

A ce stade, le Département Système d'Armes réalise un avant projet du système électronique.

En effet, ce Département, qui a une activité continue dans le développement et la mise au point de nouveaux systèmes, peut tirer profit de l'expérience acquise et de ses connaissances dans le développement des technologies électroniques, pour effectuer la synthèse des projets effectués par les différents constructeurs de matériels et proposer des systèmes intégrés complets permettant d'atteindre les objectfis fixés en réalisant un compromis satisfaisant tenant compte des paramètres : masse, prix, performances, fiabilité, maintenabilité, risques de développement.

Ces propositions sont soumises aux Services Officiels qui font les choix.

3. PHASE DE DEFINITION DETAILLEE

Le Département Système d'Armes a normalement pour responsabilité dans cette phase :

- définition des règles générales d'installation et d'intégration des matériels
- études globales de performances
- études des commandes et visualisations - aménagement des postes d'équipage
- modelisation des systèmes - simulations
- études générales de maintenance et de fiabilité
- rédaction et proposition de spécifications techniques pour les matériels composant le système
- étude des problèmes de sécurité et probabilité de réussite de mission.

Ces études sont faites avec la participation étroite de tous les coopérants, l'avionneur ayant pour responsabilité principale de coordonner les différentes études et en réaliser la synthèse.

4. ESSAIS D'INTEGRATION

Les matériels ayant par ailleurs subi leurs essais individuels en laboratoire ou sur avion de servitude l'avionneur a généralement la charge de réaliser leur intégration :

- d'une part sur des bancs globaux au sol et dans une maquette radioélectrique échelle 1/1 de l'avion sur laquelle sont plus particulièrement étudiés les parasitages et interférences
- d'autre part sur avion de servitude et sur les prototypes de l'avion d'armes.

Ces essais, effectués avec la participation de tous les coopérants, permettent d'effectuer la mise au point et de vérifier les performances globales du système.

Les modifications nécessaires sont définies pour application aux matériels de série.

5. INTEGRATION DES MATERIELS DE SERIE

L'avionneur réalise sur les lieux de la chaine de montage final des avions des essais d'intégration des matériels à avionner, sur un banc global analogue à celui qui a permis de réaliser la mise au point prototype. L'ensemble du système est ensuite réceptionné sur l'avion d'armes par les pilotes des Services Officiels.

L'avionneur et les constructeurs de matériels fournissent à l'Armée de l'Air une assistance technique pendant les premiers mois de mise en oeuvre des nouveaux avions. L'exploitation des anomalies constatées en utilisation permettent de définir rapidement des modifications destinées à améliorer le fonctionnement, l'utilisation, la maintenance et la fiabilité des systèmes.

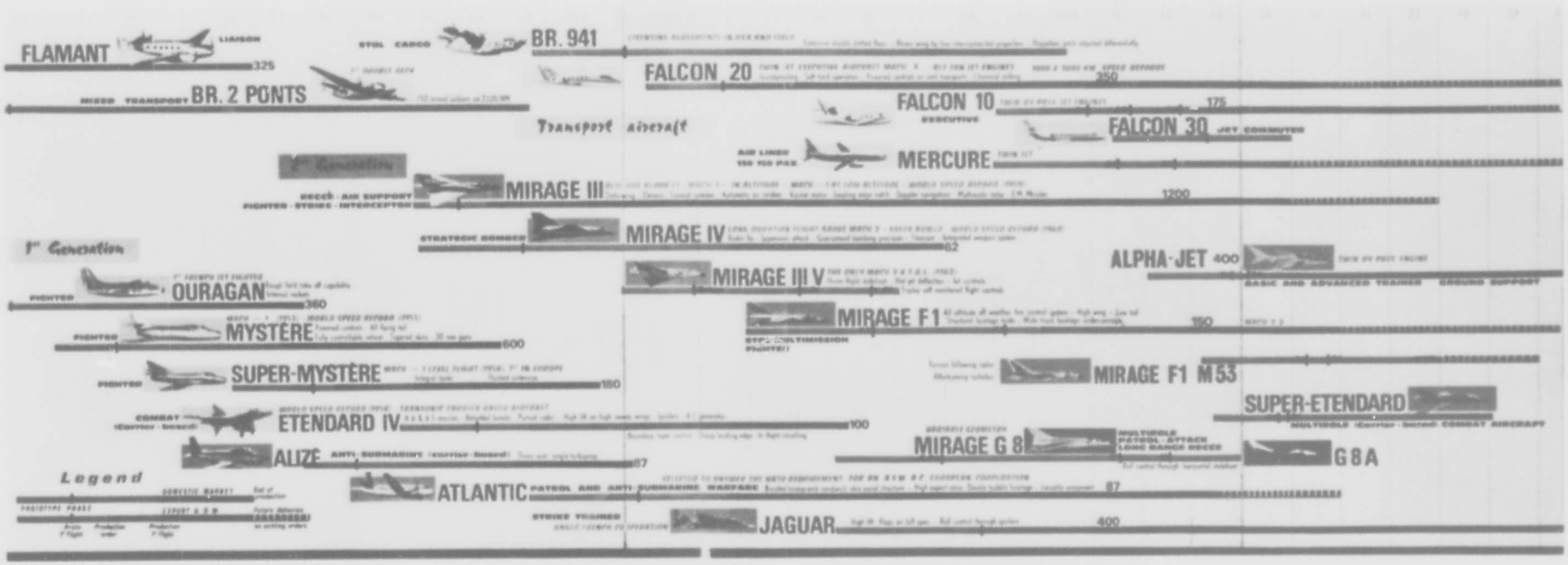


FIGURE 1 : Les principaux appareils construits par les AMD/BA.
AMD/BA productions.

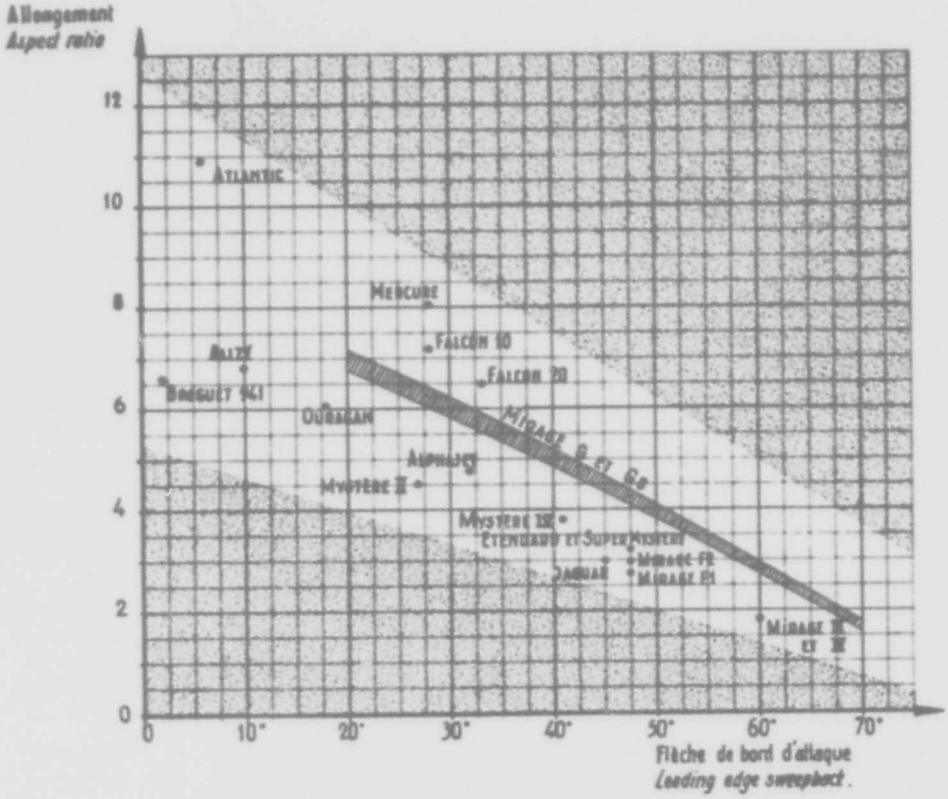


FIGURE 2 : Domaine flèche-allongement.
Sweepback-aspect ratio range.

FIGURE 3 : De l'OURAGAN au SUPERMYSTERE.
From OURAGAN to SUPERMYSTERE.

SURFACE DE REFERENCE	35 m2
FLECHE AU BORD D'ATTAQUE	60°
ALLONGEMENT	1,94
EPAISSEUR RELATIVE	4 à 3,5 %
PROPULSION REACTEUR SNECMA ATAR 9 C	
Poussée sans post combustion	4 400 kg
Poussée avec post combustion	6 200 kg
MASSE EN ORDRE D'EXPLOITATION	6 950 kg
POIDS AU DECOLLAGE AVION LISSE	9 300 kg
POIDS DE DECOLLAGE MAXIMUM	13 700 kg
COMBUSTIBLE EXTERNE MAXIMUM	4 700 l

FIGURE 4 : MIRAGE 5 (version simplifiée du MIRAGE III). Caractéristiques principales.
MIRAGE 5 - Leading characteristics.

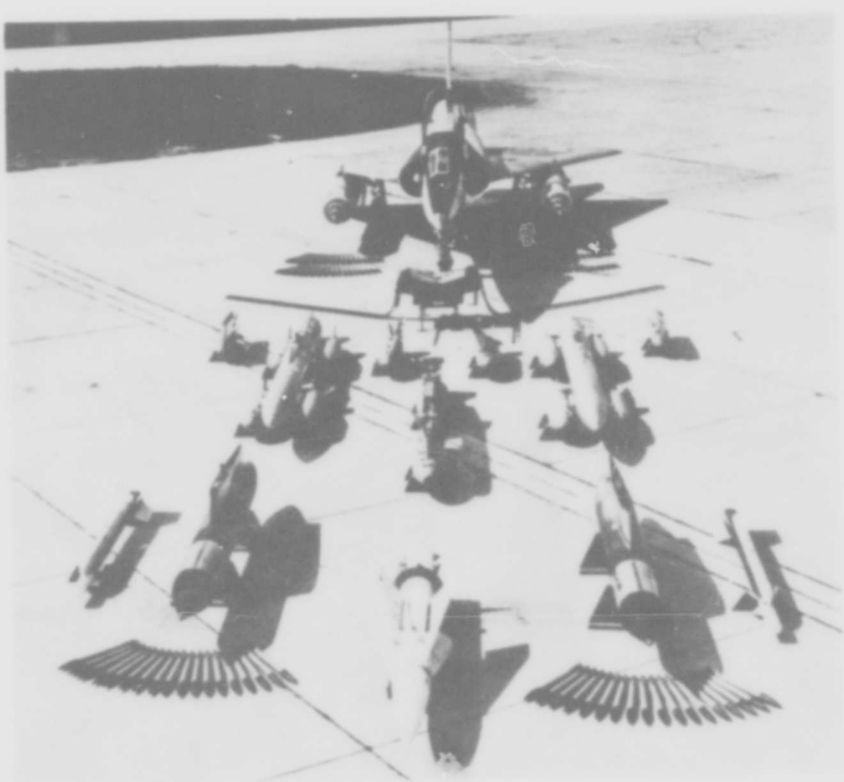


FIGURE 5 : Quelques possibilités d'emport.
Carrying capabilities of the M 5.

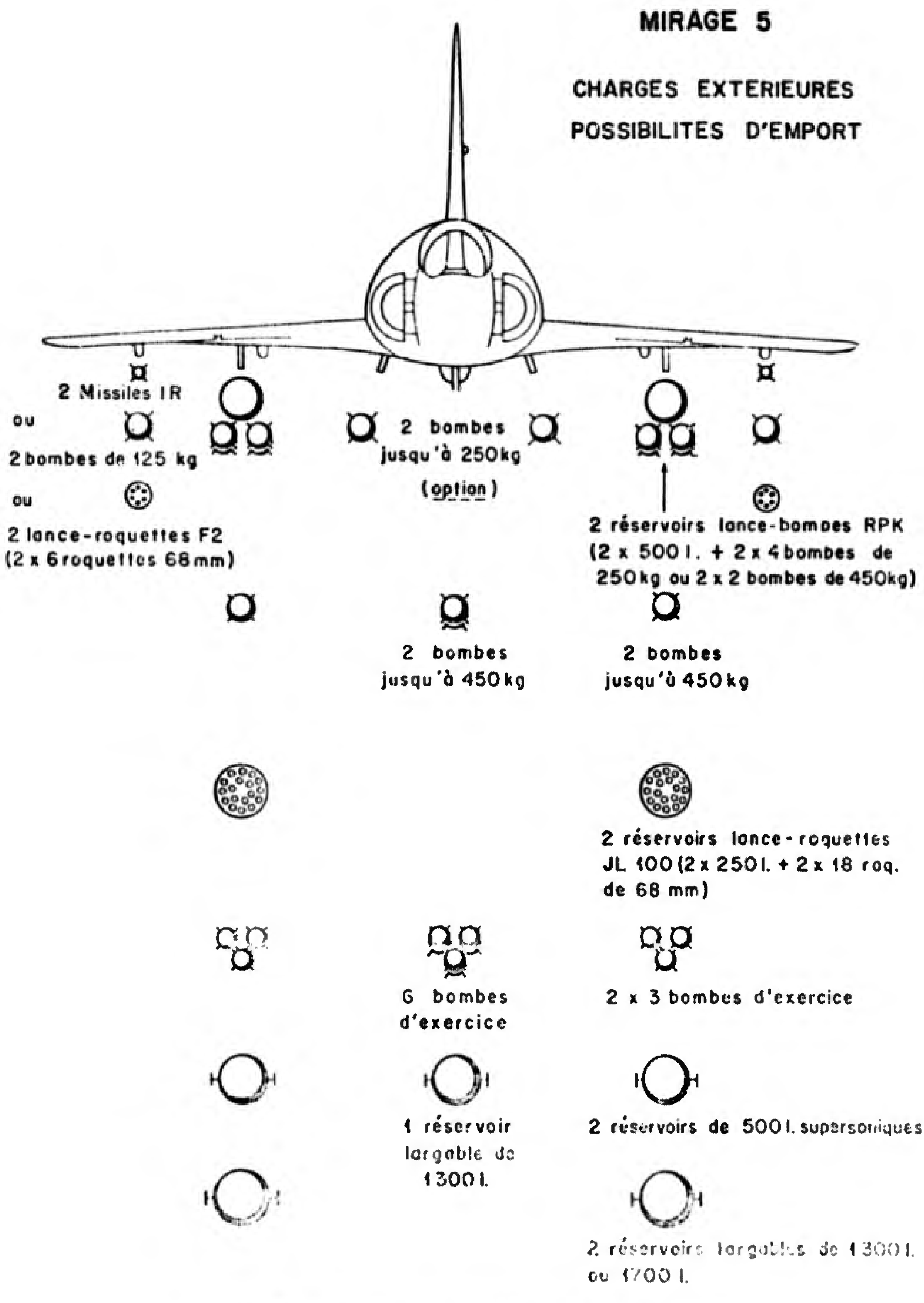


FIGURE 6 : Points d'accrochage et leurs charges.
Pylones and external stores.

FIGURE 8 : Vol en configuration d'attaque à basse altitude.
Flight in ground attack configuration.

LE M 5

EST LE SEUL AVION DE COMBAT MACH 2

CAPABLE DE DECOLLER

D'UN TERRAIN SOMMAIRE

FIGURE 10 : Décollage sur terrain non préparé.
Take-off from unprepared strip.

FIGURE 11 : Sortie des "moustaches".
Extension of the "moustaches".

FIGURE 12 : ETENDARDS en ravitaillement en vol.
ETENDARDS in flight refueling.

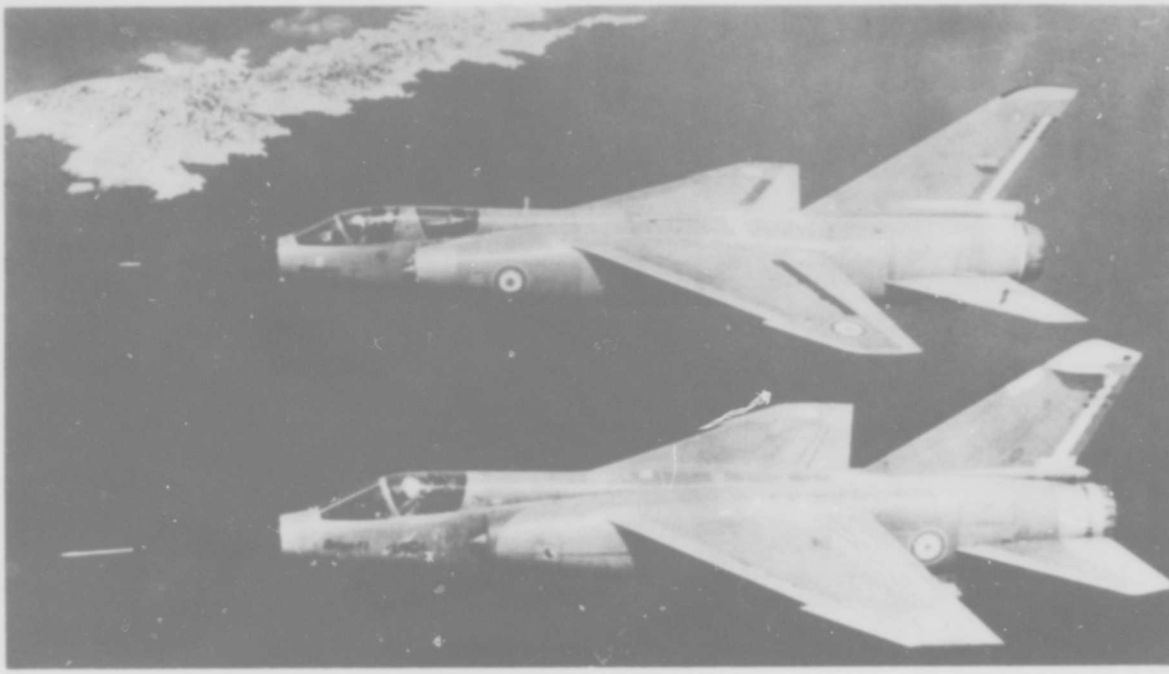


FIGURE 13 : MIRAGE F1 et F2.
MIRAGE F1 and F2.

Entrée en service en 1973

MIRAGE F 1 C : Supériorité et défense aériennes
Attaque au sol

MIRAGE F 1 A : Appui tactique et supériorité aérienne

Tout en conservant ou améliorant la robustesse, la simplicité de maintenance et la supériorité en combat du Mirage III, le Mirage F 1 C apporte des progrès importants sur le plan opérationnel :

- vitesse de combat à MACH 2.2 au lieu de 2
- TEMPS de POURSUITE supersonique TRIPLE
- TEMPS d'ATTENTE TRIPLE
- RAYON d'ACTION DOUBLE en mission d'attaque Lo-Lo
- LONGUEUR de DECOLLAGE DIMINUEE de 23 %
- VITESSE d'APPROCHE DIMINUEE de 20 %
- MANOEUVRABILITE ACCRUE

SUR LE PLAN DE LA TECHNOLOGIE AVANCEE

- gain de masse de structure et de capacité disponible (carburant interne augmenté de 43 %)
- hypersustentation développée
- poussée du réacteur accrue de 16 % avec une masse et un volume pratiquement inchangés
- système d'armes amélioré

FIGURE 14 : Le MIRAGE F1, une nouvelle génération d'avions de combat.
MIRAGE F1, a new generation fighter.

DIMENSIONS	
- Envergure	8,42 m
- Hauteur	4,49 m
- Longueur	15,23 m
SURFACE DE REFERENCE	25 m2
FLECHE DE BORD D'ATTAQUE	47° 34'
ALLONGEMENT	2,8
EPAISSEUR RELATIVE	4,5 à 3,5 %
REACTEUR	
- Poussée sans réchauffe	5020 kg
- Poussée avec réchauffe	7200 kg
MASSE A VIDE EN ORDRE D'EXPLOITATION (avec 2 canons de 30 mm)	7760 kg
MASSE DE DECOLLAGE EN LISSE * (plein interne + 250 obus)	11325 kg
MASSE MAXIMALE AU DECOLLAGE	15200 kg

* Avionique de base

FIGURE 15 : Caractéristiques principales.
Leading characteristics.

COMMUNICATION

V/UHF TRT ERA 7000
UHF TRT ERA 7200

IDENTIFICATION

IFF LMT NRAI 4A

NAVIGATION

TACAN
LMT AN ARN 52
VOR/ILS-MARKER
SOCRAT 3300-5600

AIDES AU PILOTAGE

PILOTE AUTOMATIQUE
SFENA 505
PLATEFORME GYROSCOPIQUE
SFIM 153-7
CENTRALE AERODYNAMIQUE
CROUZET 63
ENREGISTREUR DE CRASH
SFIM

CONDUITE DE TIR

RADAR DE NEZ A FONCTIONS MULTIPLES CYRANO IV
VISEUR THOMSON/CSF 196 (HUD)
BOITIER D'HARMONISATION DU MISSILE A/A MATRA 530
VOLUME CAPABLE D'UN SYSTEME DE TELEAFFICHAGE
CABLAGE COMPLET DES CIRCUITS DE TIR:
CANONS, BOMBES, ROQUETTES, NAPALM, MISSILES A/A

FIGURE 16 : Avionique de base
Basic avionics

IDENTIFICATION

IFF (différents types)

NAVIGATION

VOR / ILS MARKER (différents types)

PLATEFORME A INERTIE associée à un CALCULATEUR NAV/ATTAQUE

RADIO ALTIMETRE TRT AHV 9

OPTIONS RADAR :

- Evitement d'obstacle
- Découpe iso-altitude
- Percée aveugle

CONDUITE DE TIR

ELIMINATION D'ECHOS FIXES THOMSON-CSF

H.U.D A TUBE CATHODIQUE

TELEMETRE RADAR AIR - SOL

AUTRES EQUIPEMENTS

DETECTEUR PASSIF RADAR THOMSON-CSF

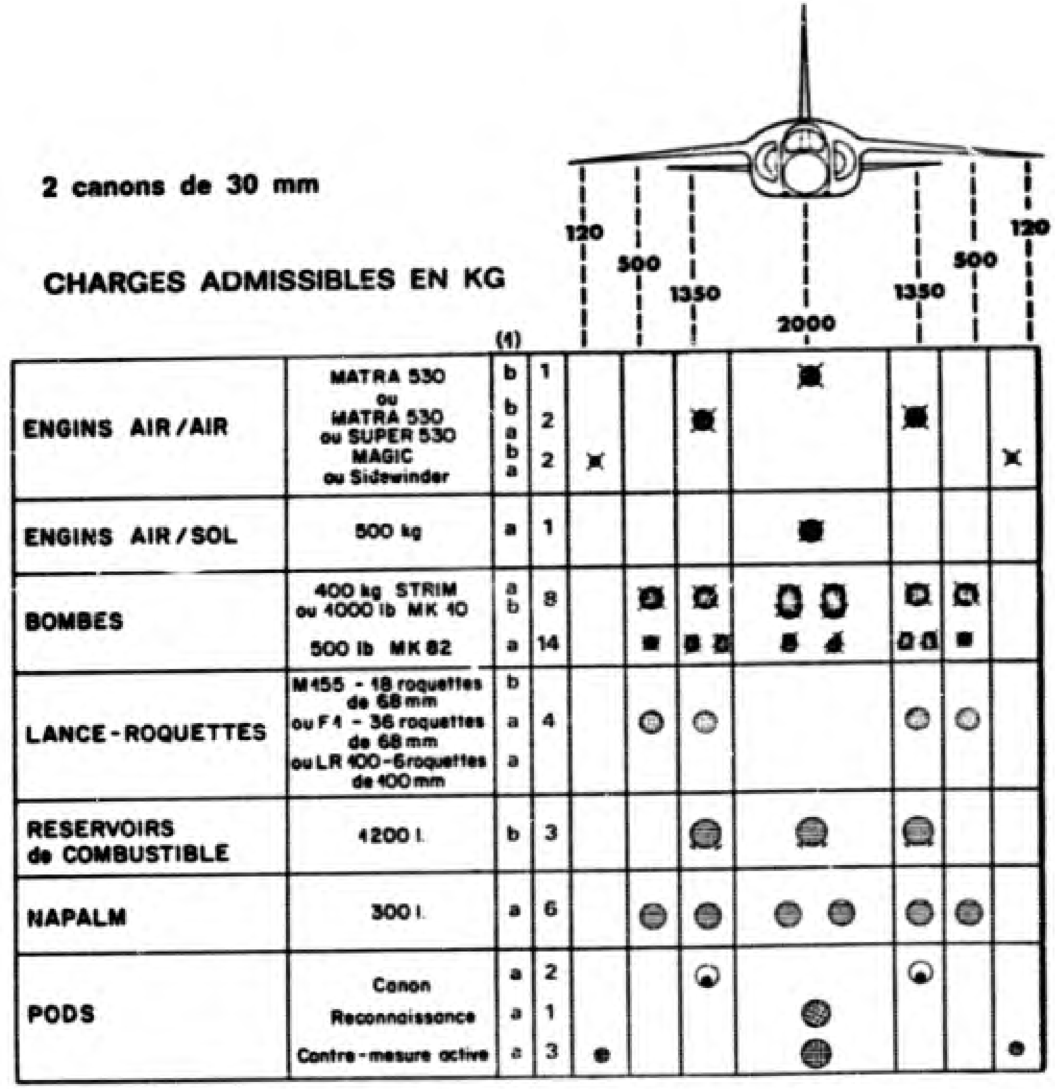
PODS DE CONTREMESURES ELECTRONIQUES

FATIGUE-METRES EMD

FIGURE 17 : Avionique optionnel
Optional avionics

2 canons de 30 mm

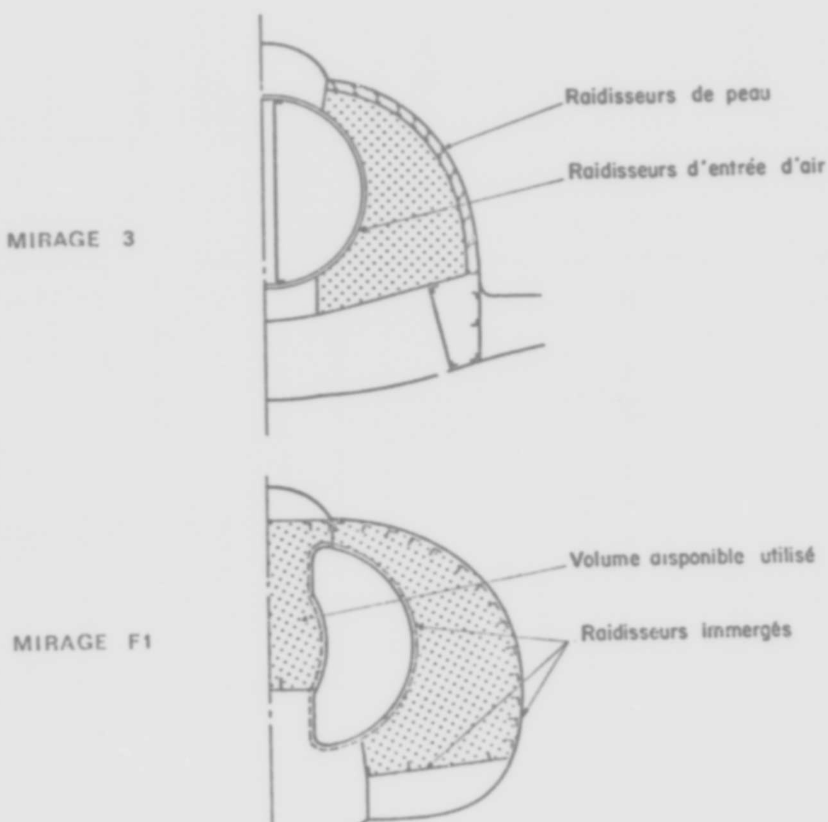
CHARGES ADMISSIBLES EN KG



(1) b : équipement de base   a : équipement additionnel

FIGURE 18 : Possibilités d'emport.
Carrying capabilities.

FIGURE 19 : Vol en configuration d'attaque au sol.
Flight in ground attack configuration.



RESULTAT : CAPACITE SUPPLEMENTAIRE = 1 TONNE = 45%

FIGURE 20 : Réservoirs intégraux de fuselage.
Fuselage integral tanks.

FIGURE 23 : Production en série du JAGUAR à TOULOUSE-COLOMIERS.
JAGUAR production line at TOULOUSE-COLOMIERS.

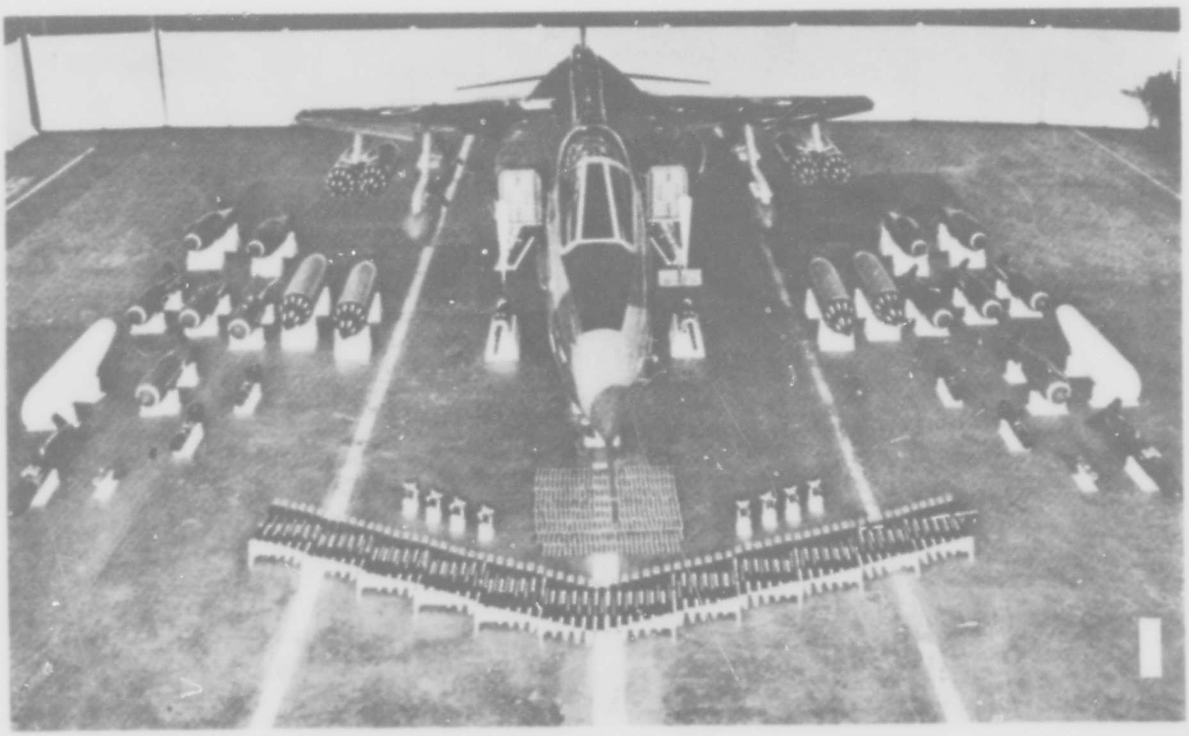


FIGURE 24 : Possibilités d'emport.
Carrying capabilities.

FIGURE 25 : JAGUAR équipé de 8 bombes.
JAGUAR with 8 bombs.

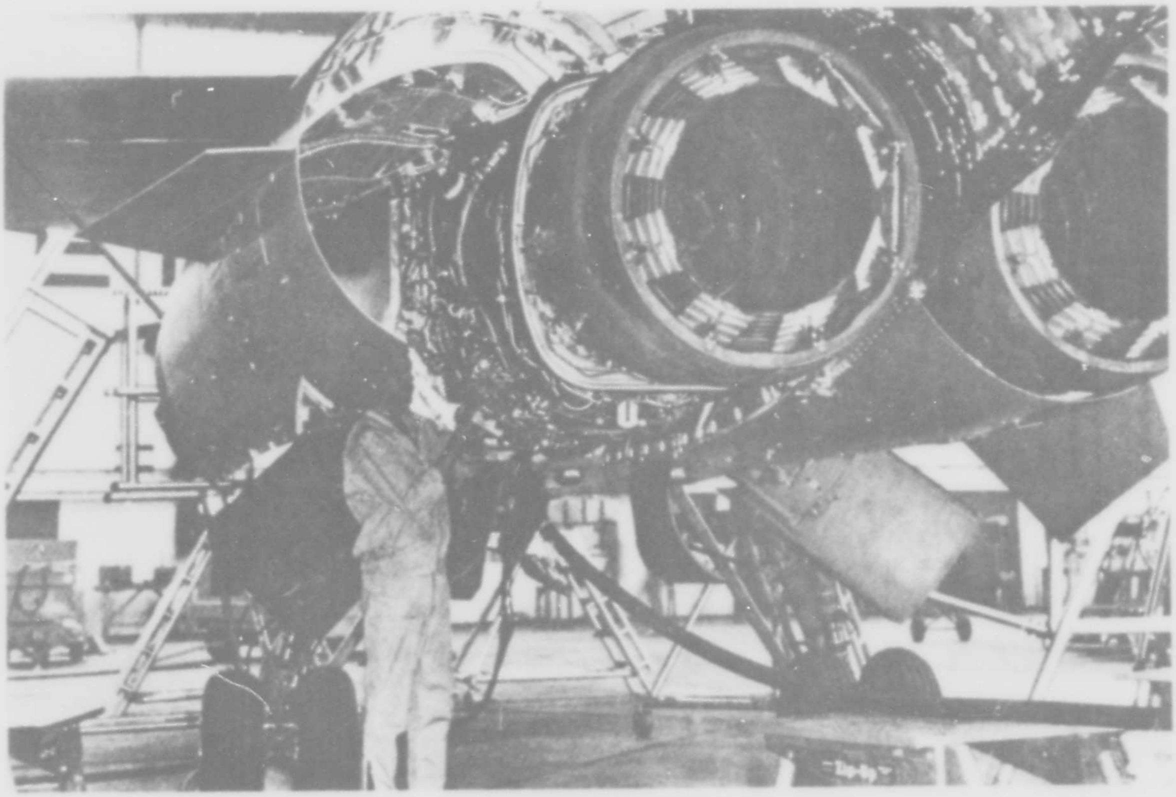


FIGURE 26 : Accessibilité des réacteurs.
Engine accessibility.

FIGURE 27 : En service dans l'Armée de l'Air et la R.A.F.
In service in Armée de l'Air and R.A.F.

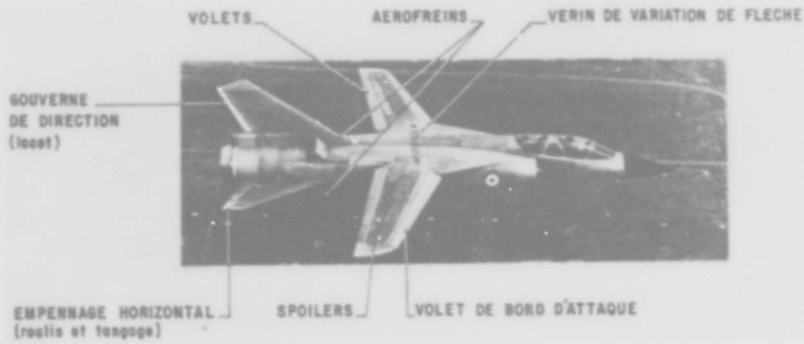
FIGURE 28 : MIRAGE G, G8 - Filiation.

MIRAGE G

OBJECTIFS

— AVEC STABILISATION ARTIFICIELLE : EXCELLENTE PLATEFORME DE TIR

— SANS STABILISATION ARTIFICIELLE : CONTROLE ASSURE DANS TOUT LE DOMAINE DE VOL



- ETUDIEES ET REALISEES PAR AMD / BA
- SERVO-COMMANDES ELECTRO-HYDRAULIQUES DOUBLE-CORPS SECURITE HYDROMECANIQUE
- VERIN A VIS DE LA VOILURE ASSURANT UN NIVEAU DE SECURITE ELEVE ET UNE SYNCHRONISATION PARFAITE
- HYPERSUSTENTATION ACTIONNEE PAR SERVO-COMMANDES DOUBLE-CORPS
- AEROFREINS SANS CHANGEMENT DE TRIM DANS TOUT LE DOMAINE DE VOL

FIGURE 29 : Commandes de vol.
Flight controls.

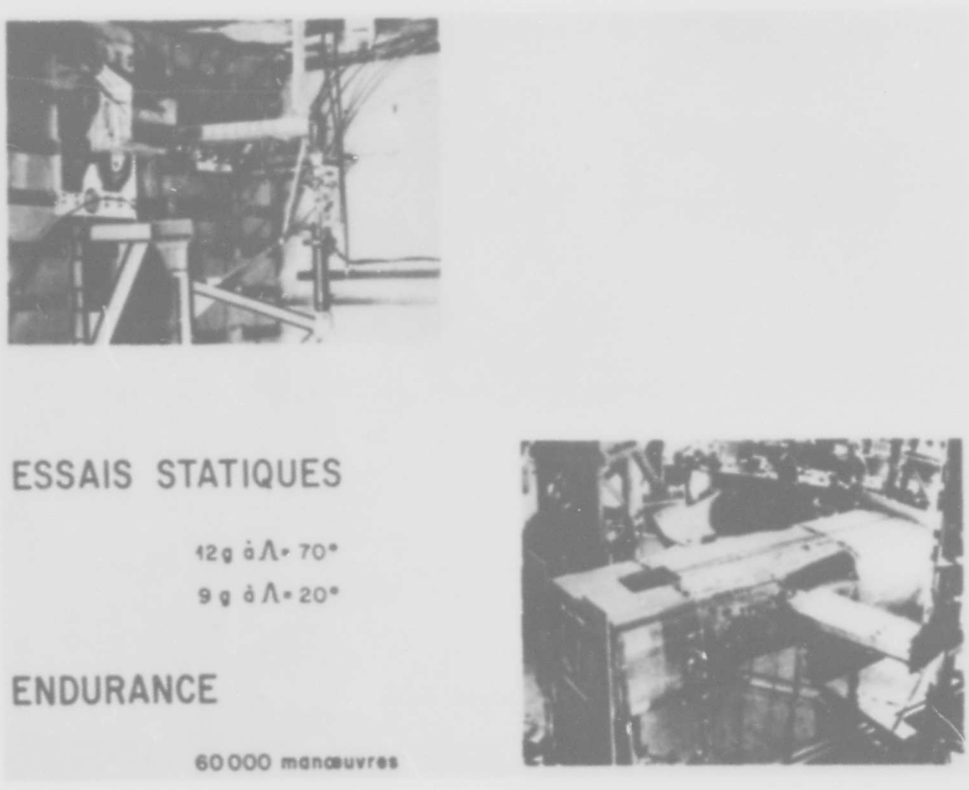


FIGURE 30 : Essais du pivot.
Wing pivot tests.

- ETUDES THEORIQUES ET NOMBREUX ESSAIS EN SOUFFLERIE
- LA POSITION DU PIVOT RESULTE DU COMPROMIS ENTRE DIFFERENTES CONDITIONS
  - . Bonne manoeuvrabilité
  - . Faible sensibilité à la turbulence
  - . Traînée d'équilibrage minimisée pour la mission de base
  - . Faibles changements de trim avec la variation de flèche
- APEX REDUIT ET PUR ASSOCIE A UNE FORTE HYPERSUSTENTATION

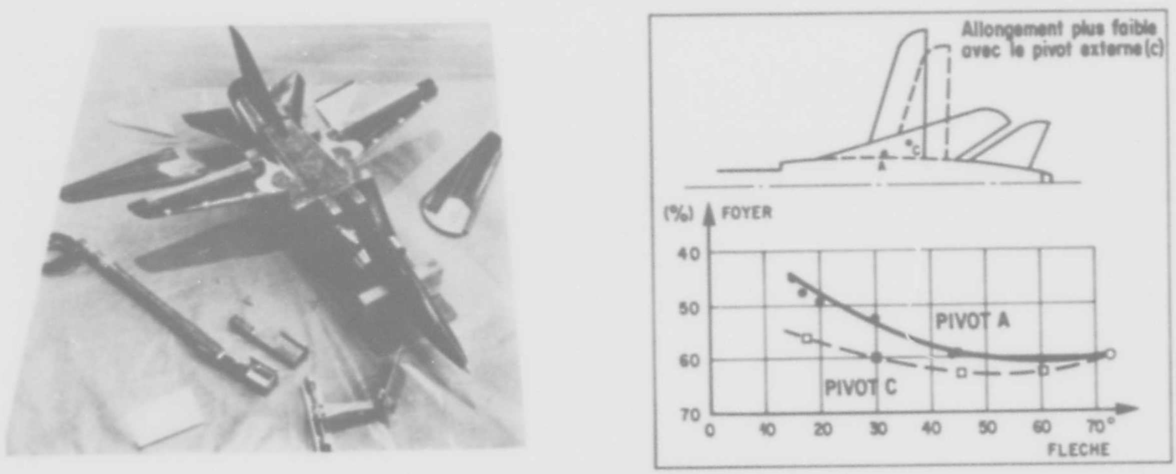


FIGURE 31 : Position du pivot.
Pivot position.

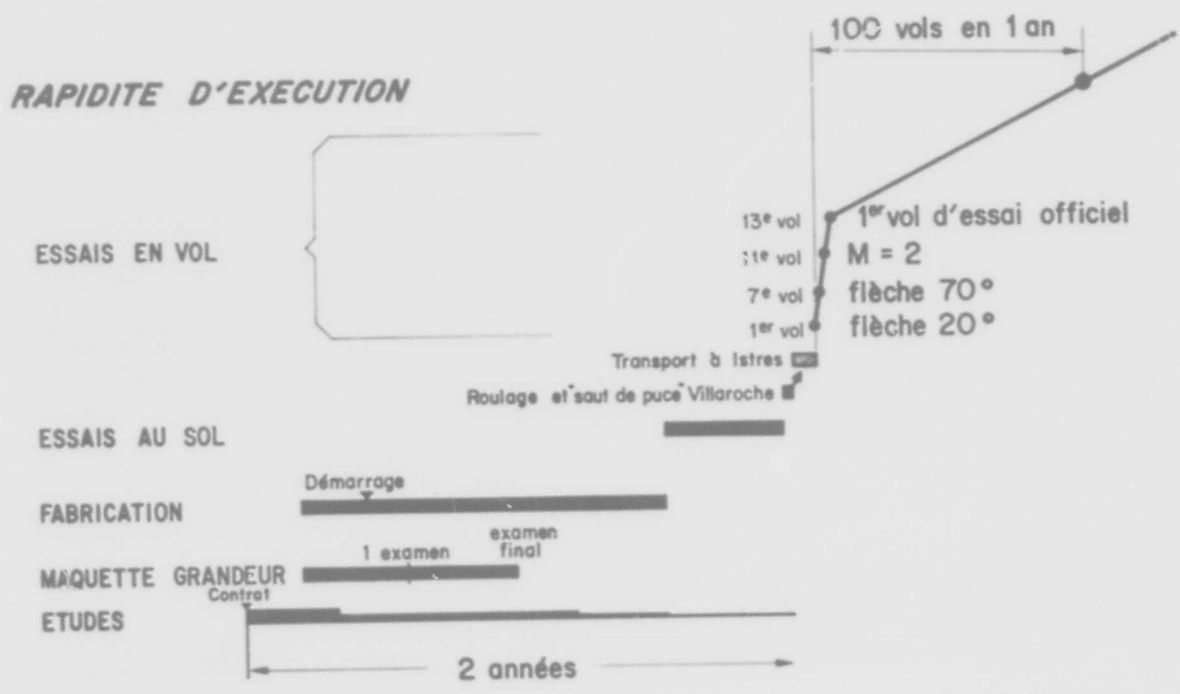


FIGURE 32 : Etapes du programme MIRAGE G.
MIRAGE G milestones.

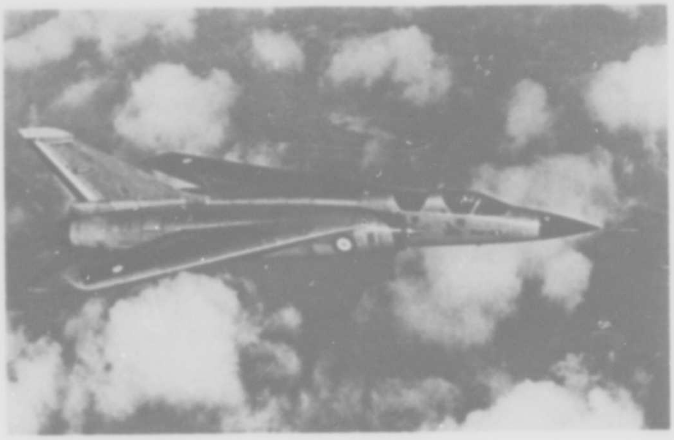


FIGURE 33 : MIRAGE G8 en vol.
MIRAGE G8 in flight.

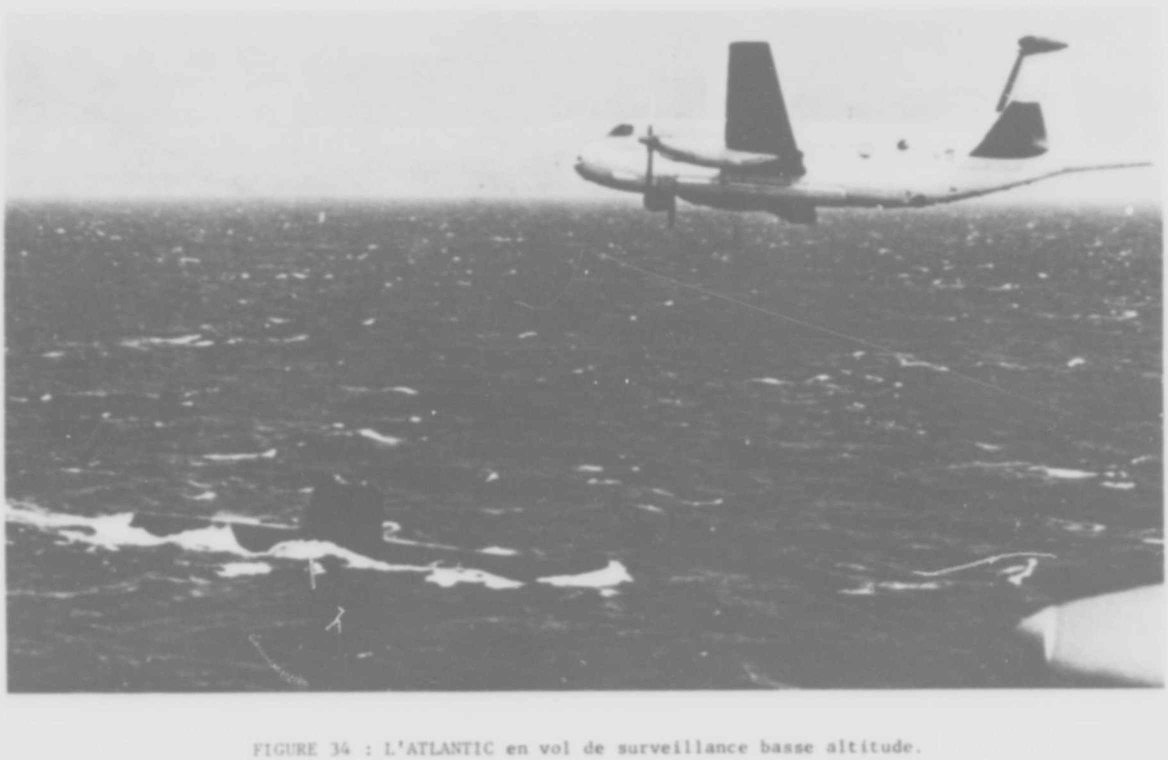


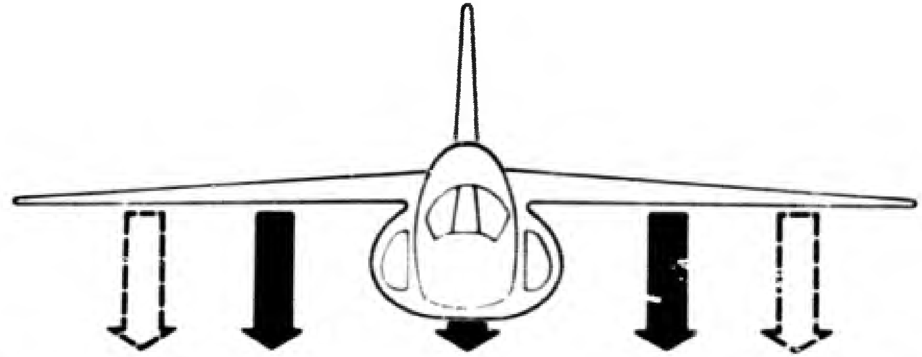
FIGURE 34 : L'ATLANTIC en vol de surveillance basse altitude.
The ATLANTIC at low altitude patrol flight.

FIGURE 35 : Soute à bombes ouverte.
Bomb bay open.

FIGURE 36 : Vue partielle de l'aménagement intérieur.
Partial view of the internal arrangement.

ARMEMENT

POSSIBILITES D'EMPORT



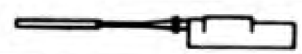
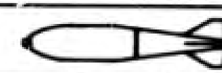
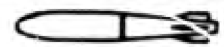
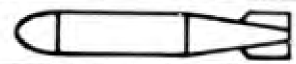
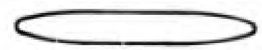
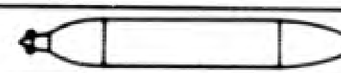
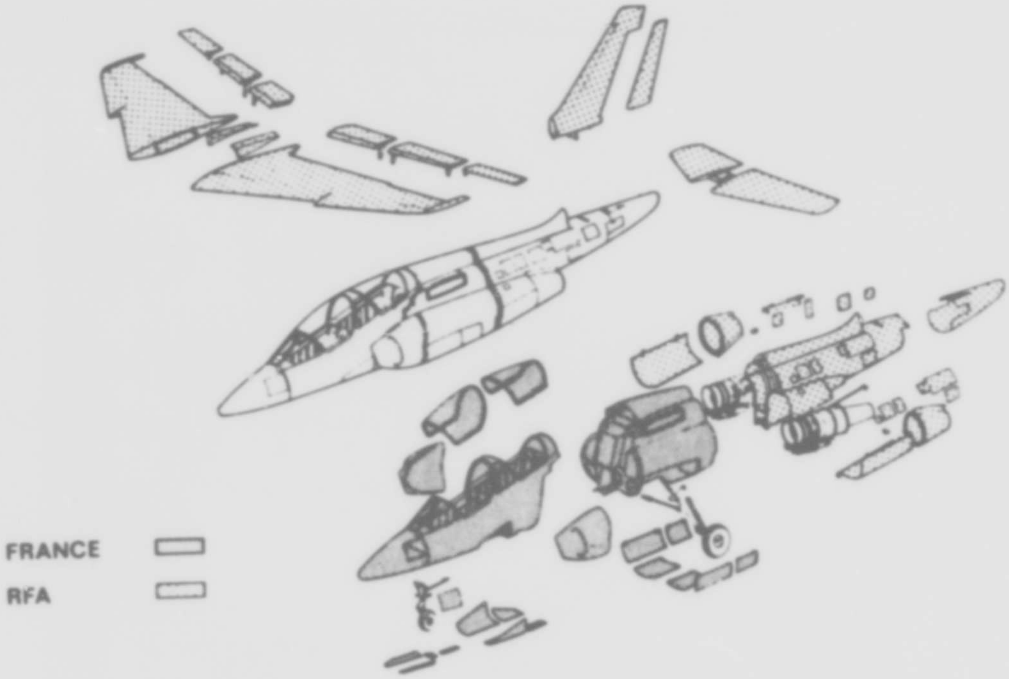
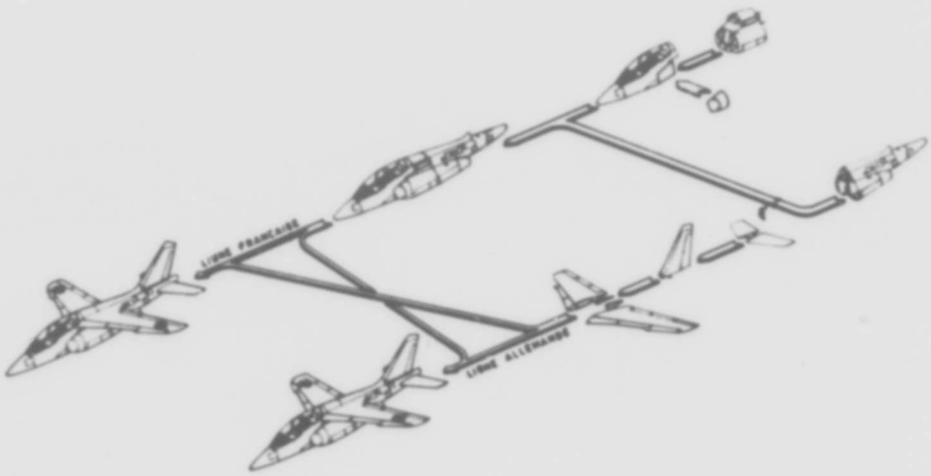
1 canon de 30 mm avec 150 obus	
Bombe de 400 kg lisse ou freinée	
Bombe de 250 kg lisse ou freinée	
Bombes de 125 kg et 50 kg	
Bombe cluster de 625 lb	
Bombes incendiaires de 690 lb et 825 lb	
Lance-roquettes classe 2.75 in 6 - 19 - 36 roquettes	
Combiné lance-bombes/lance-roquettes d'entraînement de 360 lb	
2 réservoirs de carburant largables de 310L	
1 pod reconnaissance	

FIGURE 37 : ALPHA JET : possibilités d'emport.
ALPHA JET carrying capabilities.



Les différents éléments sont réalisés dans les deux pays, sans duplication de fabrication.

The various aircraft parts are built in the two countries without duplication of the production



Deux chaînes de montage fonctionnent :

l'une en France
l'autre en Allemagne

Two assembly lines :

one in France
one in Germany

FIGURE 38 : Plan de production de l'ALPHA JET.
Production schedule of the ALPHA JET.

FIGURE 39 : ALPHA JET en vol.
ALPHA JET in flight.

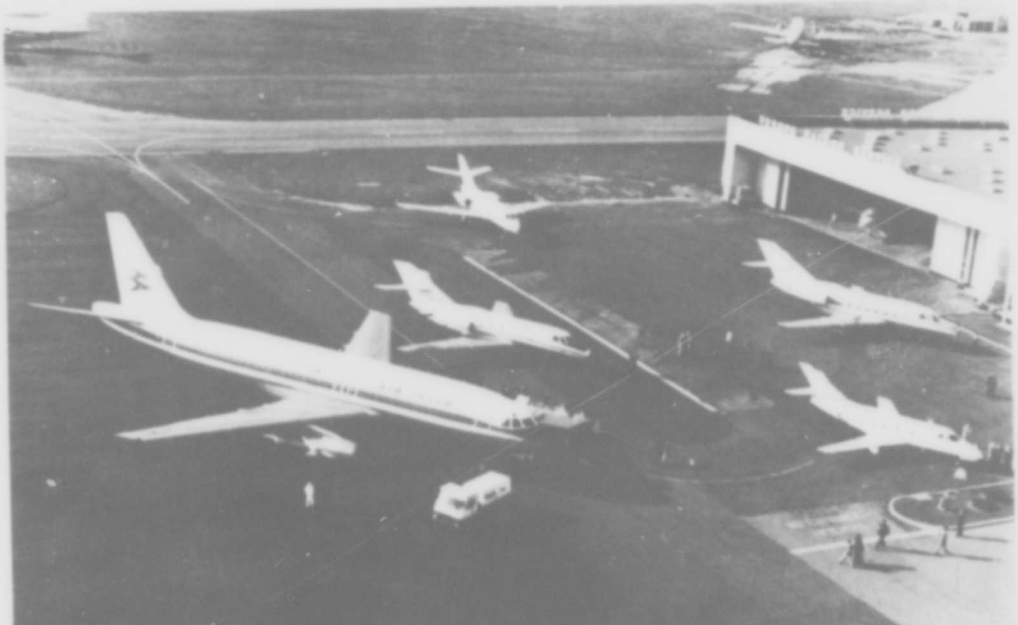


FIGURE 40 : FALCON 10, 20, 30 et MERCURE.
FALCON 10, 20, 30 and MERCURE.

ΔCx EN FONCTION DU MACH POUR Cz = 0,25
ΔCD vs MACH NUMBER AT CL = 0.25

	FALCON 20	MERCURE
FLECHE VOILURE *WING SWEEPBACK*	30°	25°
EPAISSEUR RELATIVE *RELATIVE THICKNESS*		
. Emplanture *Root*	10%	12,5%
. Extrémité *Tip*	8%	8,5%

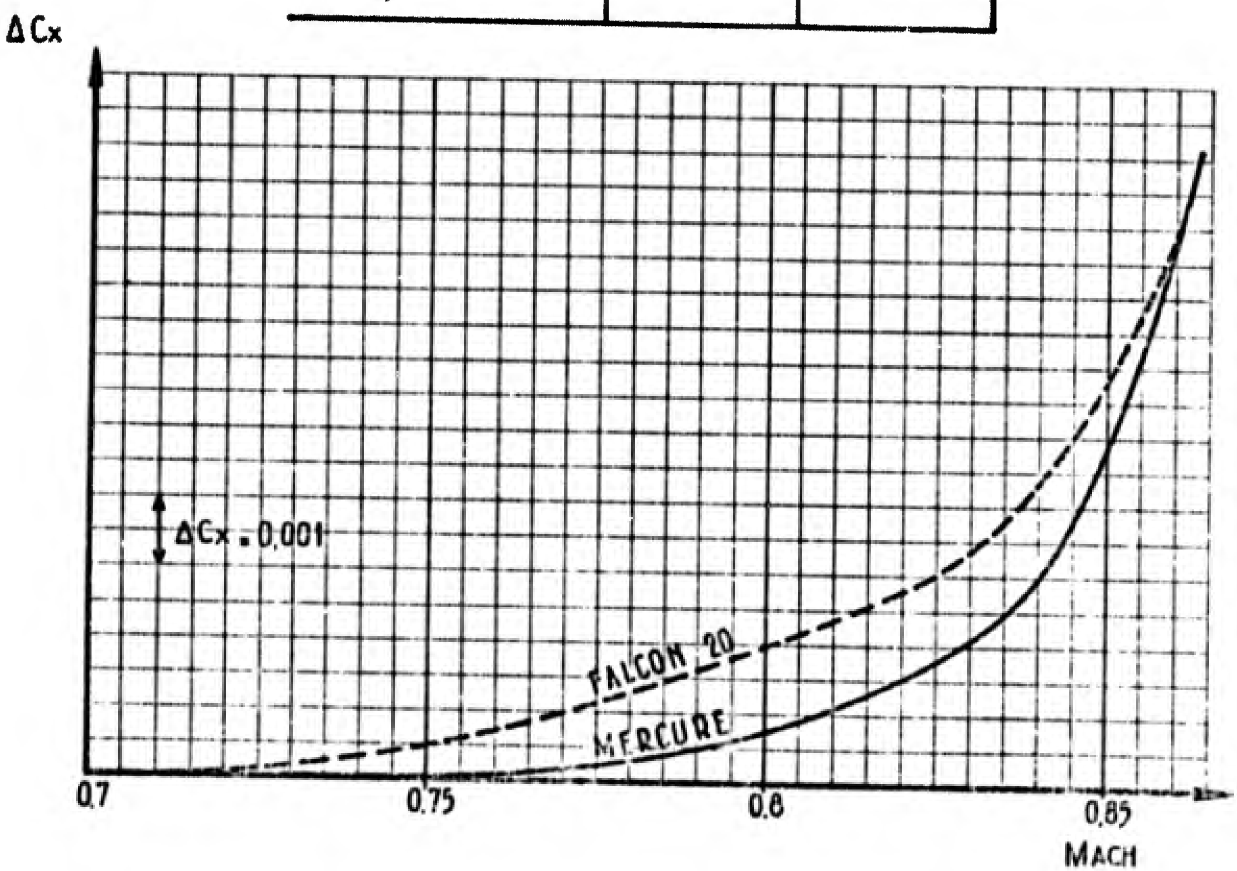


FIGURE 41 : Evolution du Cx en fonction du Mach pour les FALCON 20 et MERCURE.
ΔC_D vs Mach number of the FALCON 20 and MERCURE.

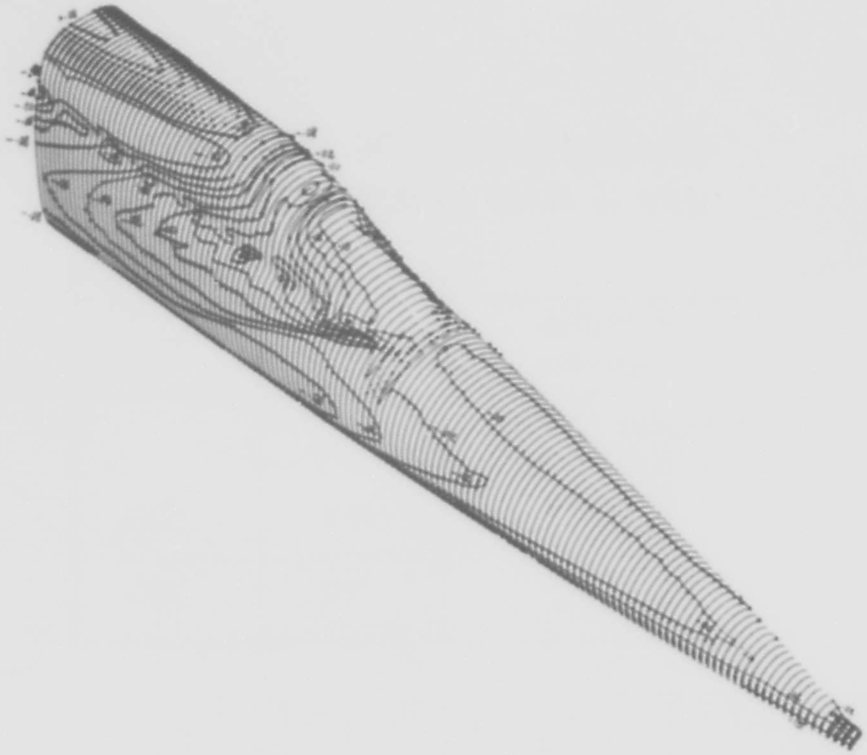


FIGURE 42 : Répartition de pression à incidence élevée, en supersonique sur un nez de fuselage.
Pressure distribution at high angle of attack in supersonic flight on a fuselage nose.

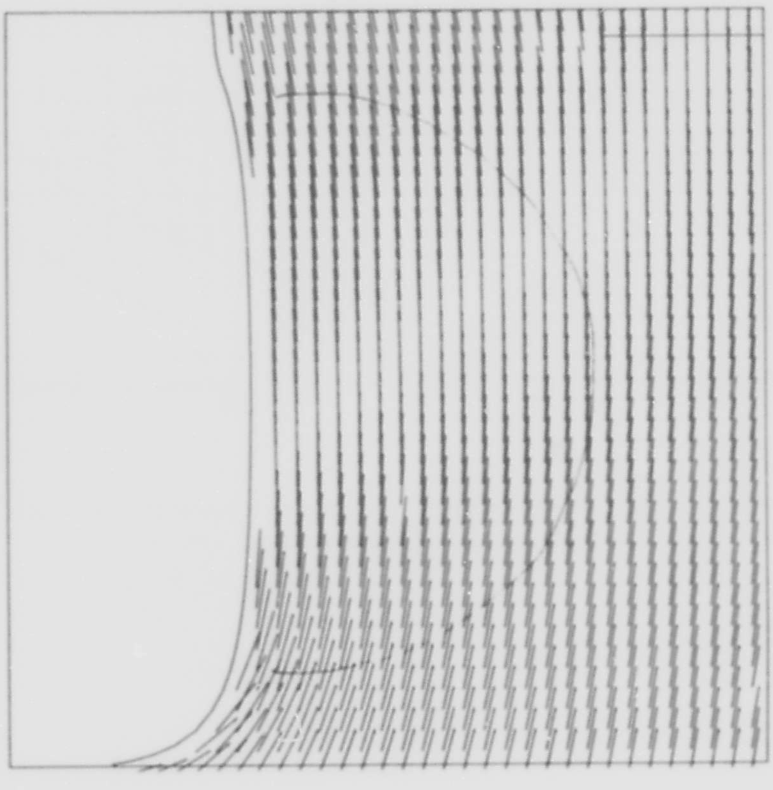


FIGURE 43 : Champ de vitesse dans le plan de l'entrée d'air.
Velocity field at the inlet.

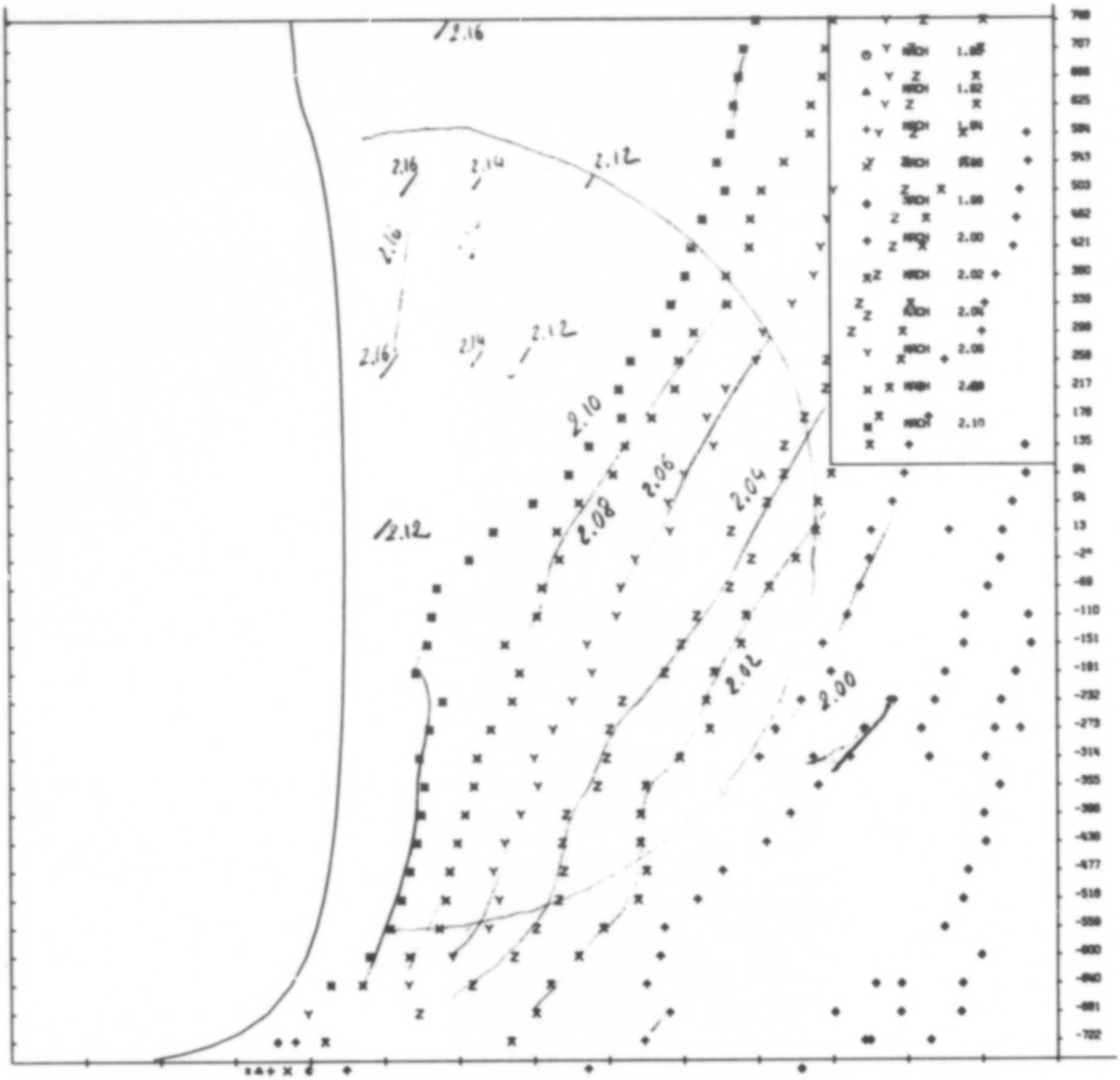


FIGURE 44 : Répartition des Mach dans le plan de l'entrée d'air à incidence élevée.
Mach number distribution at the inlet at high angle of attack.

FIGURE 46 : Répartition de pression sur avion complet.
Pressure distribution on complete aircraft.

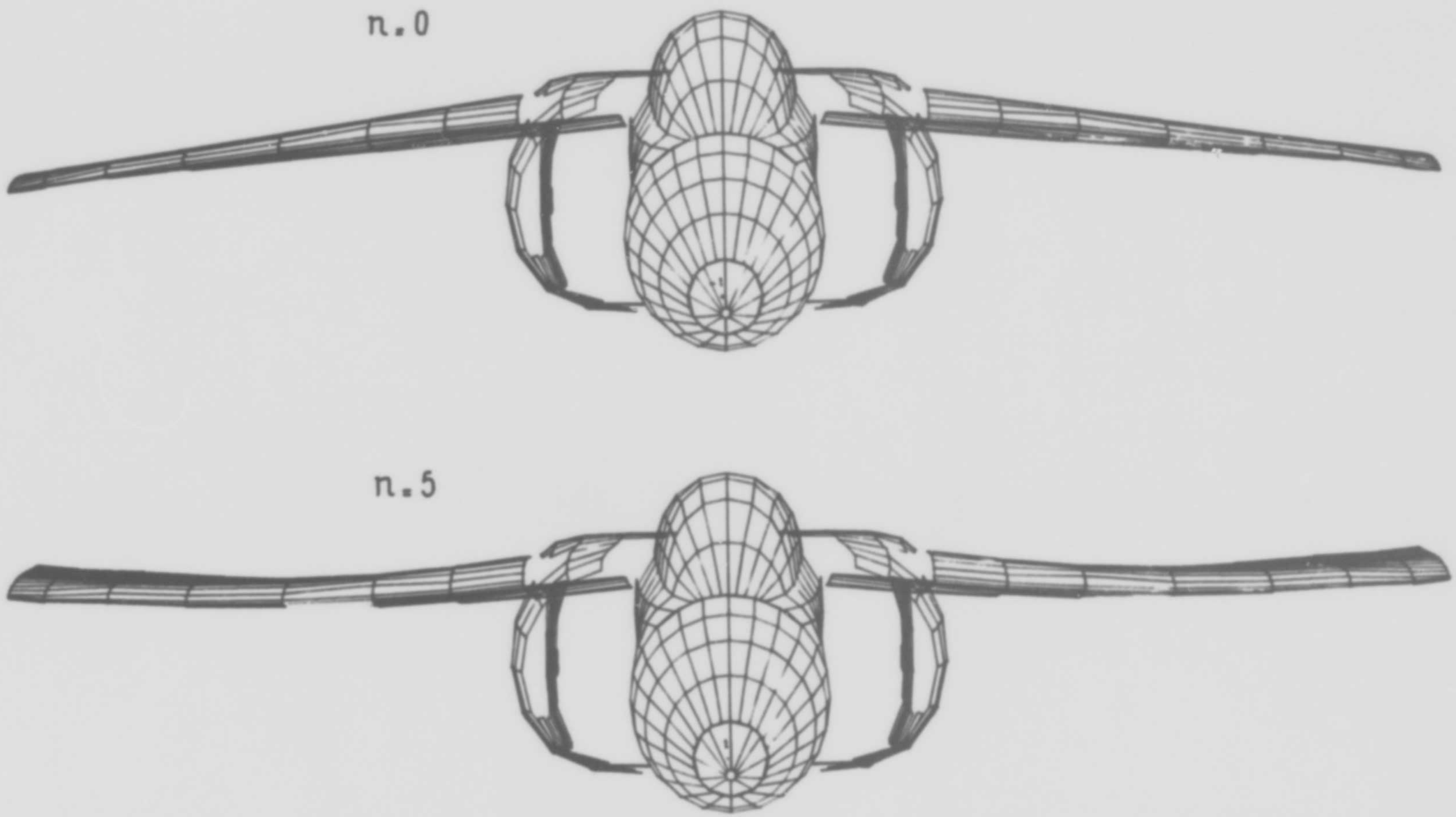


FIGURE 47 : Avion rigide et déformé sous les charges aéroélastiques.
Rigid and aeroelastically deformed aircraft.

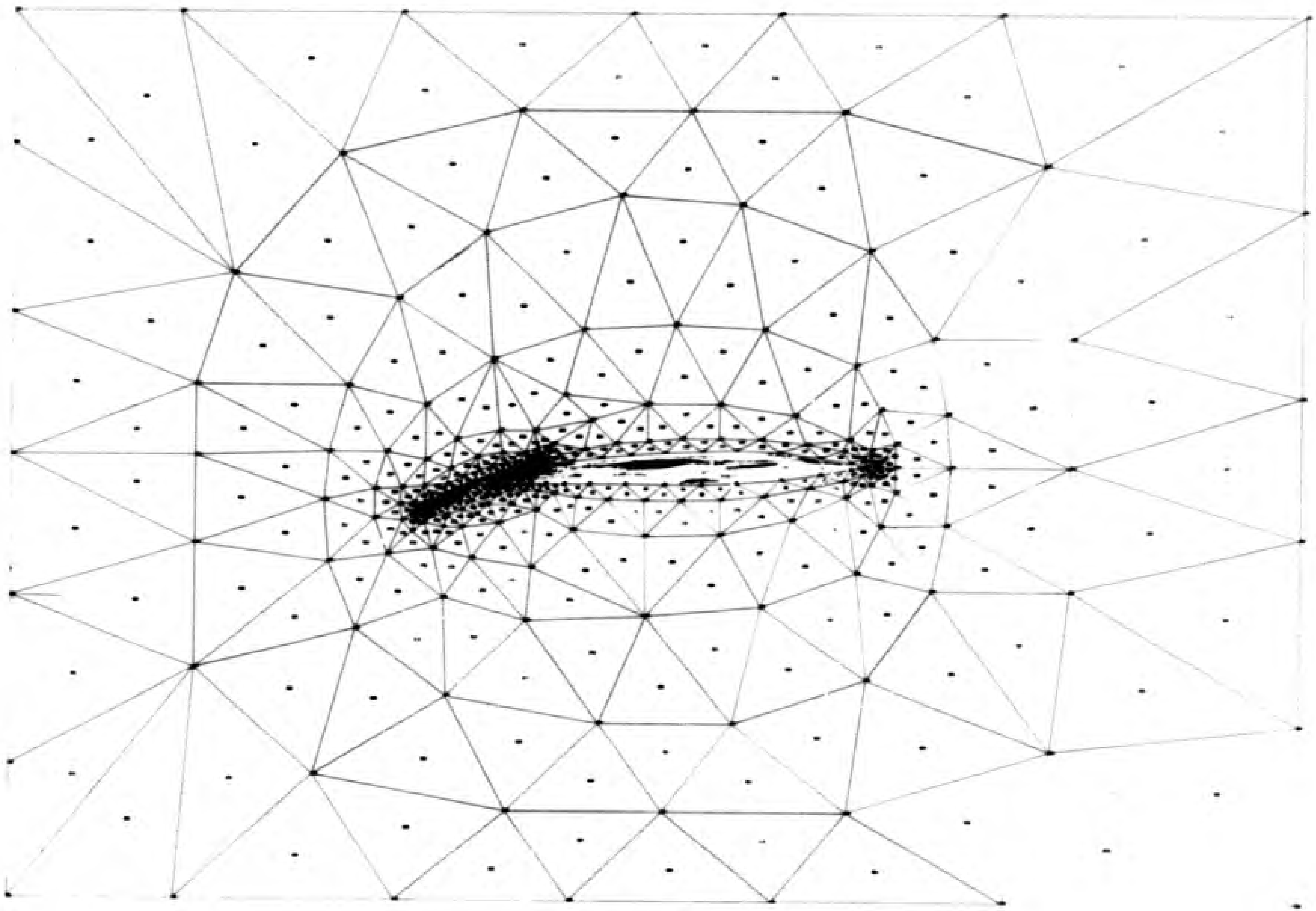


FIGURE 48 : Triangulation autour d'un profil à bec de bord d'attaque.

Triangulation around an airfoil with slat.

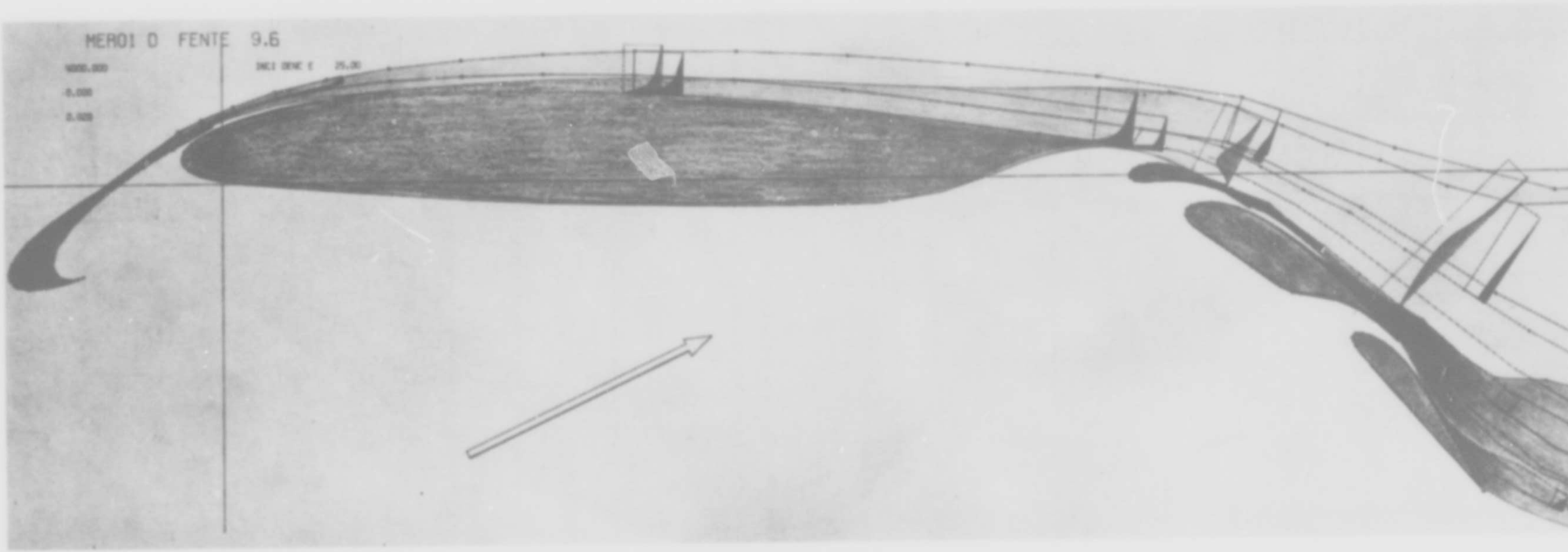


FIGURE 49 : Couche limite et sillage d'une configuration fortement hypersustentée.

Boundary layer and wake of a developed multiple flap airfoil.

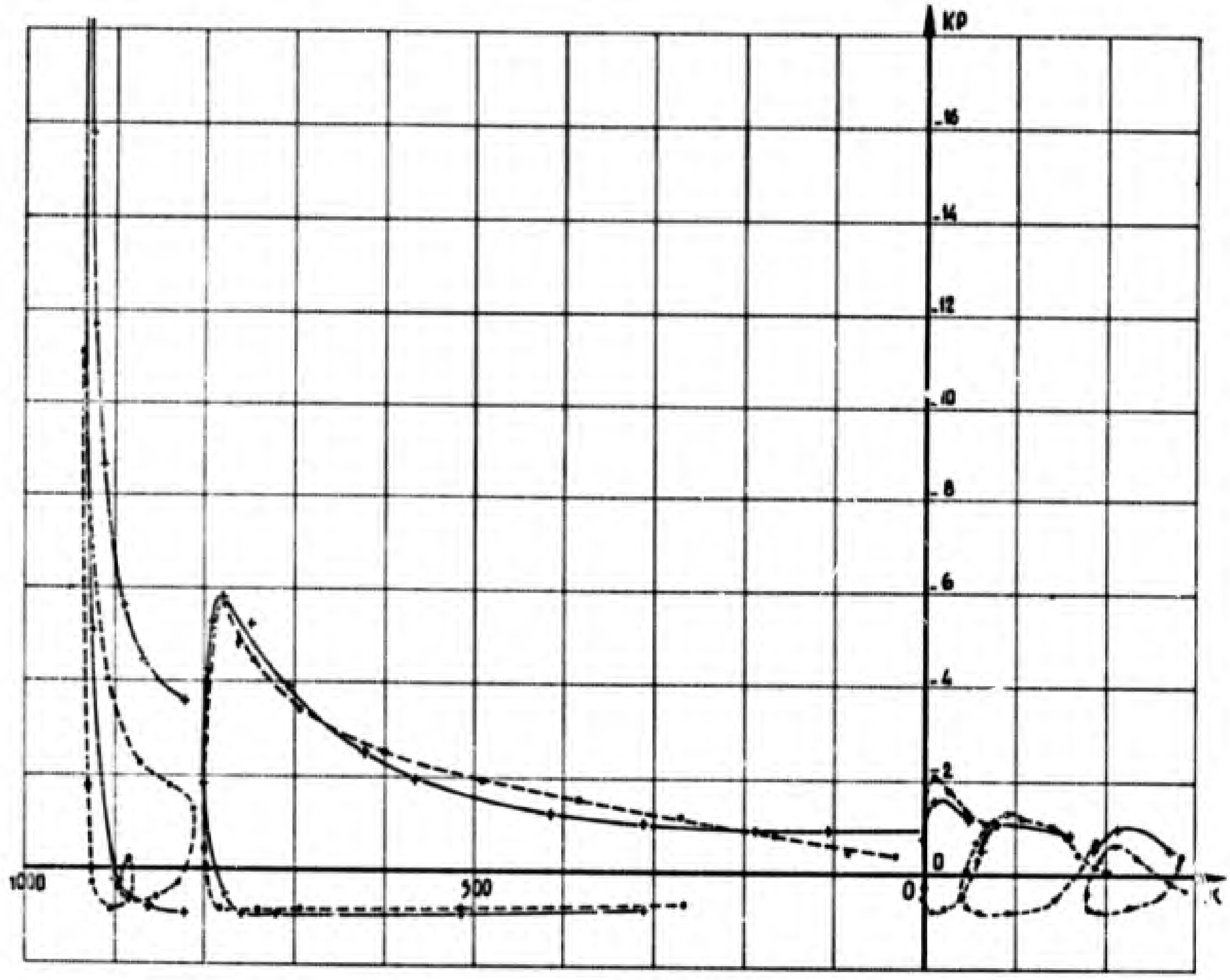


FIGURE 50 : Comparaison d'une répartition de pression théorique et expérimentale sur un profil décroché.
Comparison of theoretical and experimental pressure distributions on a stalled airfoil.

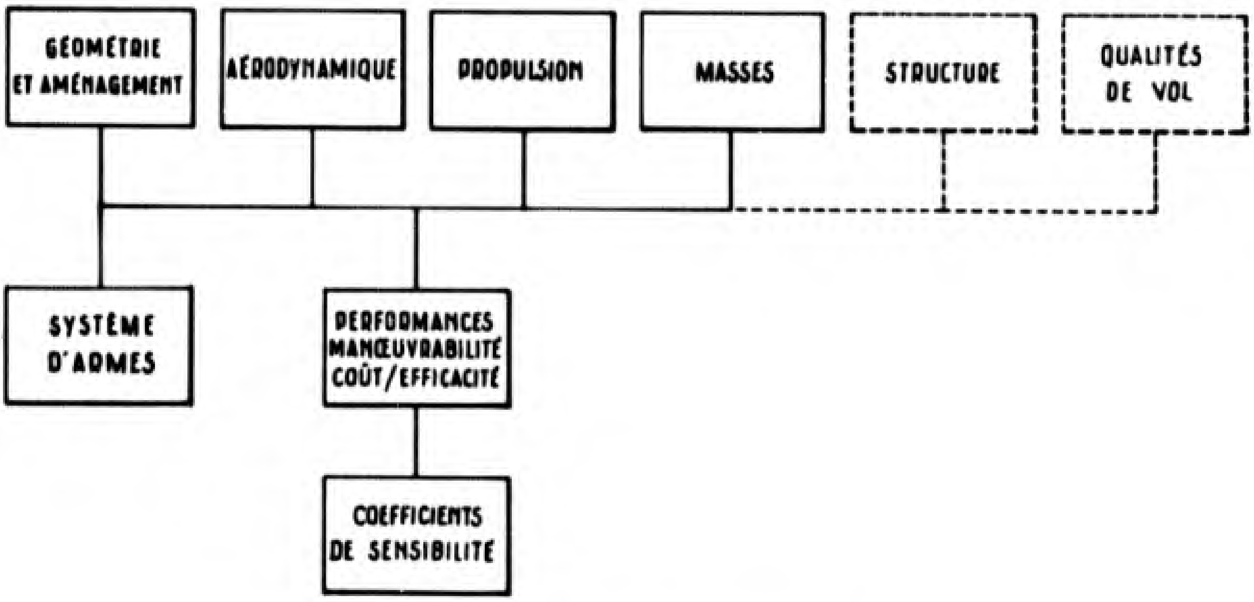


FIGURE 51 : Bloc-diagramme pour l'étude préliminaire.
Block-diagram for the preliminary design.

PRELIMINARY DESIGN OF CIVIL AND MILITARY AIRCRAFT AT AVIONS MARCEL DASSAULT-BREGUET AVIATION

by

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Avions Marcel Dassault-Bréguet Aviation
92214 Saint-Cloud

1. INTRODUCTION

In the last 25 years the AMD/BA company has designed and developed about forty civil and military prototype aircraft. This large family includes on the military side fighters (interceptors and tactical support), a supersonic VTOL, a Mach 2 bomber, carrier aircraft, trainers, variable geometry fighter-bombers, ASW and STOL transports. The civil aircraft are represented by executive aircraft by the Falcon 30 third level transport and by the short and medium-haul transport Mercure, the largest and the heaviest built by the company.

Most of these prototypes have been followed by a more or less important production. (Figure 1.) To date about 3000 airplanes have been built and others are at an initial production stage. An interesting particularity is that the participation of the company in manhours to the whole production is less than 50%. The production is realized with subcontractors, under national or international cooperation or under license.

Figure 2 representing the sweepback and aspect ratio range covered by these aircraft illustrates their large variety. Figure 3 shows another important conceptual characteristic: the progress in the continuity. The data gathered by the design and development of these aircraft constitute an unlimited source of information.

Associated to this double quality of progress and continuity are other factors, constituting the basis of the success of most of the programs:

- quality of the teams in the design office and workshop,
- efficiency and rapidity. Quick basic decisions and low response time to adjust when necessary,
- constant product-improvement research,
- reduced number of prototype drawings, detailed production drawings permitting an easy production breakdown.

This dynamic policy wherein time is a governing factor is significantly facilitated by a fruitful cooperation between Official Services and manufacturer. The procedure used in France by which a Project engineer is appointed within the STAé as well as a Project Officer within EMAA for military aircraft has constantly been improved throughout the years. Because of his technical background the Project Engineer can as readily be understood by the manufacturer as by the military staff; he develops and speeds up the necessary communication between all interested parties. The aircraft manufacturer is thus in a position to take part in program definition, to undertake feasibility studies; this gives him an opportunity to contribute to the orientation and evolution of specifications. The final program will thus be borne out of a cooperation effort involving the investigation and solution of critical problem areas. Needless to say that as early as this stage all constraints such as:

- budgetary previsions,
- program time-scalings,
- level of technology to be applied,

as well as the domestic and foreign market potential are being considered.

Following these overall remarks we propose to discuss the organization of the preliminary project staff (which will eventually provide the nucleus for the project team).

The guiding lines which should not be considered as based on rigid principles are described below:

- basic options and final decisions are taken at the highest level, Mr Marcel Dassault intervening personally at the various design and development stages;

- responsibility for the program is vested in a project engineer or program manager. The project development is achieved by the project team under his management and by the respective specialized Divisions with which he keeps in permanent contact. It should be noted that important decisions are supervised and examined in detail by the Technical Manager;
- the preliminary project team made up of a small number of engineers and technicians is mainly concerned with synthesis and drawing activities;
- analysis work is usually carried out by specialized Divisions and in particular by the Division of Advanced Studies which undertakes aerodynamic investigations and fairly detailed wind tunnel tests at this early stage.

We now reach the critical problem involving the choice of initial configurations. Rather than performing systematic parametric studies especially in the case of military programs, we prefer to assess as thoroughly as possible a number of different design concepts meeting the program requirements and possibly optimized (hard point designs).

Through a primary selection it will be possible to eliminate definitely poor solutions.

The next iterative step on the remaining solutions involves a more detailed evaluation which will afford a more accurate reassessment as well as a further selection. For the iterations subjective data such as experience gained on a given concept, the availability of certain types of tooling etc. . . are duly considered.

After a few iterations the number of satisfactory proposals is gradually reduced leaving out either one single solution, which is rarely the case, or two or three competing solutions, which although different confront us with a problematic choice of compromises.

At this crucial point the company management has to use its good engineering judgment and experience to make the most adequate choice among the few proposals retained. The general layout of the design will then be defined and the options adopted will determine the future of the program.

This shows the importance of comparative preliminary studies and in particular that accurate data are required to perform these evaluations. Any incorrect assessment might bring about an erroneous judgment and lead to a faulty choice.

In order to carry out these investigations as rapidly as possible the preliminary project team should be provided with:

- accurate weight estimation procedures (kept up-to-date),
- experimental data giving the aerodynamic characteristics of the most important proposed configurations,
- theoretical aerodynamic computer programs as well as correlation methods allowing the results to be applied to all designs considered (variation of planforms, thickness ratio, fuselage size and shape, evaluation of interaction between various elements, effect of propulsion, etc. . .),
- data on propulsion: either computer programs for the engine(s) or overall performance data in case the propulsion system is to be defined.

Concerning the weapon system, it is assumed that preliminary design studies have been carried out in the form of

- operational research,
- compatibility and optimization,
- military load definition.

Due to the strong interference between avionics, armament and airframe, many adjustments will be needed on all these elements in the preliminary design process.

Consequently a whole team of system specialists will work at this stage under the direct authority of the Project Manager.

Working in cooperation with the engineering staff a team of draughtsmen will be charged with detailed design, sizing, interior arrangement and structural concept in order to bring the state of advancement of the preliminary project different versions to the same level.

During this study stage a permanent liaison will be kept with the specialized Departments to ensure that the technical and technological solutions adopted are pushed to the limit, if not beyond, of the "state of the art".

This procedure should allow a maximum use of gained experience without too much attachment to the past and permit to draw the highest benefits from technical innovations.

2. AMD/BA COMBAT AIRCRAFT

2.1 Ouragan, Mystère and Mirage Aircraft

We shall now examine how these principles have been implemented in various fields and firstly to combat aircraft which have for a long time ranked as the leading area of activity.

The first jet combat aircraft design by Société Dassault, the Ouragan, made its maiden flight on February 28, 1949. Because of the continuity policy practised by Société Dassault it may be considered as having been for many years the only genuine prototype. Without dwelling on this airplane let us only mention that from the Ouragan fitted with the Nene centrifugal compressor jet engine – of which 360 units were built, the company using the continuous extrapolation process shown on Figure 3 has successively derived the Mystère II, Mystère IV and Supermystère models. During this six-year development phase, the thickness ratio has been brought down from 13 to 6%, the sweepback was changed practically from 0 to 45° and the engine thrust increased from 2 300 to 4 500 kg.

As many as 800 Mystère aircraft have been built and some of them are still in use in several countries. Towards the years 1952–54 a change took place when it appeared that evolution could not proceed along the previous lines. Their design at that time could not provide easily for greater speed, their military load and range were too limited and because of the increasing length of runways they were requiring they became more vulnerable.

A critical point had also been reached as regards powerplant. On the one hand there was no engine developing enough power to obtain the speed (Mach 2) considered as imperative. On the other hand in order to specify a policy concerning propulsion and develop a good size engine, appraisement of the respective advantages of the single and twin engine aircraft was required; this problem was then even more complex than at the present time.

The potential difficulties in the development of low thrust engines have not been underestimated, and the Official Services have consequently decided their parallel development with the SNECMA Atar engine equipped with afterburner.

Moreover to reach the high Mach numbers and altitudes necessary to the interception use of rockets has been required. The lightweight interceptor gave rise to a close competition; the Dassault Company has proposed two designs, one twin engine, 2 Vipers as interim engines + rocket, the other a single engine airplane with rocket. The twin engine version has been ordered, and this delta wing aircraft the Mirage I, made its first flight in June 1955 and within less than a year was flying with afterburner and rocket.

Its successor, the Mirage II aircraft equipped with the final power plant had already reached an advanced stage of realization when for various reasons developed below the company decided to modify its policy, discontinue production and come back to a single engine model using the wing which had already been produced for the Mirage II model.

This action was carried out very speedily: the Mirage III-001 model fitted with the Atar 9B + afterburner + rocket and with a fuselage which already included the area rule flew in November 1956, only 9 months after the decision had been taken. Very soon, it reached supersonic speeds in level flight, M = 1.5 on the 5th flight and Mach 2 on October 24, 1958. Finally this aircraft won the competition and opened the road to the multimission combat aircraft Mirage III. Several versions were later developed and at the present time some fifty different models are operated in 18 countries. The total number of aircraft delivered or ordered is about 1,500.

Let us now analyze the initial decisions as well as the reasons behind the change of policy:

- comparative studies, carried out during the construction of the Mirage II, have shown that a combination of single engine + rocket gives a better thrust to weight ratio, at a given wing loading, than a twin engine airplane, a definite advantage for an interceptor;
- difficulties arose in the development of the low thrust engine, with afterburning in particular. On the other hand the Atar engine both with and without afterburner presented promising improvements;
- misgiving about the operational limitations of the rocket (which proved unfounded) led the Design Office to investigate the possibility of eventually omitting it in view of the expected development of engines and through aerodynamic innovations which were beginning to loom;
- the flight test results acquired on the Mirage I confirmed the validity of the tailless delta wing concept.

In view of the combined conditions it was logical that the company should decide to keep the delta wing and adopt the single engine + rocket with the ultimate objective of coming back to the pure jet engine.

To close this part of our discussion let us add that the operational requirements included a number of provisions which were to have a more or less decisive impact on the future evolution towards polyvalence. So the French policy resisted the prevailing fashion of the airplane/missile combination. Even in the first production versions the Mirage III was equipped with two 30 mm DEFA guns.

Moreover the relatively stringent runway requirements led to the adoption of a large wing area. The low wing loading thus acquired together with engine development afforded the high manoeuvrability so successfully demonstrated in dogfights (Figure 4 to 11).

There are nevertheless areas in the flight envelope where the tailless aircraft cannot compete with the tailed model using current technology. In particular cases where these areas would be considered as important for operation, tailed airplanes or advanced technology will be required.

2.2 Lightweight Tactical Support Aircraft

Before proceeding further with the history of the Mirage airplanes which is far from being concluded let us review the lightweight tactical support program which was launched one year after the interceptor program and was not as successful as its predecessor. Several companies had again been competing among those Dassault and Bréguet today merged. These two companies both won the competition with the Etendard II and the Bréguet 1100 (2 Gabizo engines, sweepback wing with high lift devices, takeoff and landing on short grass runways) respectively. One year after the French lightweight tactical support program NATO initiated a similar program under the term of an international competition. Bréguet and Dassault ranked among the winners together with Fiat. However despite the successful development and production of several prototypes the changes introduced in French aeronautical policy caused the NATO and French light tactical support programs to be discontinued and replaced by other options. The excellent subsonic performance acquired in the meantime by the Mirage III was somewhat instrumental in reaching these decisions. The only survivor in this family was the Etendard IV, an air-carrier fighter; it was difficult for air carriers to cope with tailless aircraft. About one hundred of Etendard IV aircraft were manufactured and the modernized model – the Super Etendard – will make a new career as it will make up the second generation of air-carrier fighters (Figure 12).

2.3 Mirage (continued)

The Mirage III aircraft was used as a flying model for two very different extrapolations: the Mirage IV and Mirage III-V aircraft. The first, a Mach 2 twin engine bomber, is the French "Force de dissuasion" aircraft built in 62 units operated since 1963. The Mirage III-V powered by a P and W-SNECMA TF 306 propulsion engine and 8 Rolls Royce RB 162 lift engines is the only VTOL aircraft to have reached Mach 2 to date. As the interest shown by the military for VTOL aircraft in the early 60's had considerably decreased the program was discontinued in 1967. Together with the production of Mirage III and IV the AMD/BA company has undertaken – in cooperation with the Official Services – the study of new concepts of combat aircraft with improved performance throughout the flight envelope. The development of more and more sophisticated weapon and navigation systems, advances in propulsion and especially the lower specific fuel consumption obtained with the military turbo-fan engines have made it possible to consider the development of an all weather fighter-bomber with a somewhat longer radius and capable of operating at high speed and low altitude without being detected. This type aircraft must have a high wing loading which is inconsistent with the delta wing. To obviate the need for long runways the aircraft must have efficient high lift devices. Moreover its flying qualities should ensure safe high angle of attack flight regime. These requirements led to the development of the Mirage F2 prototype with high lift devices on the leading and trailing edges of its sweptback wing; high wing and low tail. The 18 ton. full load airplane was fitted with a 9 ton. TF 306 jet engine. It was intended to develop this aircraft into an operational all-weather fighter.

After further investigation, this program was abandoned mainly for lack of a suitable powerplant. The policy for combat aircraft was thereafter re-oriented along three lines:

- interceptor, successor of Mirage III,
- training and ground support,
- variable geometry fighter-bomber.

2.4 A New Generation of Combat Aircraft: the Mirage F1

From the aerodynamic design standpoint this single-seater fitted with a 7.2 ton. thrust Atar 9K 50 engine was derived from the Mirage F2 aircraft (Figure 13). Its flight envelope has been extended for both high and low speed in relation to Mirage III performace: definitely better takeoff and landing performance and also its manoeuvrability at some altitudes and Mach numbers. With its large fuel capacity this aircraft has an increased interception time and is capable of prolonged missions throughout the altitude range. Primarily designed for interception its ability to transport various loads (7 tie-up points) makes it a polyvalent airplane. (Figures 15 to 22.) The French Air Force has placed an order for about one hundred aircraft and more orders have been placed or are expected to be placed by several other foreign countries. The first production models became operational in 1973.

While the SNECMA M 53 turbofan engine for high Mach Numbers is under development a Mirage F1 version fitted with this engine is being manufactured. As compared to its predecessor this engine offers a better thrust, thrust-to-weight ratio and specific consumption; these improvements should benefit the F1 performance, especially manoeuvrability, supersonic acceleration and radius.

2.5 Jaguar Training and Ground Support A/C

For chronology sake we shall again leave the Mirage and outline the historical background of the Jaguar which was born out of Franco-British cooperation for the airframe as well as for the powerplant. The French operational requirements for the Trainer and Ground Support Tactical Aircraft were issued in 1964 and the design competition was won by Bréguet as regards the airframe. According to a cooperation agreement signed shortly thereafter by France and Great Britain an order for 150 units was respectively provided by each country. This quantity was subsequently increased to 200. Work was organized on a collaborative basis between Bréguet and the British Aircraft Corporation for the airframe and between Turbomeca and Rolls Royce for the engines in 1965. The highlights of the final program are mentioned below:

- operation from short runways (800 to 1,000 m) for ground support,
- radius of action for Lo-Lo-Lo mission: 450 n.m.,
  radius of action for Hi-Lo-Hi mission: 750 n.m.,
- supersonic performance,
- maximum military payload: 4,500 kg.

The aerodynamic concept was derived from the Bréguet 1001 design – NATO light tactical support – with improvements mainly for high angle-of-attack flight regime. 40° sweepback, high wing, relatively low tail afforded by the tail anhedral. Full span high lift devices with lateral control ensured by spoilers and horizontal tail differential deflection. The aircraft is equipped with two 3,350 kg thrust Rolls Royce/Turbomeca Adour powerplants. Five versions were designed:

– Tactical support	(A)	– single-seater,	French
– Trainer	(E)	– two-seater,	French
– Strike A/C	(S)	– single-seater,	British
– Advance training	(B)	– two-seater,	British
– Naval version	(M)	– single-seater,	French

The latter variant was cancelled in favour of the Super-Etendard.

After some development work due to the fact that the airframe, engine and a considerable number of equipment were prototypes, the Jaguar became operational in 1973 and to date about seventy five airplanes have been delivered to the "Air Forces" of both countries. Negotiations are under way with several countries for export purposes. (Figures 23 to 27.)

2.6 Mirage G, G8 and G8-A

The third area of activity of the new aeronautical policy was the variable geometry airplane. The extension of the concept of polyvalence to more and more various and often contradictory missions led the USA to the adoption of the F 111. In spite of a different operational environment the flexibility offered by the variable geometry induced various European Air Forces to define operational requirements which were met under the best conditions by variable geometry. Anglo/French cooperation was initiated on these program lines towards the years 1965–67 but has not succeeded. Rather than undertaking only partial studies, Société Dassault decided the design and development of an experimental prototype which made its first flight in November 1967.

The first flights of this 15 ton. aircraft powered by a TF 306 9 ton. thrust engine revealed no basic difficulties particularly as might have been expected from the variable geometry. This remarkable result was achieved through several factors:

- implementation of the principle of continuity wherever possible so as to avoid unnecessary innovations (wing/tail position, shape and general arrangement of fuselage, powerplant installation) (Figure 28),
- conventional flight controls, except for the all-electric spoiler (Figure 29),
- special care brought to design and mechanical development of pivot and sweep control unit (Figure 30),
- large scale use of theoretical aerodynamic numerical methods,
- very extensive wind-tunnel testing (Figure 31).

The aircraft flew at supersonic speed at its 5th flight; complete variation of the sweepback took place on the 7th flight and on the 11th flight it reached Mach 2 (Figures 32 and 33).

Low speed performance proved the validity of the high lift devices. Despite a relatively low thrust-to-weight ratio and high wing loading the aircraft was taking off on 600 m runways and landing on 500 m runways. Manoeuvrability proved to be excellent throughout the speed range and very tight turns were demonstrated in deceleration combined with reduction in sweepback.

As it was confirmed through flight testing that the promises of variable geometry had been fulfilled, design work started on operational versions. One of the conclusions reached on that occasion is that this formula is not valid below a certain size. This entailed the requirement that due to the military engines available the aircraft had to be equipped with two powerplants, which was in conformity with the Etat-Major ideas. Following these studies two twin-engine (Atar 9K 50) aircraft were ordered in 1968 and the first of these aircraft, the G8, flew for the first time in May 1971, and the second A/C in 1972. The development was once again very expeditious and the opportunity of having two prototypes was used for the study of new shapes of air intakes and nozzles. Accessorily, these tests revealed that a variable geometry airplane constitutes a good large-scale flying model for fixed geometry aircraft. The data acquired during flights with different sweepback angles have been analyzed and will permit the development of a new generation of aircraft.

We will now end the history of the Mirage family and deal with other achievements.

2.7 Bréguet Atlantic

This anti submarine warfare aircraft orginated from a NATO competition issued in 1957 and won by Bréguet over some thirty competitors. The program established by the International Management Board included long distance and long duration surveillance missions. For the surveillance phase exceptional handling qualities were required at low speeds which called for a relatively reduced wingloading. A large variety of jettisonable loads transported in a large bomb-bay were to be used for the detection and attack of submarines at very low altitude. For this type of missions the best suited powerplant among the limited number of existing types was then (and even now) the Rolls-Royce Tyne engine. This utilization was satisfactory for cruising at 300 kts at 30 000 ft as well as for low speed low altitude detection. The conventional aerodynamic design, the use of power-controls and thorough powered model wind-tunnel testing made it possible to avoid almost completely aerodynamic development in flight. On the other hand the new technology of the structural design based upon the wide use of honeycomb in primary structures entailed a few difficulties which have now been solved (Figures 34 to 36).

The aircraft was built from the design stage, in cooperation between Bréguet, Sud-Aviation, Fokker and Dornier. This was the first important aircraft built under international cooperation. It entered service with the French and German Navy in 1965, the Netherlands Navy in 1971 and the Italian Navy in 1972. Various modernized versions are being considered with improvements both in performance and operational capabilities.

2.8 Franco-German Trainer Alpha Jet

The most recent AMD/BA production, designed and developed in cooperation with Dornier, a primary trainer which will also be used for ground support missions, benefited from the advanced studies outlined thereafter. This aircraft, powered by two 1 350 kg thrust Larzac 04 SNECMA-Turbomeca turbofan engines and with a weight of 4 500 to 7 000 kg according to configuration, is expected to be built in large quantity in French-German cooperation and eventually with other countries. Prototype 01 made its first flight on 26 October 1973 at Istres and prototype 02 on 9 January 1974 at Oberpfaffenhoffen and as of February 1974 the two models had accumulated 60 hours of flight between them and had been operated throughout their whole flight envelope. After this first phase which proved satisfactory both from the standpoint of airframe and propulsion the spinning tests will be undertaken as well as testing with external stores. (Figures 37 to 39).

3. AMD/BA COMMERCIAL PRODUCTION

3.1 Falcon 20 and Falcon 30/40

In the early sixties when the future of the manned combat aircraft was reconsidered, a number of aircraft manufacturers turned to the commercial market. To break into this difficult area, the company did not depart from its continuity policy and sought the area where its experience with military aircraft might be most profitable whereas technical and financial risks would be minimized. Thus was undertaken the development of the Mystère 20 executive aircraft which was to become later on the Falcon 20 or Fan Jet Falcon (1962).

One of the objectives of the design was the maximum use of components common with existing airplanes (already commonality) as well as the implementation of the same technology wherever possible. It was thus natural to resort to high cruising speeds following the example of prevalent fighters. Along the same lines experience acquired with power-controls as well as the advantages they provide to flying qualities and rapidity of development determined the decision of adopting them rather than manual controls. It was thus possible to report savings realized through the use of common elements on specific items of commercial aircraft, such as fuselage arrangement, equipment, engine installation etc. . . After a first flight in 1963, the testing of the prototype was promptly discontinued to substitute the aft-fanned GE CF 700 turbofan engines to the P and W powerplants in order to take into account Pan American views on such a version. The first flight test results of this new version were sufficiently promising to receive an order from Pan American (160 airplanes). French and US Certifications were obtained in 1965 and since then 300 models have been delivered, the total number of orders and options to date amounts roughly to 400. In 1970 the basic certification was completed with noise certification in compliance with Part 36, the Falcon was thus the first airplane in the world to have received this qualification.

The commercial success of the Falcon 20 encouraged the Company to proceed further in the civil aircraft field and three programs designed for three different outlets were almost simultaneously initiated:

- the Falcon 10, high speed executive aircraft, second generation,
- the Mercure, high-capacity, short-medium range, transport aircraft,
- the Falcon 30/40, third level transport A/C.

These three models together with the Falcon 20 are shown on the picture taken at Le Bourget during the 1973 Airshow (Figure 40).

It is not recommended for an AGARD paper to deal too extensively with the special features of these aircraft. Nevertheless, for the reasons outlined below it would have been illogical to omit them completely:

- as mentioned above they have benefited from the experience and technology applied to military aircraft,
- in the AMD/BA company the same specialized Departments conduct studies on all airplanes, civil or military (on a collaborative basis with the teams affected to each model),
- in some programs "commonality" may be pushed even further, to the advantage of both types of aircraft,

Therefore we will only touch briefly on some general points.

Chronologically the last model is the Falcon 30/40 directly derived from the Falcon 20. The outer wing has been retained while the center wing and the fuselage diameter were increased and the tail adjusted accordingly. Propulsion provided by the high by-pass ratio Lycoming ALF 502 turbofan engines is of particular interest both from the standpoint of consumption and noise. With relatively high cruising speeds the operational cost of this aircraft compares favourably with potential competitors.

3.2 Falcon 10 and Mercure in the Light of New Aerodynamic Design Methods

In parallel to the design of the Mirage G, these two aircraft benefited from the computer programs prepared for modern wing design, wing-fuselage interference, high lift devices, air intakes, etc. . .

A whole Department has been devoting its efforts to this problem for many years. In order to illustrate the efficiency of these methods and show the advances that they have allowed, we will outline some examples hereafter.

Figure 41 shows the evolution of C_D vs Mach number for the Falcon 20 and Mercure. The $\Delta M = 0.30$ increase in the critical Mach number is even more remarkable if we consider that the sweepback of the latter aircraft is only 25° as against 30° for the Falcon 20 and also that the thickness at wing root is 12.5% as against 10%. Moreover these differences made it possible to significantly reduce wing weight, to considerably increase fuel capacity and simultaneously improve the efficiency of high lift devices. These devices were designed using multi-element airfoil computer programs taking into account viscosity and certain non-linearities. Thus on the Mercure a very satisfactory L/D could be obtained at take-off with $C_{L_{max}}$ higher than 2, which is of prime importance for a twin engine aircraft, and at landing a maximum lift coefficient close to 3. Through another computer program it is possible to calculate pressure distribution on the whole aircraft taking all interactions into account. Applications of these computer programs are given on Figures 42 to 50. The results thus obtained find numerous applications:

- investigation of pressure distribution allows detection of possible separation areas; through local modifications these areas may be reduced or eliminated,
- aerodynamic load calculations may be effected with a fairly great accuracy in the preliminary design phase, which will improve structural weight assessment,
- aeroelastic effect can be included in the computation, structural rigidity may be evaluated and monitored,
- all aerodynamic static and dynamic derivatives necessary to the study of flying qualities or simulator work may be determined and these studies undertaken from this early stage.

To illustrate the efficiency of these methods by means of a concrete example, it is possible to compare the Mercure maximum cruising Mach number with that of two transport aircraft of the same category equipped with the same engines (JT 8 D-15). Whereas the Mercure dimensions, designed to carry 15 to 25% more passengers according to the case, expressed in wing area or wetted surface are proportionately larger, its maximum cruising Mach number is still higher by 0.01 to 0.03 than the Mach number of these aircraft which besides have the same wing sweepback.

The large scale use of the computer is now being extended to other disciplines. The integration of the corresponding programs to those of aerodynamics is becoming the basic tool of preliminary design. This expanded role of computers will be briefly outlined in the following chapter.

4. PRESENT AND FUTURE PRELIMINARY DESIGN OF MILITARY AIRCRAFT

4.1 Operational Requirements

After having recalled the methods used in the past as well as their recent evolution, we will now examine the direction in which they are likely to evolve while computers are being developed.

We will limit our discussion to combat aircraft but after an adequate adaptation the procedure can be applied to military as well as to commercial aircraft.

At the initial stage considered the A/C manufacturer will be provided with operational requirements which although not entirely finalized nevertheless specify a number of basic requirements derived from preliminary studies. These requirements cover:

- the various military loads to be carried,
- mission profiles (primary and secondary),
- the environment in which the aircraft will be operated (friendly or enemy territory),
- the weapon-system (range, precision, amount of automatization, etc. . .).

To these requirements should be added cost and time constraints and frequently conditions relating to performance, manoeuvrability, structure, operational limitations, maintainability etc. . . In some cases when the powerplant is not specified in the program it will be imposed through the limited number of engines available which significantly reduces acceptable options.

It often happens that it is not possible to find a solution satisfying the various requirements; in such cases the program shall be reviewed so as to find an acceptable compromise.

Whatever the problem data might be the first task will consist in making a rough estimate of the main dimensions, weights and performace of the various formulae considered.

Once this has been completed successive iterations will improve result accuracy through a more and more detailed investigation of problems.

4.2 Preliminary Design of the Various Configurations ("0" Approximation)

After selection of the configuration to be studied under the program terms, a rough evaluation is carried out in accordance with the diagram shown on Figure 51.

The various items in the diagram will be determined by analogy with existing aircraft or through other simple means. At this stage the structure and flying qualities boxes play no important part. Weight assessment will be performed through simple statistical considerations. For example for a combat aircraft (air superiority or tactical support) the following percentages may be adopted for the weight schedule (% of takeoff weight).

Structure	35 to 38%	
Powerplant	15 to 18%	(depending on engine technology and T/W ratio desired)
Equipment	12 to 14%	
Equipped empty weight	62 to 70%	

A useful load will then amount of 38% to 30%. For the basic mission fuel percentage is practically imposed. The percentage of known military load for this mission will be obtained by calculating the difference between the useful load and fuel percentage. Whence takeoff weight and weight schedule (configuration corresponding to the complete flight envelope).

This example is only given to show that for this type of aircraft the military load is bound to represent a small portion of the takeoff weight and that any error in the weight schedule is changing considerably the military load or the radius of action. It also shows that any increase in the military load entails, for the same performance, an increase in takeoff weight which is 3 to 5 times greater than the load increment.

Thereafter the engine thrust will be determined through technological factors, in particular the T/W ratio and the weight allotted to propulsion.

The fuselage will then be designed with engine installed and full equipment. The wing geometric data will still have to be determined. Various planforms will be considered and regarding wing area the value adopted will be based on wingloading or a set of data will be used. Within this phase tail definition will be based upon statistical considerations.

Now that the aircraft geometry has been defined simplified procedures will be used for evaluating its drag and lift characteristics. Assessment of the first line of the block diagram being completed, we can proceed with calculation of performance and manoeuvrability data (no cost evaluation at this point) and with calculation of the various sensitivity coefficients.

Usually performance and manoeuvrability data set forth in the program are not obtained. The sensitivities shown on the block diagram are then used to correct the main parameters in order to get within reasonable reach of the requirements.

It should be emphasized that the approximation just mentioned is only necessary when the type of aircraft to be designed is significantly different from the concepts already experienced. In the opposite case this first approximation may already be fairly accurate and the number of iterations to be effected will then be significantly reduced.

4.3 Iterations and Finalization of the Preliminary Design

From the approximation defined in the previous paragraph we can now re-estimate the items in the block diagram shown on Figure 51.

Now that the geometry of the various versions proposed has been defined, some parameters remaining variable (thickness ratio, wing area etc. . .) lift and drag characteristics will be evaluated using more or less sophisticated procedures. These procedures are based upon theoretical and experimental results already available or compiled for these particular preliminary projects. (Refer to Appendix 1 for instance).

Propulsion performance data will generally be available for the various types of engines considered.

Simultaneously with calculations fuselage and wing layout studies are undertaken in order to specify shapes, dimensions, and volumes for accommodating fuel, armament, landing gear etc. . .

The new weight breakdown will be established:

- through proven formulae for structure (References 2, 4, 5),
- using data furnished by the engine manufacturer and the powerplant detailed study,
- through the most accurate specification of the equipment list and weapon system (Appendix 2).

Internal iteration loops will allow to take into account the flight envelope and characteristic weights verification will be performed through detailed drawings to cross check the main structural components weights.

These evaluations will also serve to consider weight saving that might be derived from technical and technological advances, the use of new materials, etc. . .

The problems related to flying qualities will be approached simultaneously with the investigation of configurations and their performance data.

The following points will be examined:

- the various tail configurations and size selection in view of obtaining first rate flying qualities in normal mode and acceptable qualities in failure modes,
- the effect of external stores throughout the flight envelope and the operational centre of gravity range,
- different solutions for the development of flight controls, artificial feel and artificial stabilization.

Aircraft behaviour:

- in rapid manoeuvres,
- in turbulence,
- upon load jettisoning,
- at high angles of attack,
- possibly in spin.

As previously outlined the study may be partly conducted through theoretical methods. The remainder must be investigated using models if not entirely compliant at this preliminary stage, at least representative.

The results obtained may guide selection between several fairly equivalent configurations. Thus the Franco-German Alpha Jet trainer configuration was mainly adopted because of its satisfactory behaviour in spinning conditions as evidenced by comparative tests conducted during the preliminary project.

It should be added that these considerations are applicable to conventional flight controls or possibly to the present intermediate type situation. When the fly-by-wire controls will be developed and the CCV (Control Configured Vehicles) become operational they will have to be adapted to the new situation.

Let us now return to our main objective i.e. finalizing of the preliminary project. With the basic data now improved we can undertake realistic performance and manoeuvrability calculations on all configurations considered.

Among the large number of solutions obtained resulting from the various combinations of variable parameters, those which are closest to the requirements specified by the program should be retained.

Through the use of the sensitivity factors derived from the previous results it will then be possible to readjust certain parameters so that the specifications are best satisfied within the limits of existing technology. Additional iterations will be made in case the accuracy obtained would be deemed inadequate.

The final selection among the different solutions developed will be based not only on the previously mentioned flying qualities considerations but also on complex cost/efficiency, development potential, etc. . .

Thus we have emphasized the ever increasing part played by computers in preliminary project development. Without their extensive use it would not have been possible to tackle simultaneously the necessary investigations. Owing to the developments that can already be expected to date, considerable improvements will be obtained from the standpoint of precision and integration of the various disciplines.

The analytical methods which have just been briefly recalled are consistent with a dynamic prototype development policy. Their judicious association should yield a fairly improved production.

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NOTE: English versions of some of the figures referred to in this paper follow. The remainder will be found on Pages 2-15 to 2-42.

APPENDIX 1

LIFT AND DRAG EVALUATION AT THE PRELIMINARY DESIGN STAGE

1. Various computer programs are now available to establish the drag breakdown for a given aircraft configuration. According to the sophistication accepted computer time may vary between seconds and hours.

   All these programs are even applicable to preliminary designs, the choice depending principally on the time allowed to establish the necessary computer inputs.

2. However, in spite of precautions taken and program sophistication, there subsists an amount of uncertainty on the drag results at this stage, mainly due to the lack of complete definition of the aircraft.

   To illustrate by examples, it is noted first that the fuselage forebody of a supersonic fighter, i.e. from the nose to the air-intake, may need many months to be defined. This is due to the fact that before obtaining a shape with a satisfactory velocity distribution at the inlet in the whole Mach number and angle of attack range many iterations are necessary. A simple proportionality rule with a previous airplane is not generally sufficient. A new airplane has to eventually accept a different radar diameter, wider lateral cockpit pedestals etc. requiring basic changes and according to our experience this may give significant drag variations.

   The afterbody design and matching with the engine nozzles is not fully defined at the initial stage. It can be strongly influenced by the position of the horizontal tail and by the dimensions of the tail attachment frames. Neither is the position of the horizontal tail finalized at this stage and three-dimensional theoretical calculations or wind tunnel tests frequently modify the initial choice.

   The operation of the various air intakes used for engine and equipment cooling is affected by their position on the airplane which is currently changed when the centre of gravity and the aircraft structure are better known.

   It can also be said that the inlet additive drag depends on the final compromise selected for the inlet adaptation throughout its operating range.

   Production cost reduction trade-offs will be realized when the aircraft is more accurately defined. For example a local shape modification may be decided so as to avoid machining some fuselage frame on a 5-axis numerical control machine; however this change can be prejudicial to the curvature continuity favoured by aerodynamicists.

3. Finally a large number of minor details may be instrumental in the drag breakdown, i.e. shape of fairings, slat and door seals, panel junction tolerances etc. Actually all these development details may be estimated, based upon experience with previous aircraft. Our company having developed a number of airplanes which are quite different one from another as mentioned previously this makes our task easier. Nevertheless, experience has shown that it may be advisable that the engineer responsible use correction coefficients taking into account the particularities of the team charged with aircraft manufacturing. It will also be useful that this engineer be aware of the probable trends in quality/cost and quality/timescale trade-offs when guaranteed performance data have to be furnished under a given contract.

4. There is a long list of parameters decreasing the accuracy of the drag breakdown estimate at the preliminary design stage, tending to justify the use of rough methods. However this procedure would not be advisable for the following reasons:

   - firstly it is important to know at an early stage the critical areas of the aircraft, for instance areas in which separations are liable to develop under certain Mach, angle of attack or sideslip conditions,

   - secondly it is important to establish partial derivatives with the maximum accuracy in order to reach the best compromise for the aerodynamic design of the aircraft (the essential parameters are the wing area, body and air intake dimensions etc. . .).

5. Regarding lift determination the situation is roughly similar.

   First, a collection of theoretical tools and computer programs are available. As they take into account viscous flow, checks can be made by carrying out computations at the flight and wind tunnel Reynolds numbers and comparing the results with experimental data obtained on models of existing airplanes. Experience has shown that these tools, applied in the difficult area of high-lift devices, are able to define the shape of slats and flaps as well as to give a good estimate of their performance. Second, many details are also to be considered in this case. An inaccurate junction between two slats or flaps, a wake produced by a flap rail are often liable to spoil the longitudinal stability or the lift expected from a good flap. The performance of a high-lift device will also suffer from an amount of uncertainty, but the general conclusions stated for drag evaluation remain valid for estimating the lift coefficient in preliminary design.

APPENDIX 2

PART PLAYED BY AMD/BA IN THE DESIGN AND DEVELOPMENT OF WEAPON SYSTEMS

1. INTRODUCTION

It is no longer possible to dissociate the value of a new type of aircraft from the value of its avionics and armament. The efficiency of a weapon system is largely dependent upon investigations and testing conducted in order to obtain the best possible integration between armament and avionics as well as between this equipment and the airframe.

AMD/BA has been aware that this problem was vital as soon as the first integrated electronic systems were used on military airplanes. A "Weapon System Department" has therefore been set up to take care of the design, definition and integration tests of aircraft avionics and armament.

2. PRELIMINARY PROJECT PHASE

For each new military aircraft this Department undertakes simultaneously the study of the characteristics of the armament and avionics selected in relation to the objectives specified by the user.

The following system components which have a direct impact on the airframe performance are first considered:

- equipment weight,
- definition of external stores, with a special emphasis on missiles,
- radar antenna diameter,
- external projections: antennae – infra-red detectors – Laser – etc. . .,
- electrical load requirements and air-conditioning.

A parametric mathematical model is established for the main components likely to affect the overall efficiency of the weapon system in view of comparing all solutions considered.

As an example, for an interceptor preliminary project, solutions are compared while considering:

- the dimensions and drag of air-to-air missiles and the various aircraft installation characteristics,
- radar antenna size and radar characteristics,
- interceptor detection, navigation and guidance,
- ground detection and guidance.

Complete mathematical modelling of the weapon system makes it possible to direct the selection of solutions as well as the specification of performance requirements for electronic equipment.

The radar and the missile manufacturers take an active part in these studies.

At this stage a preliminary project for the electronic system is made by the Weapons System Department.

This Department which is permanently engaged in the design and development of new systems may use its experience and knowledge in the advancement of electronic technologies to make a synthesis of the projects realized by various equipment manufacturers and propose complete integrated systems meeting the objectives specified through a satisfactory compromise between weight, cost, performance, reliability, maintainability, development potential etc. . .

These proposals are submitted to the Official Services which operate the final selection.

3. DETAILED DEFINITION PHASE

In this phase the Weapons System Department is normally responsible for:

- equipment installation and integration specifications,
- overall performance studies,
- investigation of controls and visualizations – cockpit equipment,
- system mathematical modelling – simulation,

– maintenance and reliability studies,
– formulation of technical specifications for the equipment making up the system,
– investigation of safety problems and chance of success for the mission.

This work is carried out in close cooperation between all parties concerned, the aircraft manufacturer being charged mainly with the coordination and synthesis of the various studies.

4. INTEGRATION TESTS

After the various equipment items have been individually tested in laboratory or on utility aircraft, the aircraft manufacturer usually realizes their integration:

– on ground global test benches and in a radioelectrical mock-up of the aircraft on which noises and interferences are more particularly studied, on one hand,

– on the utility aircraft and on the prototypes of the aircraft itself, on the other.

These tests, performed in cooperation with the various interested parties, are conducted to finalize the development and to check the system global performance.

The necessary changes are specified to be applied to production articles.

5. INTEGRATION OF PRODUCTION EQUIPMENT

The A/C manufacturer conducts on the final assembly production line integration tests of the equipment installed on the aircraft; this testing is conducted on a global test bench similar to that used for prototype development. The overall system is then accepted for installation on the operational airplane by the pilots of the Official Services.

The A/C and equipment manufacturers provide the Air Force with technical assistance during the first months of operation for the new airplanes. Deficiencies observed in operation are used to specify within a short time the changes required to improve system operation, utilization, maintenance and reliability.

WING REFERENCE AREA	375 sq. ft
SWEEPBACK (LEADING EDGE)	60°
ASPECT RATIO	1.94
THICKNESS RATIO	4 to 3.5 %
POWERPLANT	SNECMA ATAR 9 C
- Military thrust	9,700 lbs
- Max. thrust	13,700 lbs
BASIC OPERATIONAL WEIGHT	15,300 lbs
TAKE-OFF WEIGHT CLEAN A/C	20,500 lbs
MAX. TAKE-OFF WEIGHT	30,200 lbs
MAX. EXTERNAL FUEL	1 240 U.S. Gal.

Fig.4 Mirage 5 – leading characteristics

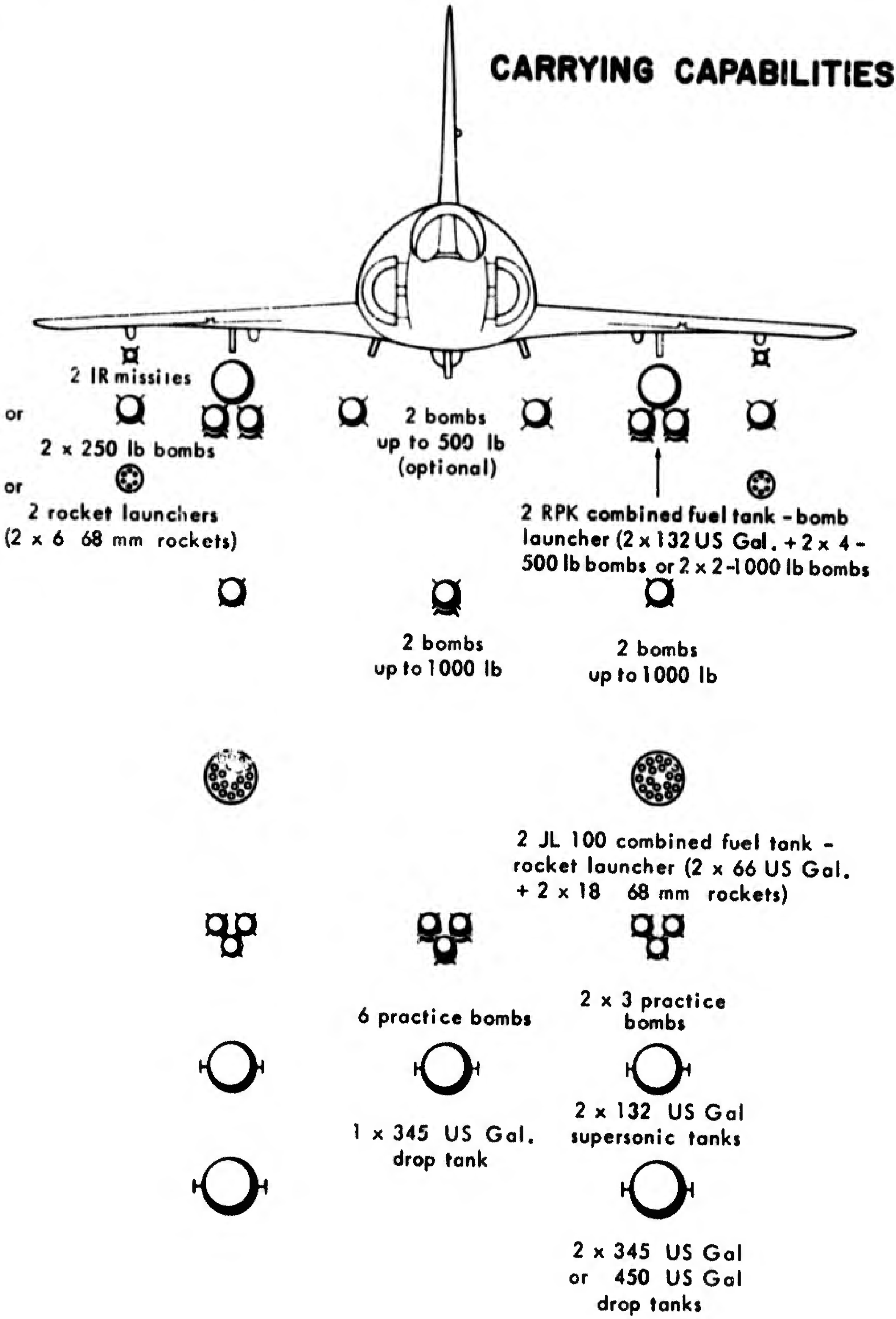


Fig.6 Pylones and external stores

FULLY PROVEN..

The first preproduction aircraft made its maiden flight on March 1969. Since that date, the three preproduction aircraft have successfully flown more than 1500 flight hours. A first operational squadron is formed in late 73.

The aircraft is currently mass produced to meet French and overseas Air Forces orders in two versions :

- **MIRAGE F 1 C**
  Air superiority : Air defence as primary mission and ground attack as secondary mission
- **MIRAGE F 1 A :**
  Ground support mission and Air Superiority.

WITH SIGNIFICANT PROGRESS..

While retaining and improving the ruggedness, easiness of maintenance and combat superiority of the MIRAGE III, the MIRAGE F 1 C demonstrates a general improvement in operational capability :

- **MACH 2.2** combat capability instead of Mach 2
- **3 TIMES the PURSUIT TIME** in high supersonic flight
- **3 TIMES the PATROL TIME**
- **TWICE the RADIUS** in Lo-Lo attack mission
- **23 % LESS STRIP LENGTH** at max. take-off weight
- **20 % APPROACH SPEED REDUCTION**
- **Up to 80 % MORE MANOEUVRABILITY**

BY MEANS OF ADVANCED TECHNOLOGY

- New structural technologies allow weight savings and increased fuel capacity (+ 43 %)
- Elaborate high lift system
- [illegible] % more powerful engine with nearly unchanged weight and volume
- More advanced fire control system with twice the range and new capabilities
- Improved weaponry

Fig.14 Mirage F1, a new generation fighter

OVERALL DIMENSIONS		
- Span	27 ft 8 in	8,42 m
- Height	14 ft 8 in	4,49 m
- Length	49 ft 7 in	15,23 m
WING REFERENCE AREA	269 sq. ft	25 m 2
SWEEPBACK (LEADING EDGE)	47° 34'	
ASPECT RATIO	2.8	
THICKNESS RATIO	4.5 to 3.5 %	
POWERPLANT	SNECMA ATAR 9 K 50	
- Military thrust	11100 lb	5020 kg
- Max. thrust	15850 lb	7200 kg
BASIC OPERATIONAL WEIGHT (1) (with 2 x 30 mm guns)	17110 lb	7760 kg
TAKE-OFF WEIGHT CLEAN AIRCRAFT (1) (full fuel + 250 rounds)	24930 lb	11325 kg
MAX. DESIGN TAKE-OFF WEIGHT	33520 lb	15200 kg

Fig.15 Leading characteristics

COMMUNICATION

V/UHF TRT ERA 7000

UHF TRT ERA 7200

IDENTIFICATION

IFF LMT NRAI 4A

NAVIGATION

TACAN LMT AN ARN 52

VOR ILS-MARKER

SOCRAT 3300 5600

FLIGHT MONITORING

AUTOPILOT SFENA 505

GYRO-PLATFORM SFIM 153 7

AIR DATA COMPUTER

CROUZET 63

CRASH RECORDER SFIM

FIRE CONTROL SYSTEM

FORWARD LOOKING RADAR CYRANO IV

HEAD UP DISPLAY THOMSON CSF 196

MATRA 530 A A MISSILES CONTROL UNIT

SPACE PROVISION FOR DATA LINK SYSTEM

GUNS, BOMBS, ROCKETS, NAPALM, A A MISSILES

complete firing circuit

Fig.16 Basic avionics

IDENTIFICATION

IFF (different types)

NAVIGATION

VOR ILS-MARKER (different types)

INERTIAL UNIT with associated NAV. ATTACK COMPUTER

RADIO ALTIMETER

OPTIONAL RADAR MODES :

- Terrain avoidance
- Contour mapping
- Blind penetration

FIRE CONTROL SYSTEM

MOBILE TARGET INDICATOR THOMSON-CSF

CATHODE RAY TUBE H.U.D.

AIR GROUND RADAR RANGE FINDING

OTHER EQUIPMENTS

ANTI RADAR TAIL WARNING SYSTEM THOMSON-CSF

SELF DEFENCE E.C.M. POD CAPABILITY

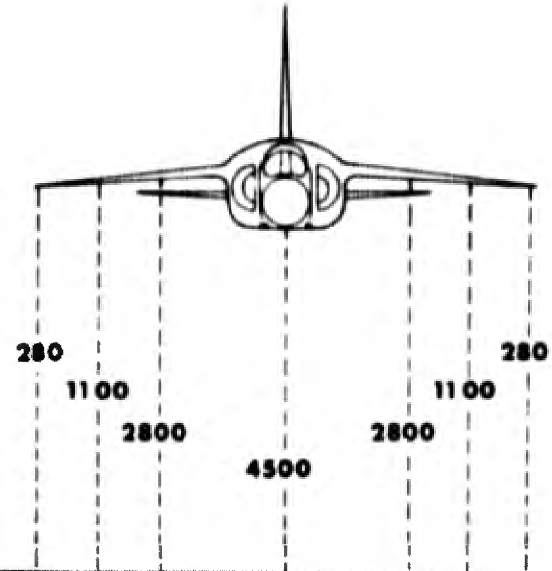
ESCORT E.C.M. POD CAPABILITY

FATIGUEMETER EMD

Fig.17 Optional avionics

2 x 30 mm Guns

LOAD CAPABILITY IN LB



		(1)		280	1100	2800	4500	2800	1100	280
AIR TO AIR MISSILES	MATRA 530	b	1				●			
	or MATRA 530 or SUPER 530	b a	2			●		●		
	MAGIC or Sidewinder	b a	2	●						●
AIR TO SURFACE MISSILES	500 kg class	a	1				●			
BOMBS	STRIM 400 kg or 1000 lbs MK 10	a b	8		●	●	● ●	●	●	
	500 lbs MK 82	a	14		●	● ●	● ●	● ●	●	
ROCKET LAUNCHERS	M 155 - 18 rockets (68 mm) or F1 - 36 rockets (68 mm) or LR 100 - 6 rockets (100 mm)	b a a	4		●	●		●	●	
FUEL TANKS	1200 l / 315 USG	b	3			●	●	●		
NAPALM	300 l / 80 USG	a	6		●	●	● ●	●	●	
PODS	Gun pods	a	2			●		●		
	Recce container	a	1				●			
	ECM pods	a	3	●			●			●

(1) b = basic a = additional

Fig.18 Carrying capabilities. Armament and external stores

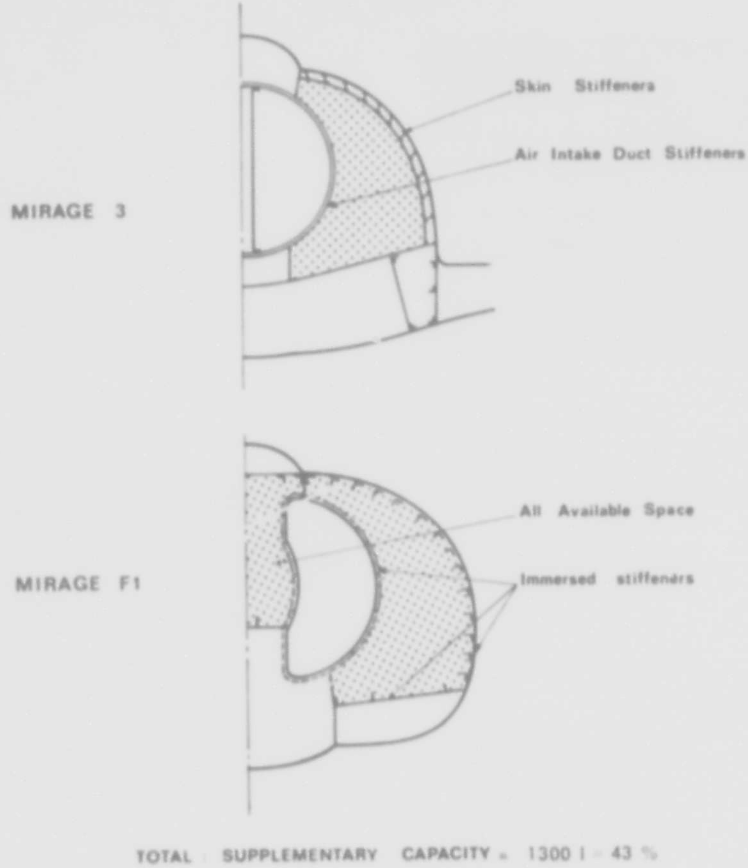


Fig.20 Fuselage integral tanks

DESIGN PHILOSOPHY

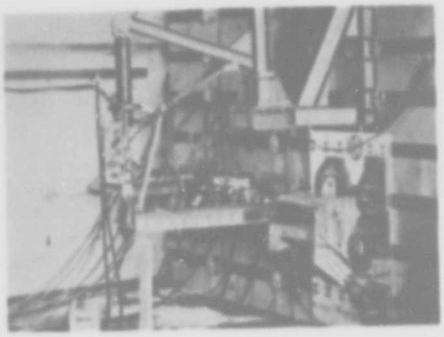
— WITH STABILITY AUGMENTATION ON : EXCELLENT FIRING PLATFORM

— WITHOUT : CONTROL ASSURED THROUGHOUT THE FLIGHT ENVELOPE

- DESIGNED AND PRODUCED BY A. M. D.
- TWIN ELECTRO-HYDRAULIC SERVO-CONTROLS WITH HYDRO-MECHANICAL SAFETY
- WING SCREW JACK GIVING HIGH LEVEL SAFETY AND PERFECT SYNCHRONIZATION
- ALL POSITION HIGH-LIFT DEVICE FULLY DUPLICATED
- AIR BRAKES WITHOUT CHANGE OF TRIM IN THE COMPLETE ENVELOPE

Fig.29 Flight controls

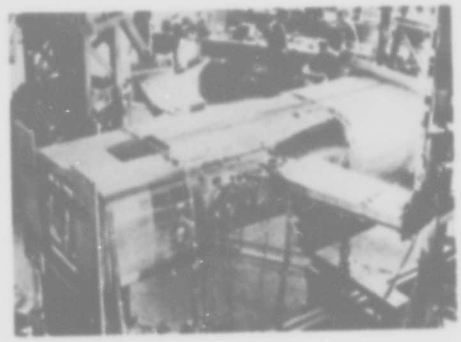
FATIGUE TEST



STATIC TEST

12 g à $\Lambda = 70°$

9 g à $\Lambda = 20°$



ENDURANCE

60 000 manœuvers

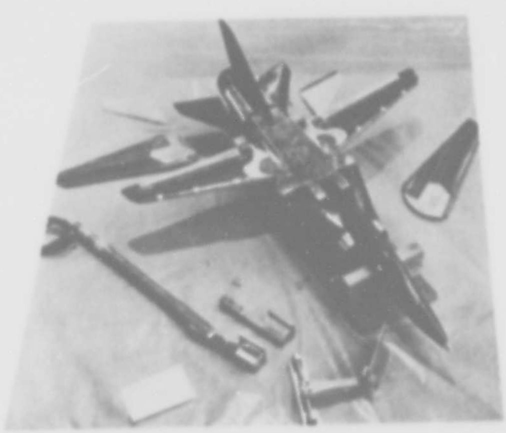
Fig.30 Wing pivot tests

- RESULTS FROM MANY THEORETICAL STUDIES AND WIND TUNNEL TESTS

- CHOICE OF PIVOT POSITION IS A COMPROMISE BETWEEN

  . Good manoeuvrability
  . Low turbulence sensitivity
  . Trimmed drag matched to primary mission
  . Low trim variation during wing movements

- SMALL AND CLEAN GLOVE CONFIGURATION ADAPTED TO HIGH LIFT POSSIBILITY



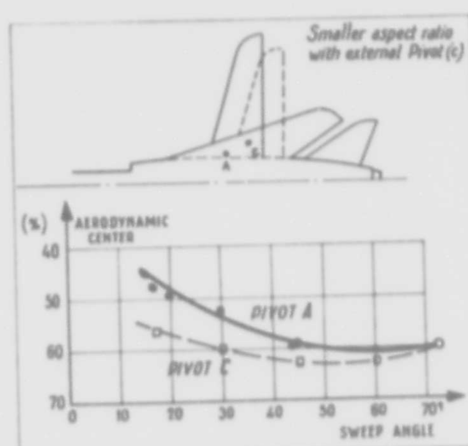


Fig.31 Pivot position

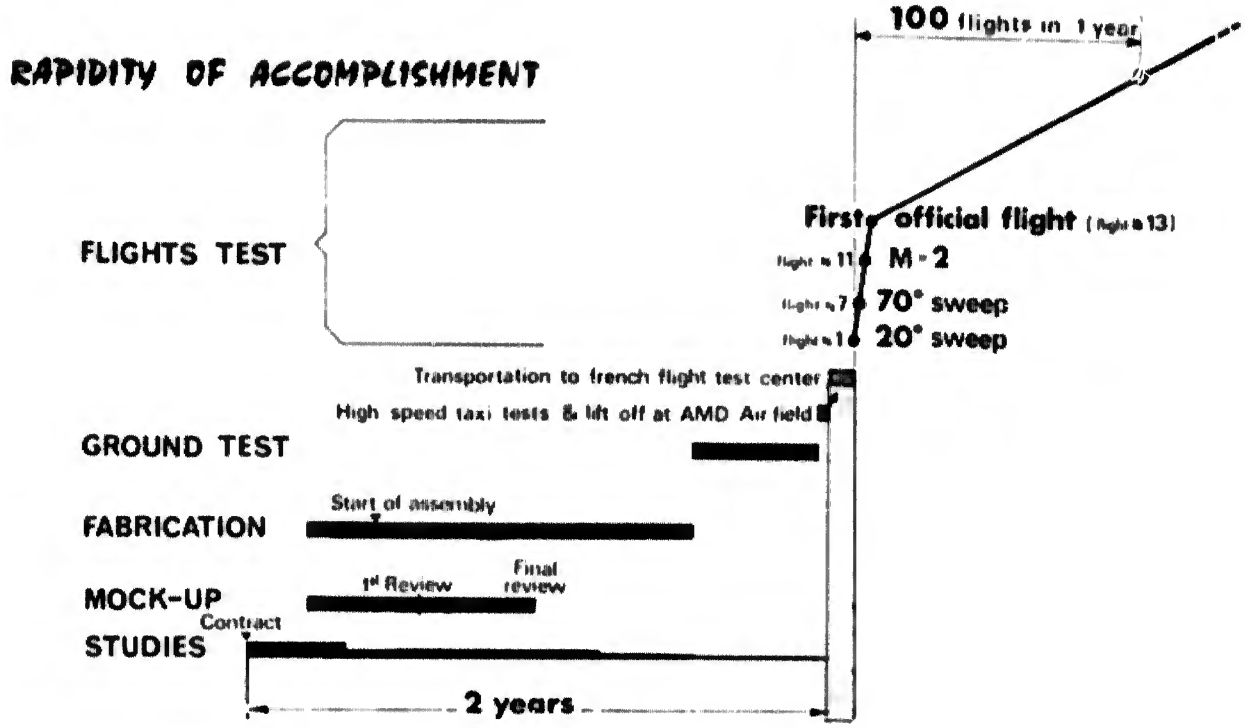
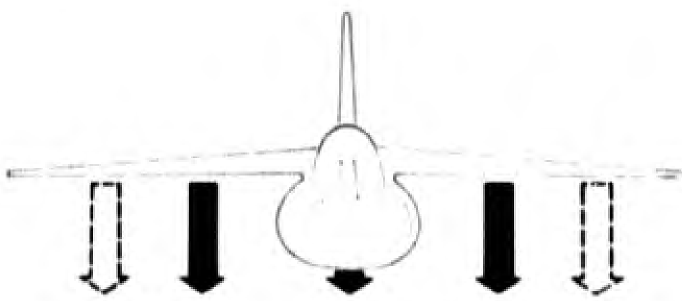


Fig.32 Mirage G milestones

ARMAMENT

TYPICAL CONFIGURATIONS



30 mm guns with 150 rounds	
400 kg bomb clean or retarded	
250 kg bomb clean or retarded	
125 kg and 50 kg bombs	
625 lb cluster bomb	
690 lb and 825 lb fire bombs	
Class 2.75 in rocket launcher 6 - 19 - 36 rockets	
360 lb combined bomb and rocket launcher for training	
2 x 310 l (82 USG) drop tanks	
1 reconnaissance pod	

Fig.37 Alpha jet carrying capabilities

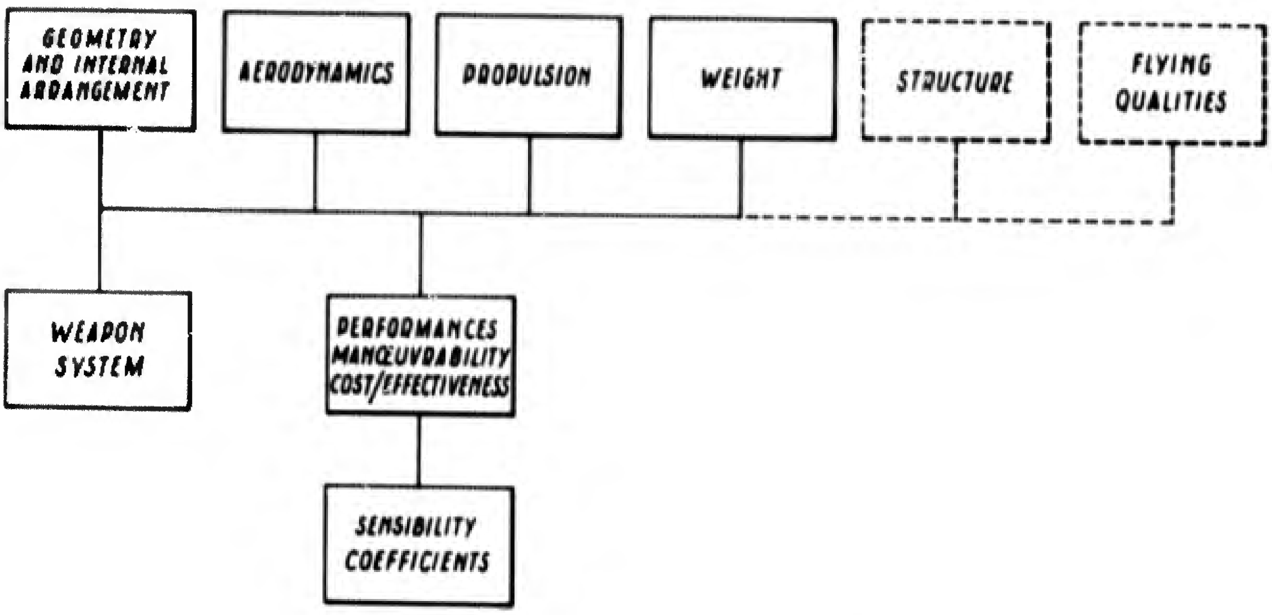


Fig.51 Block-diagram for the preliminary design

PROPULSION/AIRCRAFT DESIGN MATCHING EXPERIENCE

by

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Director of Advanced Systems and Technology
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PRESTON
Lancs

The Wright brothers succeeded by understanding the practical power plant, wind tunnel and other numbers required. Subsequent improvements through to jet propulsion, economic subsonic and supersonic flight have required increasing understanding of the many practical numbers involved. Further improvements (particularly with airframes made from alloys basically similar to those used well over 60 years ago) cannot occur without an understanding at least equivalent to this paper. Such understanding must start from the top.

For quicker reference, the text is cut to match the diagrams. The paper starts from proven first principles as requested, rather than from computer or other overall results whose generality can be questioned.

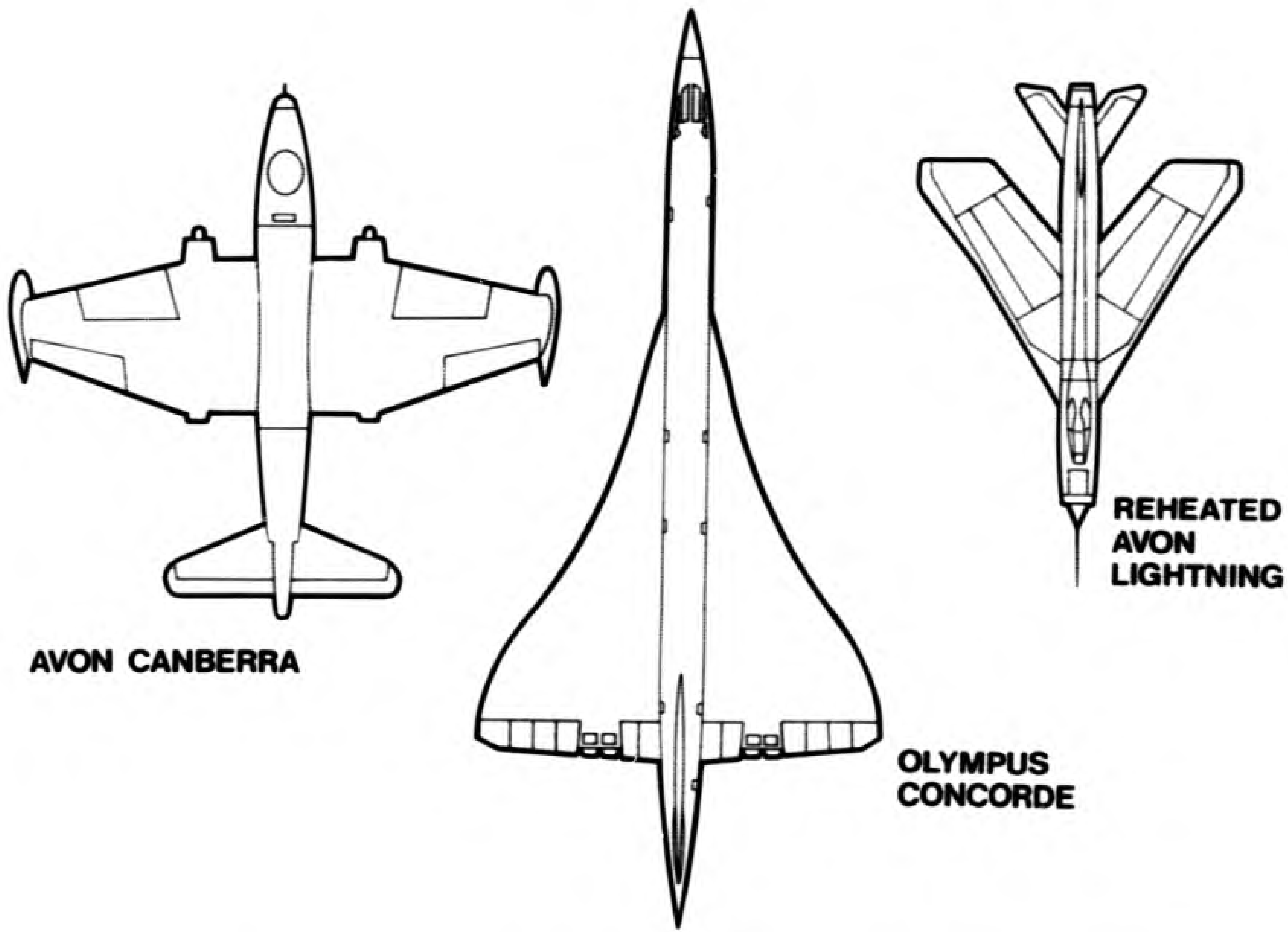


FIG.1. INTRODUCTORY ENGINE/AIRCRAFT MATCHINGS CONSIDERED

PROPULSION/AIRCRAFT DESIGN MATCHING EXPERIENCE

Before becoming too involved in optimisation of sizes and the other parameters, much of which can be helped by massive computer programmes, I was asked to introduce some understanding of the important thrust and fuel required/available problems.

More convincing than too many heavily disguised current performances, or unsubstantiated future claims, are real examples from the spectrum of successfully proven designs (Fig. 1). Security on the military side forces me back to early versions of:-

(i) The Canberra jet reconnaissance/attack aircraft. This example allows illustration of penetration at very high altitudes, or down at the very low altitudes currently required in NATO and elsewhere. (International security inhibits real data for more recent aircraft like Jaguar and MRCA, which will make up much of the front line of the RAF and her main European allies).

(ii) The P1 Lightning fighter with Mach 2 supersonic performance. Reconnaissance/attack versions of this aircraft were developed, which is typical of today's multi-role trend.

Although preliminary design of (i) and (ii) occurred in the mid and late 1940s respectively, more advanced versions of both aircraft are still in service. Their performances approach or even exceed some designs today.

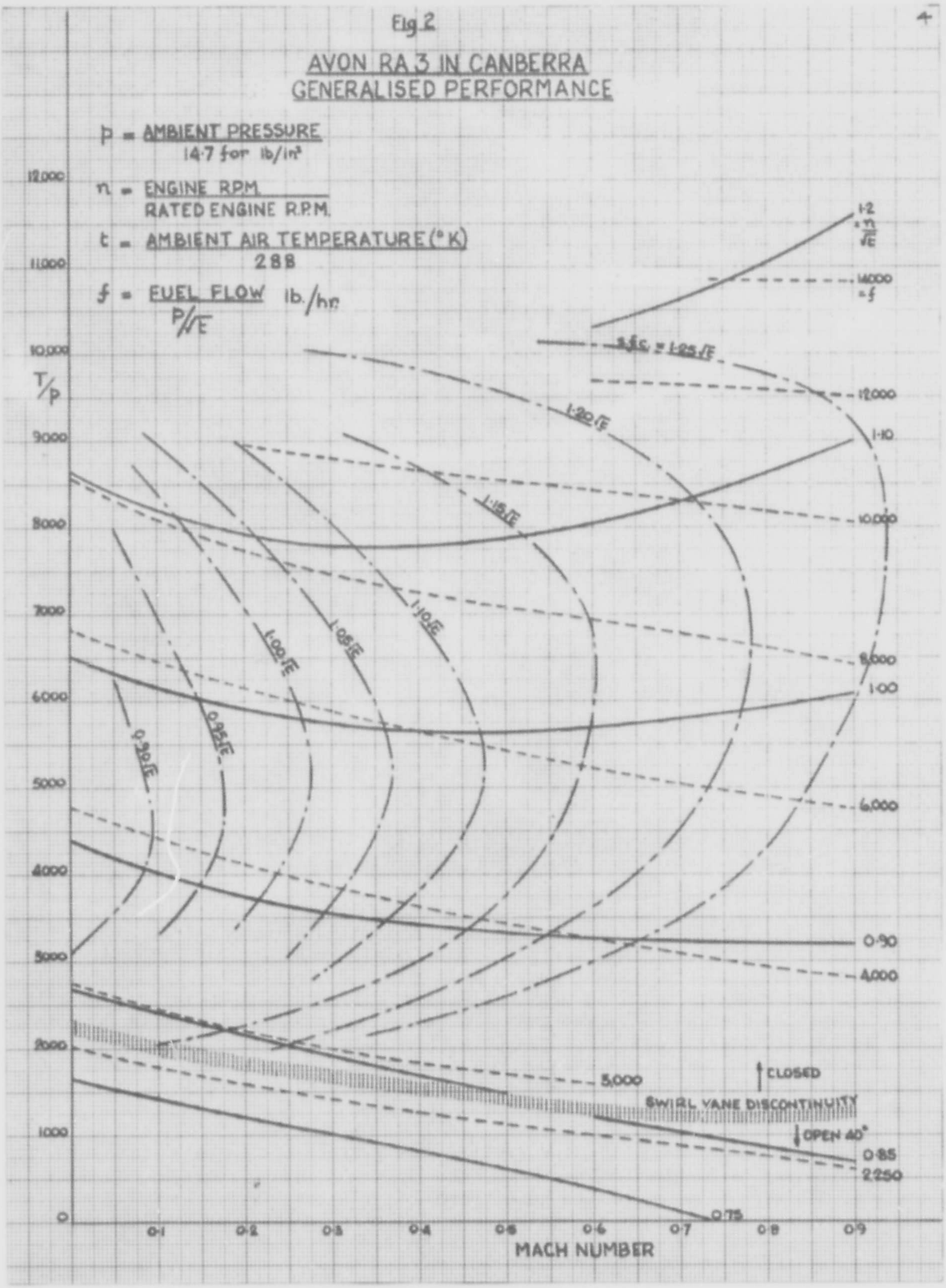
(iii) The Concorde supersonic transport is topical on the civil side, but the military emphasis in the other lectures requires me to discuss it only briefly.

(iv) The subsonic transport matching problems are still surprisingly similar to Canberra, and can be illustrated very simply by reference mainly to turbofan characteristics.

This leads into a general discussion of future transport and military propulsion, including computer methods extending to RPVs and V/STOL. Cross-matching is possible around common engine cores, but the Appendices are required to interpret modern engine computer data.

Methods: Preliminary design from the 1940s takes us back before our widespread use of computers, when there was no alternative to graphical methods for all performance and preliminary design problems. The method used will give simple feel for the complete aircraft problem, without getting too involved at the start in gas turbine theory. This was asked for by the Director to encourage "judgment, innovation and originality". It may help to explain why some of us can make intuitive guesses during discussion, or make back-of-the-envelope assessment of key problems before a computer card is punched; certainly where to look in hundreds of feet of computer print-outs.

One soon gets used to reading the complete performance of an aircraft and its propulsion from a single diagram, like Fig. 11 for the Avon Canberra. In case this is too much at once for the uninitiated, I will construct this first diagram in stages, giving practical illustration of its meaning and uses as we go along.


4
Fig 2
AVON RA 3 IN CANBERRA
GENERALISED PERFORMANCE
p = AMBIENT PRESSURE / 14·7 for lb/in²
n = ENGINE RPM / RATED ENGINE R.P.M.
t = AMBIENT AIR TEMPERATURE (°K) / 288
f = FUEL FLOW / P/√t lb./hr.
T/p
MACH NUMBER
s.f.c. = 1·25√t
1·20√t
1·15√t
1·10√t
1·05√t
1·00√t
0·95√t
0·90√t
1·2 = n/√t
14000 = f
12000
1·10
10,000
8,000
1·00
6,000
0·90
4,000
5,000
CLOSED
SWIRL VANE DISCONTINUITY
OPEN 40°
0·85
2250
0·75

1. AVON CANBERRA PERFORMANCE

FIGURE 2 shows the complete Avon RA3 performance in various ways. The non-dimensional thrust (full lines) varies only with Mach number and non-dimensional rpm. The fixed Avon nozzle area is already substituted for simpler use, and atmospheric pressure is further non-dimensionalised as little p relative to standard sea level. You will rarely find it necessary to get a feel for actual thrusts at altitude, so I will deal with p later in connection with Fig. 5.

t is the atmospheric temperature divided by that for standard sea level. n is the rpm non-dimensionalised relative to maximum rpm - related to the way most pilots' instruments were later calibrated.

The diagram up to $n/\sqrt{t} = 1$ can thus be interpreted as a straight plot of thrust in lb against Mach number for various pilot rpm at standard sea level. Completed, it represents the full performance of the Avon at all altitudes in all climates.

The rpm factor that will always stick in my mind is just over 1.15 in the standard stratosphere above 36,000 feet, falling almost linearly to 1.05 near 14,000 ft. Table 4 overleaf gives more than enough atmospheric data for other purposes.

You will soon find that you can use these curves in the preliminary design phase with less numbers or thought than when you first used a C_D curve against M, plotted for different values of C_L.

As is typical for engines with good intakes, net thrust falls away with M at first due to the ram drag of the intake air. Ram pressures through the engine then build up a fairly flat curve at subsonic speeds, with more marked recovery at high $n/\sqrt{t}$ and M. This will be further discussed towards the end.

One non-typical feature of the Avon was the discontinuous drop in thrust (and jump in sfc) when the swirl vanes opened in front of the compressor - an early form of variable geometry to solve the low $n/\sqrt{t}$ surge problem.

FUEL FLOW: The dotted lines complete the flight performance of the Avon. Operation is very simple from the pilot's viewpoint. He opens the fuel throttles and rpm increases to increase the thrust. This increase is very rapid at high rpm and Mach number. At a given M, increase in thrust becomes pro rata with fuel at $n/\sqrt{t}$ approaching 1, but this is better seen in terms of sfc.

Alternatively sfc is crossplotted as fuel flow divided by net thrust. Most professionals are more used to thinking in terms of specific fuel consumption, but it has little meaning at low rpm. It is seen that sfc reaches a minimum at $n/\sqrt{t}$ approaching 1. This again is typical for many engines. Sfc also increases slowly with forward speed. This means that propulsive efficiency $\left(\frac{M}{sfc}\right)$ increases almost linearly with forward speed up to high M.

Fig 5 GENERALISED CANBERRA DRAG

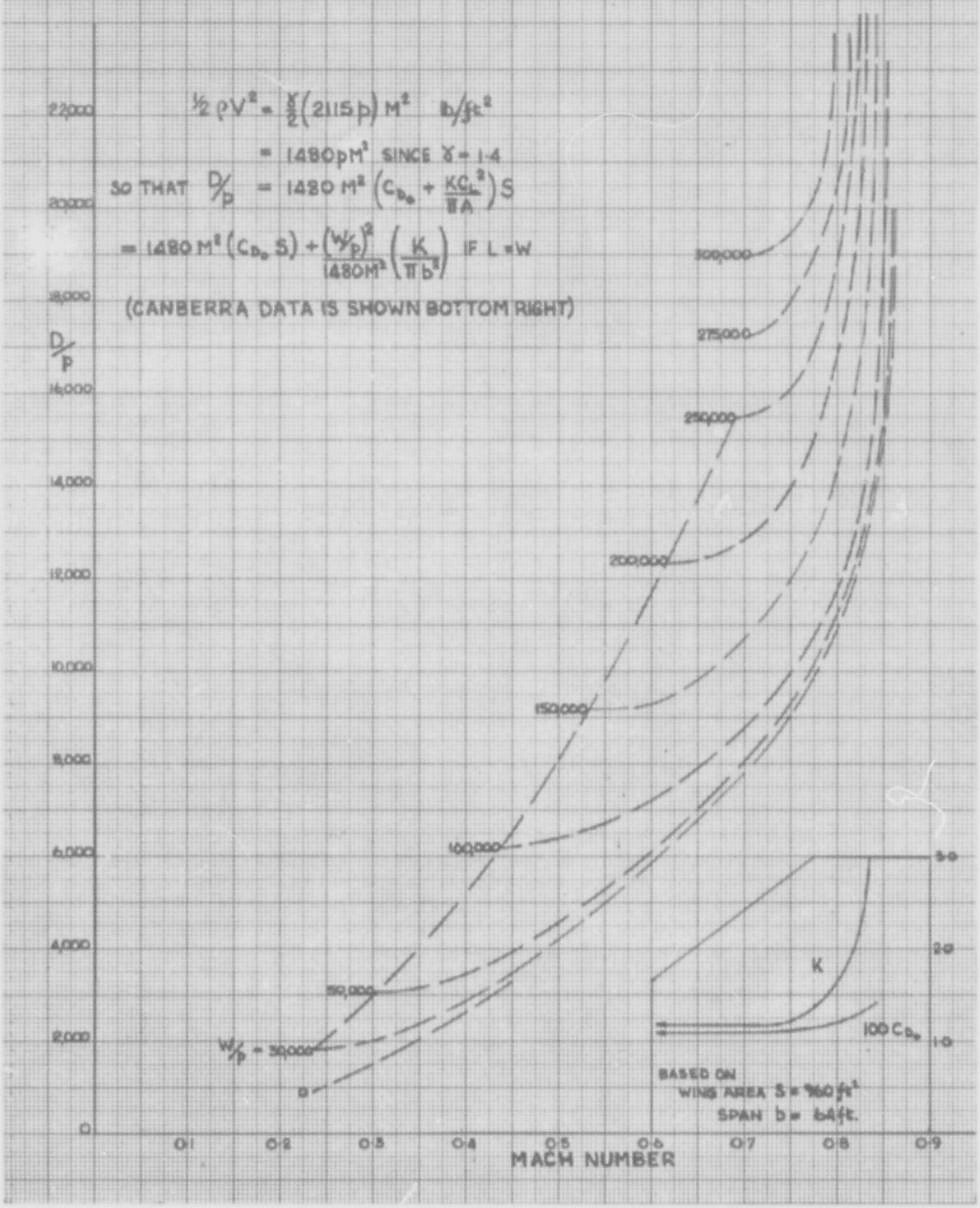


FIGURE 5 converts the standard drag equation into a form suitable for relating to the engine diagram just discussed. Since other people will discuss airframe design and drag problems in detail, my treatment will be the minimum for understanding the main design integration problems.

To get a quicker understanding of most numerical performance, the initial equation assumes that Lift = Weight. (We deal with flight under g, and below minimum drag speed later on Fig. 11.)

The first term defines the curve for $W/p = 0$ at the bottom. This is a simple parabola up to M = .75. Having drawn any other curve, vertical increments vary as $(W/p)^2$ so that all are constructed very quickly.

Since K is constant for Canberra over a wide range of C_L, this other curve can be drawn very quickly to beyond M = .7. Anticipate the result on Fig. 13 that Canberra max L/D is 16 when $C_D = 2\ C_{D_o}$, i.e. on a second parabola, at twice the height of the first. Thus, taking W/p = 100,000 and dividing by 16, marks off D/p = 6250 lb. The drag increment over our first parabola then decreases inversely as M^2 up to nearly M = 0.75. The discussion of Fig. 13A will confirm that the Canberra was a very clean design over this speed range. Its thick wing was not designed to fly much beyond M = .85, and we will be dealing with higher speed designs later.

More important at this stage is to develop a feel for numerical values of W/p. This of course equals the aircraft flying Weight at sea level pressure altitude, and the last column of Fig. 4 gave values of p at other pressure altitudes. p = 0.2234 will always stick in my mind as the value at 36,090 feet, the tropopause in the standard atmosphere; in other words, W/p equals nearly 4½ times the aircraft weight. This factor increases by nearly 5% per 1000 feet in the standard stratosphere, and doubles every 14,400 feet. The latter will also be useful later.

FIGURE 6 merely superimposes relevant parts of the previous diagrams, and should be folded out from page 3A when studying the next few pages. The vertical scale of the engine curves are doubled to summarise the complete performance of the Canberra with its two Avon engines.

The easiest way to get a feel for engine/aircraft matching is to start with all aspects of steady level flight performance. These occur where thrust and drag are equal, of course. To help get used to reading these directly where T and D curves intersect on Fig. 6, you might prefer to discuss the more conventional Figures 7, 8 and 9 first. Each of these is derived directly from Fig. 6.

(Figs 4 and 6 overleaf.
Figs 3 and 12 eliminated.)

Fig. 6
UNFOLD FOR NEXT FEW PAGES

3.8a

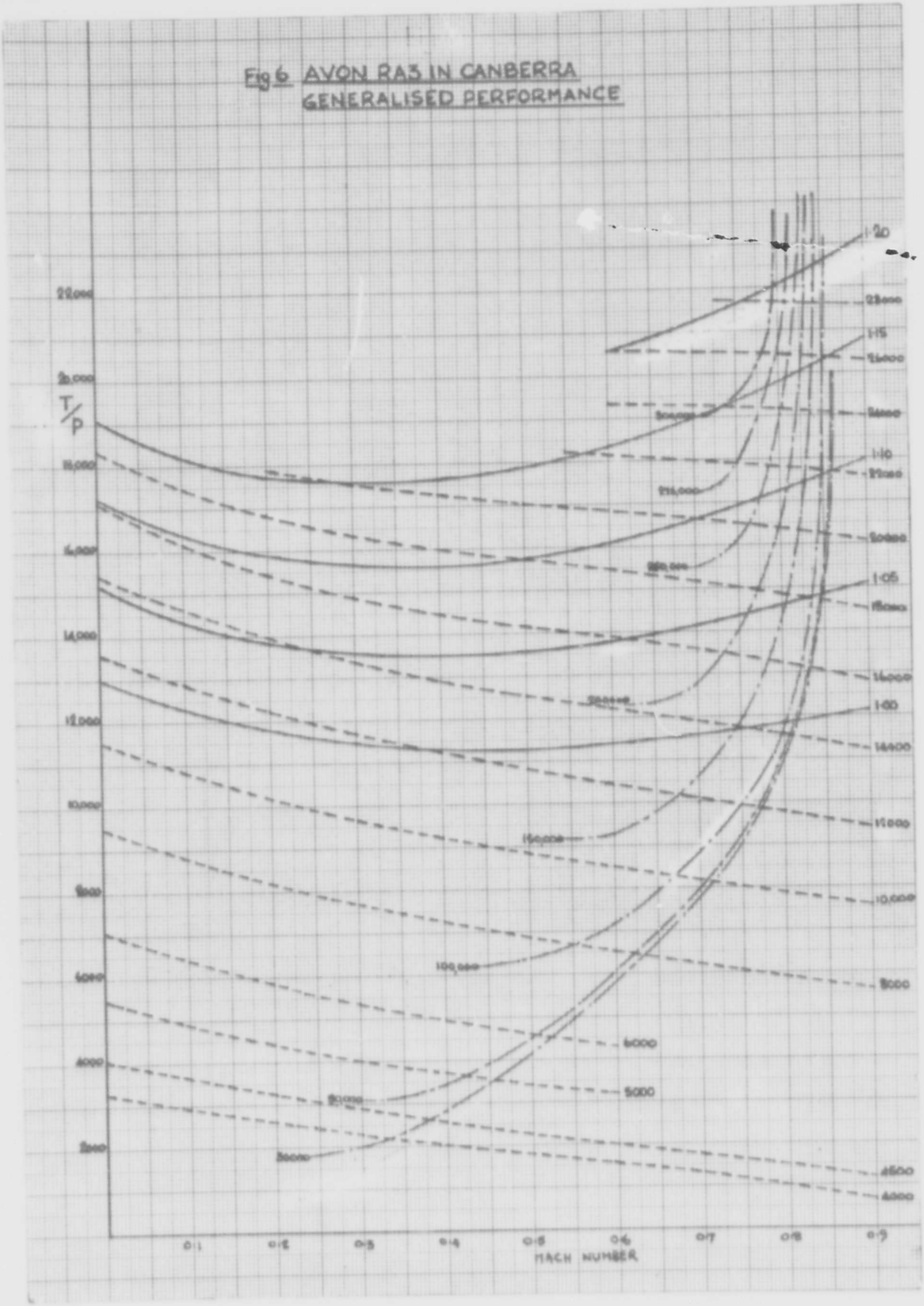


Fig 6 AVON RA3 IN CANBERRA
GENERALISED PERFORMANCE
T/P
22,000
20,000
18,000
16,000
14,000
12,000
10,000
8000
6000
4000
2000
0·1
0·2
0·3
0·4
0·5
0·6
0·7
0·8
0·9
MACH NUMBER
1·20
1·15
1·10
1·05
1·00
12000
10,000
8000
6000
5000
4500
4000
150,000
100,000

FIG. 4

PROPERTIES OF THE STANDARD ATMOSPHERE

h = pressure altitude (note that 36,089 ft. = 11 km)
t = absolute temperature relative to standard h = 0 or 288°K (15°C) or 518°R (60°F)
a = speed of sound = $661.4\sqrt{t}$ kts TAS = $661.4\sqrt{p}$ kts EAS
p = pressure relative to standard h = 0 value of 1013.2 millibars = 2116.2 lb/ft^2 = 14.7 lb/in^2

h ft.	$\frac{1}{\sqrt{t}}$	$\sqrt{t}$	Speed of sound knots TAS	Speed of sound knots EAS	p
0	1.000	1.000	661.5	661.5	1.000
1,000	1.003	.9966	659.2	649.6	.9644
2,000	1.007	.9931	656.9	637.3	.9298
3,000	1.011	.9896	654.6	626.2	.8963
4,000	1.014	.9862	652.3	614.7	.8637
5,000	1.018	.9827	650.0	603.4	.8321
6,000	1.012	.9792	647.7	592.1	.8014
7,000	1.025	.9756	645.4	581.0	.7717
8,000	1.029	.9721	643.0	570.1	.7428
9,000	1.032	.9686	640.7	559.2	.7148
10,000	1.036	.9650	638.3	548.5	.6877
11,000	1.040	.9614	635.9	538.0	.6614
12,000	1.044	.9579	633.6	527.5	.6360
13,000	1.048	.9543	631.2	517.2	.6113
14,000	1.052	.9507	628.8	507.0	.5875
15,000	1.056	.9470	626.4	496.9	.5644
16,000	1.060	.9434	624.0	486.9	.5420
17,000	1.064	.9397	621.6	477.1	.5203
18,000	1.068	.9361	619.1	467.4	.4994
19,000	1.073	.9324	616.7	457.9	.4792
20,000	1.077	.9287	614.3	448.4	.4596
21,000	1.081	.9250	611.8	439.1	.4406
22,000	1.085	.9213	609.4	429.9	.4223
23,000	1.090	.9175	606.9	420.8	.4047
24,000	1.094	.9138	604.4	411.8	.3876
25,000	1.099	.9100	601.9	402.9	.3711
26,000	1.104	.9062	599.4	394.2	.3552
27,000	1.108	.9024	596.9	385.6	.3399
28,000	1.113	.8986	594.4	377.1	.3251
29,000	1.118	.8948	591.8	368.7	.3108
30,000	1.122	.8909	589.3	360.5	.2970
31,000	1.127	.8871	586.7	352.3	.2837
32,000	1.132	.8832	584.2	344.3	.2709
33,000	1.137	.8793	581.6	336.4	.2586
34,000	1.142	.8754	579.0	328.6	.2467
35,000	1.148	.8714	576.4	320.9	.2353
36,000	1.153	.8675	573.8	313.3	.2244
36,089	1.153	.8671	573.6	312.6	.2234

h ft.	$\frac{1}{\sqrt{t}}$	$\sqrt{t}$	Speed of sound knots TAS	Speed of sound knots EAS	p
36,089	1.153	.8671	573.4	312.6	.2234
37,000				305.6	.2138
38,000				298.6	.2038
39,000				291.5	.1942
40,000				284.6	.1851
41,000				277.8	.1764
42,000				271.2	.1681
43,000				264.8	.1602
44,000				258.4	.1527
45,000				252.4	.1456
46,000				246.3	.1387
47,000				240.5	.1322
48,000				234.7	.1260
49,000				229.2	.1201
50,000				223.8	.1145
51,000				218.5	.1091
52,000				212.7	.1040
53,000				208.2	.09909
54,000				203.3	.09444
55,000				198.4	.09005
56,000				193.7	.08578
57,000				189.1	.08176
58,000				184.6	.07792
59,000				180.2	.07426
60,000				175.9	.07078
61,000				171.7	.06746
62,000				167.7	.06429
63,000				163.7	.06127
64,000				159.8	.05840
65,000	1.153	.8671	573.4	156.0	.05566

In real atmospheres the last two columns are unchanged as a function of pressure altitude. Except near the ground, even the other columns vary 5% only on very rare occasions. Perhaps surprisingly this is most frequent as a temperature drop around 60,000 ft. in the tropics. (Mech. Eng. 269).

WARNING OF "APPROXIMATIONS"

I am simplifying this introduction by a form of my non-dimensional methods that should strictly be called "approximations". I developed these methods privately for a Doctoral thesis whilst based at Vickers during the 1939 War, in what is now the headquarters and subsonic civil section of BAC. I was impressed by the performance simplicity of the early Whittle engine that we flew in a Wellington test bed, and soon accumulated some understanding of gas turbine characteristics. It was apparent that simpler methods would allow much better understanding of the complete propulsion and aircraft design integration problem than endless working through thick files of dimensional calculations, as with piston engined designs. (The increasing importance of Mach number on the Spitfire and our other fighters was beginning to confuse conventional methods, already complicated by cooler systems, lean/rich mixtures, supercharger gearings and propeller pitch settings.)

Most engine people soon welcomed more general methods for summarising engine/aircraft performance. I re-wrote my general analysis into a design manual* whilst awaiting permission to move to English Electric, now the military division of BAC. I was already applying my "approximations" to the early Canberra design. My responsibilities in this and later designs increased so rapidly that I never found time to submit my thesis when security restrictions were lifted later. (It is interesting that our German friends, who were ahead in many aspects of high speed propulsion and aerodynamics, did not use such methods until after the War).

I put "approximations" in inverted commas because we were never able to measure any consistent error in the early Whittle, Canberra or Lightning days. We have only been able to detect small effects in fairly extreme flight situations with the help of more recent high altitude engine test beds. We apply corrections fully only with computers, and Appendix 2 shows why we are running out of time or money even to do this. It is much simpler to give you this "approximate" feel for the preliminary design problem first. But I have restricted the engine curves from Fig. 6 to flight conditions that were actually used in the Canberra matching. (A current problem is that the engine companies continued to supply data well outside this, e.g. Fig. 23. The corrections in such conditions can well be important, but academic to aircraft designers and operators).

---

*Summarised for our new and very small staff in a handbook that is freely drawn upon here: "Analysis of Jet Propelled Aircraft Flight Performance", English Electric Report No. Ae 21. (The preceding 20 numbers were left open for reports written in the previous period when English Electric designed aircraft before 1925; but their Chief Designer had retired without known staff, and a complete list of reports was never located.)

Fig 7 CANBERRA B.MK.2, P.R.MK.3 AND T.MK.4
LEVEL SPEEDS AND CEILINGS
(NO DROP TANKS)

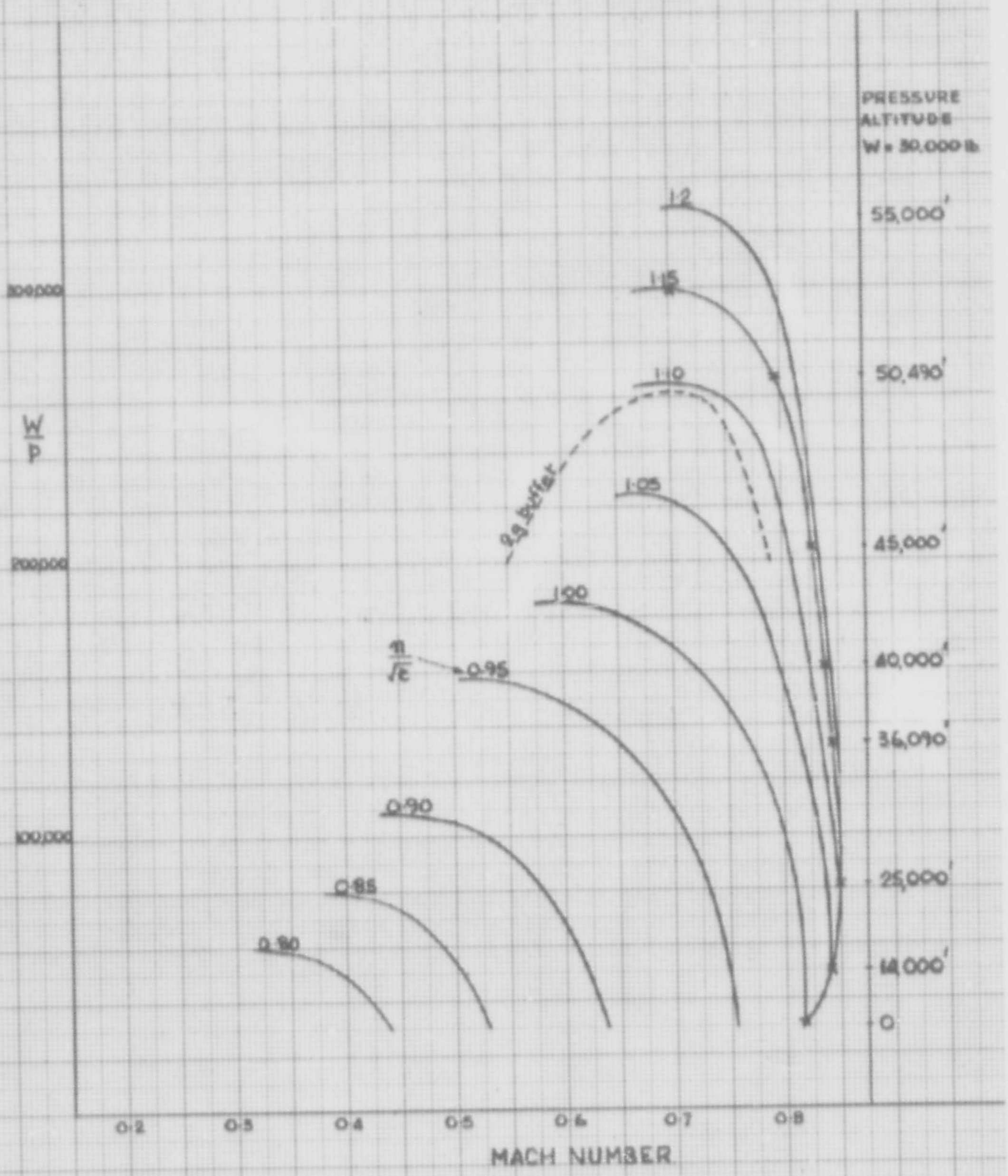


FIGURE 7 plots all the level speed and ceiling intersections on Fig. 6.

The altitude scale on the right has been added for W = 30,000 lb., which defines SL. Other points for h = 36,000 and 50,500 feet (14,400 feet more) can be remembered, since W is factored by nearly 4.5 and 9 respectively.

The curve with crosses marks out the particular case of maximum level speed (n = 1.0) in the standard atmosphere. $^n/\sqrt{t}$ is just over 1.15 above 36,000 feet, and 1.0 at sea level of course. If intermediate heights are often required, it may be useful to remember that 1.05 and 1.10 at 14,000 and 25,000 feet respectively correspond to 1.7 and 2.7 factors on W.

If Fig. 4 is to hand, it is easy to plot a scale every 5,000 feet say. Also to convert M into true airspeed or EAS by multiplying the appropriate speed of sound.

This particular curve shows a maximum level speed above M = 0.8 up to nearly 50,000 ft. and a ceiling of 53,000 feet. Other particular curves can be picked off for max continuous n = 0.95, or for any other rpm or Weight. Similarly for the non-standard atmospheres that occur in actual operation.

Due to its completely general form, Fig. 7 was useful for correlating all later level speed and ceiling measurements. It was even more useful to measure thrust and fuel flow to correlate directly with Fig. 6. Canberra flight test correlations were excellent, allowing for deliberate conservatism in the drag estimates.

Whilst we were designing the Canberra, we were also manufacturing Vampires under licence, and carrying out combat research with a Meteor. Our pilot concluded that it would be desirable to be able to pull nearly 1g extra without buffet on a long high altitude penetration in fighter defended areas. The dotted curve (obtainable later from Fig. 11) matched the continuous curve pretty well at likely temperatures just above standard, i.e. $^n/\sqrt{t}$ nearly 1.10.

FIGURE 8 converts the fuel intersections from Fig. 6 into W times air miles per lb. of fuel. Specific range is the direct yardstick for range calculations, and will be a most useful measure of transport efficiency later. It is most simply obtained by multiplying Fig. 9 ordinates by 661 M.

Canberra range is seen to be maximised by flying just above the minimum drag speed at a Mach number just below 0.75, at altitudes corresponding to constant $^W/_p$ = 250,000. The previous Figure showed this possible with maximum continuous rpm in the stratosphere, with the aircraft climbing slowly past 50,000 feet as fuel burnt off. (Range is seen to be insensitive to small departures from these conditions and the Canberra proved very easy to fly, despite early critics who suggested we ought to descend to allow flight at 30% above minimum drag speed!)

The aircraft could climb even higher, or to Mach numbers well above 0.75 for a few minutes at max rpm without much effect on range. Together with around 1g buffet margin, these were our main design points at a time when it was correctly anticipated that the Russians or their friends would have to struggle to achieve even their early jet fighter performances. Much attention was paid to maintaining this match of airframe and engine in 1945/46.

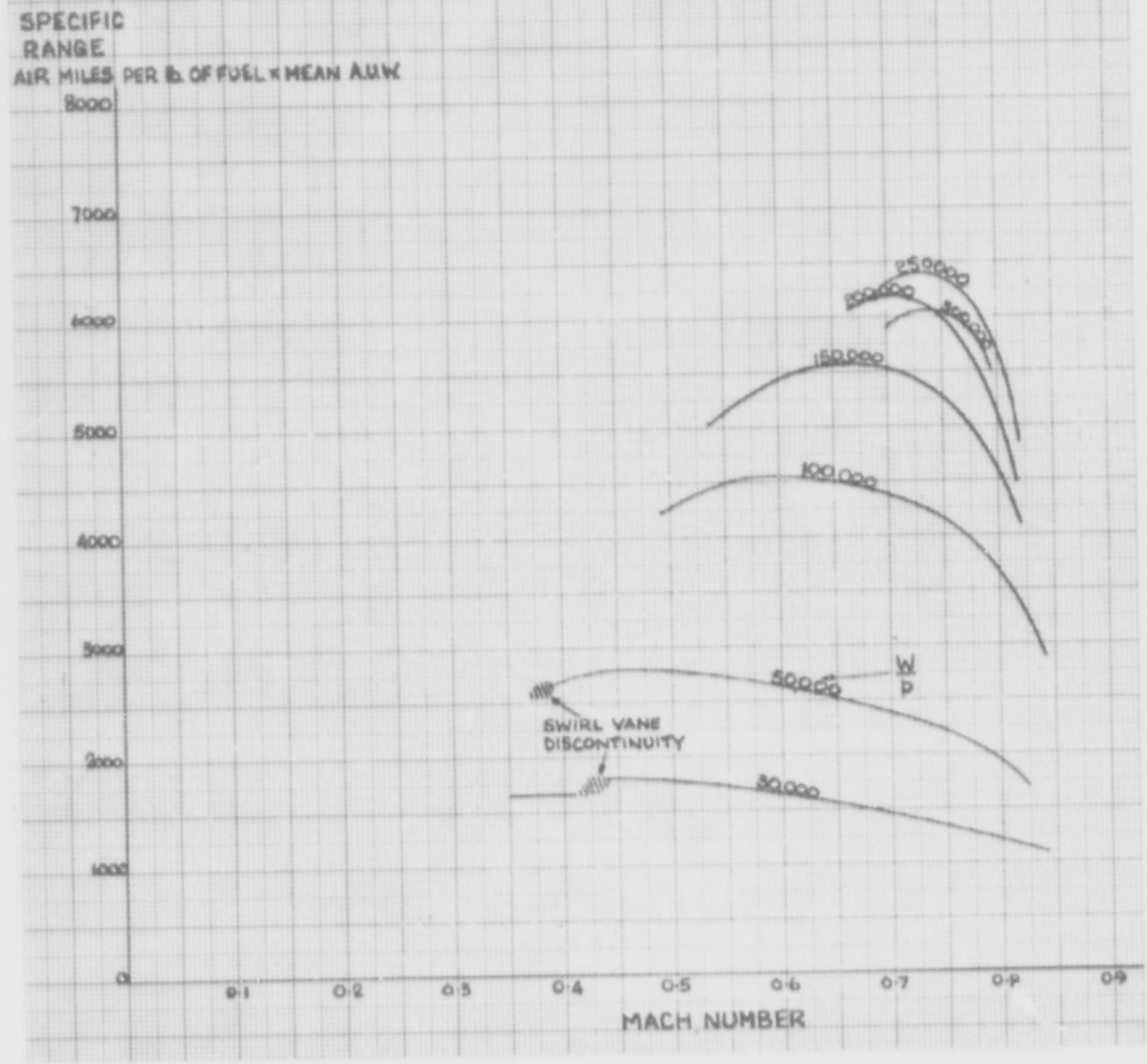
Fig 8 CANBERRA B.MK 2, PR MK 3 AND T.MK 4
SPECIFIC RANGE
(NO DROP TANKS)

RANGE INCREMENT $= (\text{SPECIFIC RANGE}) \log_e\left(1 + \frac{W_f}{W_2}\right)$ AT CONSTANT ALTITUDE

$\simeq (\text{SPECIFIC RANGE})\left(\frac{W_f}{\text{"MEAN" } W}\right)$

WHERE W_f = FUEL WEIGHT USED

W_2 = END WEIGHT



At the average specific range of 6500 on typical missions,

$$\text{Range} = 6500 \log_e \left(1 + \frac{W_f}{W_2}\right), \text{ where } W_2 = 28{,}750 \text{ lb and } W_f = 22{,}150 \text{ lb.}$$

= 3700 nm for a PR mission with the later integral wing tanks.

We had Empire commitments in those days!

This range increased well beyond 4000 nautical miles by adding 4000 lb. of fuel in drop tanks (these increased D_o/p by 15% and reduced specific range by nearly 10% after working back through Fig. 6).

When dropping tanks or other stores, or changing mission altitudes and speeds, the approximation gives a safer understanding of each stage of the mission. I wrote a chapter defining a "mean" (slightly below the arithmetic mean) weight, including analytical allowance for the slow climb implied at constant W/p. It is seldom worth the trouble in practice. If you have data accuracy to justify it, break the mission into further artificial stages that use less than 10% of the weight in fuel.

Constant altitude cruise (including civil)

Subsonic airliners are no longer allowed to cruise climb. They operate lower down at constant altitude, so that W/p is falling as fuel is used. As this necessitates mean specific ranges, artificial stages are again the safest approach. (The pilot will do this on a long flight by changing flight level by 2000 feet or more in stages.)

Fig. 8 illustrates why my range chapter showed that the optimum speed for range even at lower altitudes is never as much as 30% above the minimum drag speed, dependent on the rate of increase of sfc (around 25% for turbojets, dropping to 15% for the latest turbofans, due to their higher sfc rate). But operating costs, military considerations or transport competition (before the energy crisis!) can increase optimum speeds slightly.

Some Canberra missions were forced down towards sea level for part of the time. Fig. 8 shows that specific range falls to about one third of the high altitude value with W/p between 30,000 and 50,000 lb. The fuel used in extra climbs and descents can also be important and will be discussed later.

This secondary design emphasis has given the Canberra an extended lease of life since the pendulum swung further towards low level operation. Induced drag (span loading) is then of small importance, and span is a very positive disadvantage at high speeds in rough air. Without the Canberra's primary high altitude emphasis, it has been possible to give much more extreme low altitude emphasis in our latest aircraft. Throttling back can then prevent the reduction of sfc at lower speed, so that the optimum speed can exceed 130% of minimum drag speed in the lightest, cleanest configurations.

Even more extreme loiter endurance like AX would have put back emphasis on induced drag, but the operators did not fancy prolonged loiter over the defences we already had to face in Europe.

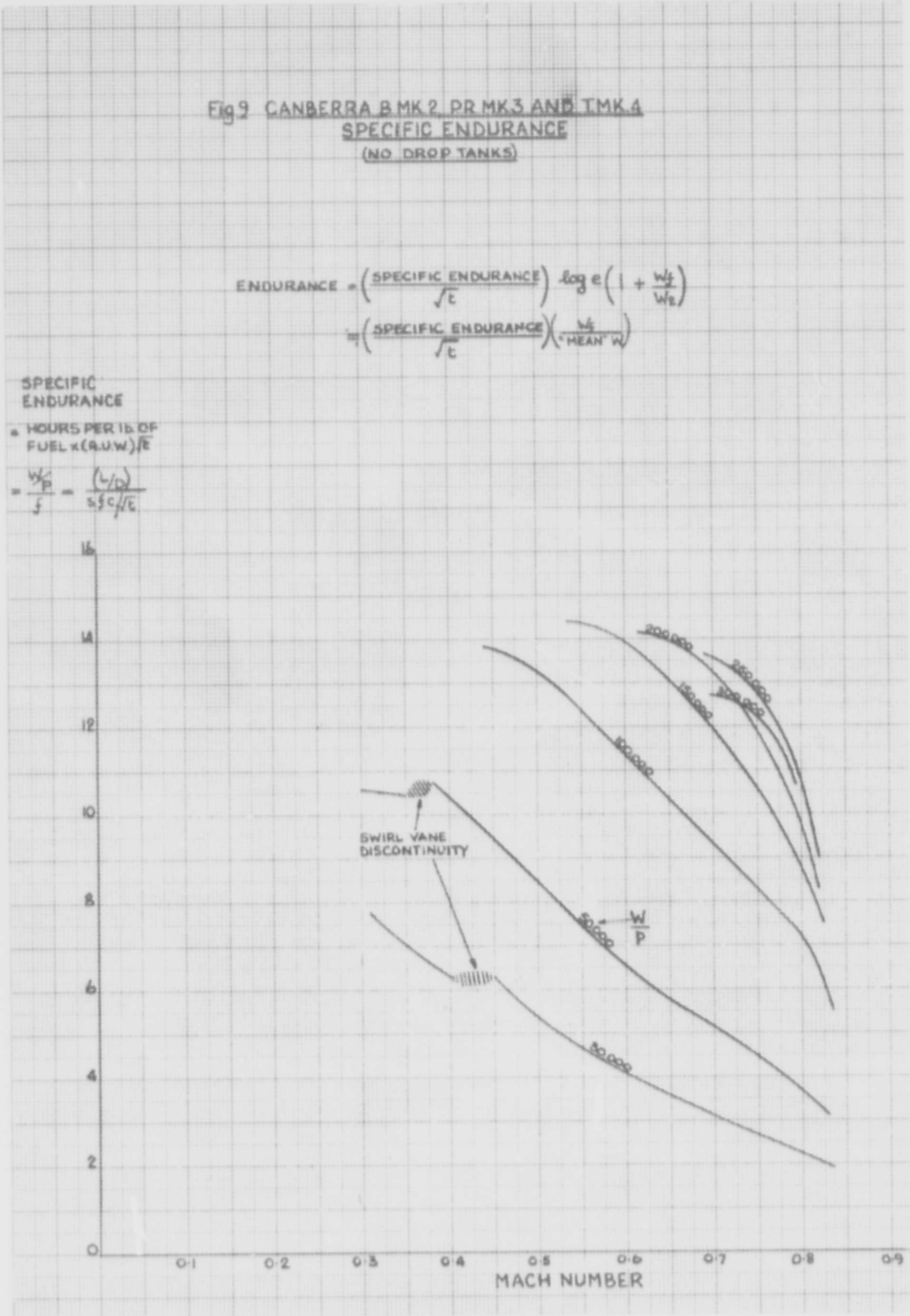


Fig 9 CANBERRA B MK 2, PR MK 3 AND T MK 4
SPECIFIC ENDURANCE
(NO DROP TANKS)

FIGURE 9 is for those specifications that have any significant loiter emphasis. The fuel flow intersections from Fig. 6 are divided into W/p to give specific endurance. The equations at the top of Fig. 9 then calculate the loiter endurance in similar ways to that described for range.

Even the early Canberras had very many hours endurance near minimum drag speed, particularly at altitude (flying steadily below minimum drag speed, as desired for the latest turbofans, is not very practical). Swirl vanes also show graphically, inhibiting maximum endurance at low altitude. Quite apart from the sfc penalty, pilots did not like prolonged operation with swirl vanes. Calculation and test showed that cutting one engine gave improved loiter. (As well as halving the thrust and fuel flow on Fig. 6, add in a windmilling drag coefficient of around 0.5 based on nozzle area, which is usually conservative enough to allow for cruise rudder. My full method embraced the emergency cases at full thrust, with much larger amounts of rudder, sideslip and lateral control down to minimum control speeds).

Compressor improvements on later engines extended this vast Canberra endurance more normally to sea level. If greater emphasis still is required at the expense of all else, this is the case for the fan engines discussed later for subsonic transports.

Conclusion on steady level flight: All the results on previous pages were easily obtainable from the intersections of T and D on Fig. 6, giving direct understanding of all the steady level flight factors involved in the preliminary design phase. Similar charts are still in use for deriving all actual operational planning 30 years later. NB: FIRST READERS MAY LIKE TO SKIP TO PAGE 9, NOTING THE FURTHER WARNINGS ON COMPUTER OUTPUTS THAT DO NOT PLOT OUT BASIC UNDERSTANDING.

FIGURE 10 merely adds two simple examples on how to extend this understanding to the unsteady flight conditions away from such intersections. The first will lead to practical pilot techniques for acceleration and climbs in terms of moving across and up Fig. 6.

Example 1: Consider opening full throttle after a loiter at 0.4 M near sea level. The simple sum shows that level Mach number is increased by 0.08 in 10 seconds. (The pilot could alternatively accelerate at 0.18g on a gradient of 0.10 radians, or decelerate at -0.10g whilst climbing at 0.38 radians. Only our computer programs now think they know what he might be doing in combat. I will leave my alternative analysis of energy height to the Lightning, where it proved more useful).

The other alternative is a climb at constant true speed of 7500 feet per minute. It actually pays to accelerate to higher M to the right where the rate of climb is higher (particularly with the much greater thrusts that the Avon developed later). We used to call such academic numbers "partial rates of climb". They are given the more impressive name today of SEP - Specific Excess Power = $(\frac{T-D}{W})V$. Others may try and explain why some have given it such a precise air of new importance.(See also Fig. 11).

A pilot would not know how to climb at constant true speed in practice, except in the stratosphere. An operator will usually want to know precisely what happens when he climbs under rules he can follow. Climbing at constant IAS or M usually implies acceleration or deceleration. The simple equations summarise my results for turning partial into true rates of climb.

My original analysis also showed that the quickest climb occurred at a speed more than half way between the minimum drag speed and the maximum level speed (even if the latter neglects the drag rise beyond M_{crit}). If flight endurance were important, minimum fuel to altitude occurred at slightly lower speed and fuel flow. Range was more usually the deciding factor on Canberra, so the optimum speed was even faster. This analysis becomes academic, after providing early

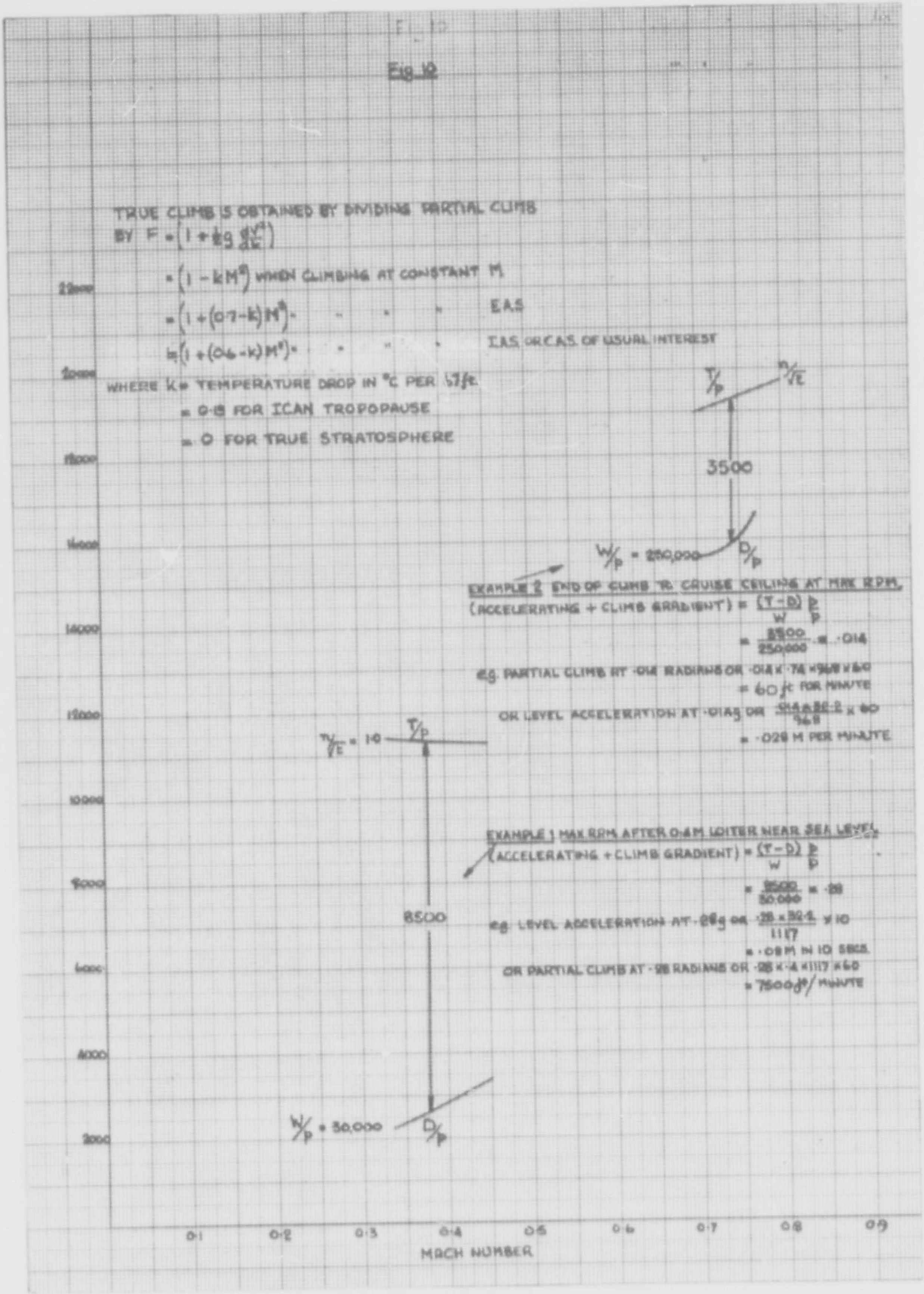


Fig 10
TRUE CLIMB IS OBTAINED BY DIVIDING PARTIAL CLIMB
BY F = (1 + 1/2g dV²/dh)
= (1 − kM²) WHEN CLIMBING AT CONSTANT M
= (1 + (0·7 − k)M²) " " " EAS
≈ (1 + (0·6 − k)M²) " " " IAS OR CAS OF USUAL INTEREST
WHERE k = TEMPERATURE DROP IN °C PER 57 ft
= 0·19 FOR ICAN TROPOPAUSE
= 0 FOR TRUE STRATOSPHERE
T/p
n/√θ
3500
W/p = 250,000
D/p
EXAMPLE 2 END OF CLIMB TO CRUISE CEILING AT MAX RPM
(ACCELERATING + CLIMB GRADIENT) = (T − D)/W · p/p
= 3500/250,000 = ·014
e.g. PARTIAL CLIMB AT ·014 RADIANS OR ·014 × ·74 × 968 × 60
= 60 ft PER MINUTE
OR LEVEL ACCELERATION AT ·014g OR (·014 × 32·2)/968 × 60
= ·028 M PER MINUTE
n/√θ = 1·0
T/p
EXAMPLE 1 MAX RPM AFTER 0·4M LOITER NEAR SEA LEVEL
(ACCELERATING + CLIMB GRADIENT) = (T − D)/W · p/p
= 8500/30,000 = ·28
e.g. LEVEL ACCELERATION AT ·28g OR (·28 × 32·2)/1117 × 10
= ·08 M IN 10 SECS
OR PARTIAL CLIMB AT ·28 RADIANS OR ·28 × ·4 × 1117 × 60
= 7500 ft/MINUTE
8500
W/p = 30,000
D/p
22000
20000
18000
16000
14000
12000
10000
8000
6000
4000
2000
0·1
0·2
0·3
0·4
0·5
0·6
0·7
0·8
0·9
MACH NUMBER

guide lines. The optimum IAS on any criterion varied with altitude, and soon approximated the critical Mach number. Climb at constant IAS gave negligible increase in time or fuel, and the only way to determine the practical optimum was to compare two or three values of IAS. With more thrust from the Avon, this became the maximum acceptable IAS at low altitude (to avoid a shattering sonic boom in the Lightning case discussed later).

If any of your have gone straight into the computer age without doing graphical acceleration/climbs, it is useful to get some feel for a few areas by hand. I am not suggesting you do this employing steps of a second or so, with the meticulous accuracy of a computer - we had much prettier calculators to do that, even before electronic computers. Cheat with larger steps, and you soon get the feel of guessing ahead to "mean" conditions over a large increment. After our more patient calculators had done a number of cases, we confirmed the Canberra approximation that the net effect of an optimum climb/descent was around 50 nm off the low altitude range, or 150 nm off the high altitude. This was usually good enough in a flight of a few thousand miles, but my generalised analysis showed how to correct even this correction for other design conditions.

Example 2: The other simple calculation showed that the Canberra reached cruise altitude with the climb rate falling rapidly to 60 feet per minute. Since the climb rate was much more just before, this was quite fast enough to climb the last bit in a mission calculation of some hours.

But the actual number immediately revealed the poor climb or acceleration (0.028 M per minute) when the throttles were opened after sighting a fighter. We immediately realised that it would be too late to rely on warning from the eyes, so simple radar was developed.

We are now doing all such calculations and many more by computer, with accuracies way beyond those justified by the basic data. This is perhaps sufficient for some detail design, flight test and operational stages. But most of the print-outs cannot be scanned by all the top people in the more fluid early stages, and there can be inevitable lack of feel for such problems or solutions, e.g. sitting round the table with the engine company, trying to agree the best design matching. It is useful to have some graphical illustration of the main problems, even if plotted directly from the computer.

One other dreadful possibility comes to mind, and which almost happened on the Concorde for different engine reasons. If we had had computer outputs only in the mid 1940s, the engine people would have dutifully supplied card deck engine data from max thrust down to a large idling drag. Swirl vane operation would have been neatly incorporated out of sight in the middle, with the bleed valve operation that was there from the earliest days.

Fortunately, Fig. 6 showed it as a graphical discontinuity across the middle, and it was not certain to be so reliable or low an rpm in earlier days. It struck us immediately that there might be difficulty in getting the aircraft down quickly to low altitude. Pressure cabins were not all that proven in those early days, quite apart from what might be needed over enemy defences. Although air brakes were something of a novelty in that era of large propeller idling drags, we modified the design in time! (It can be difficult or impossible to incorporate powerful air brakes at a late stage without major trim, buffet or structural problems.)

The Canberra air brakes nearly doubled the drag. Everything is calculated as before, with (T-D) and everything else negative, i.e. numbers give descent gradient plus deceleration.

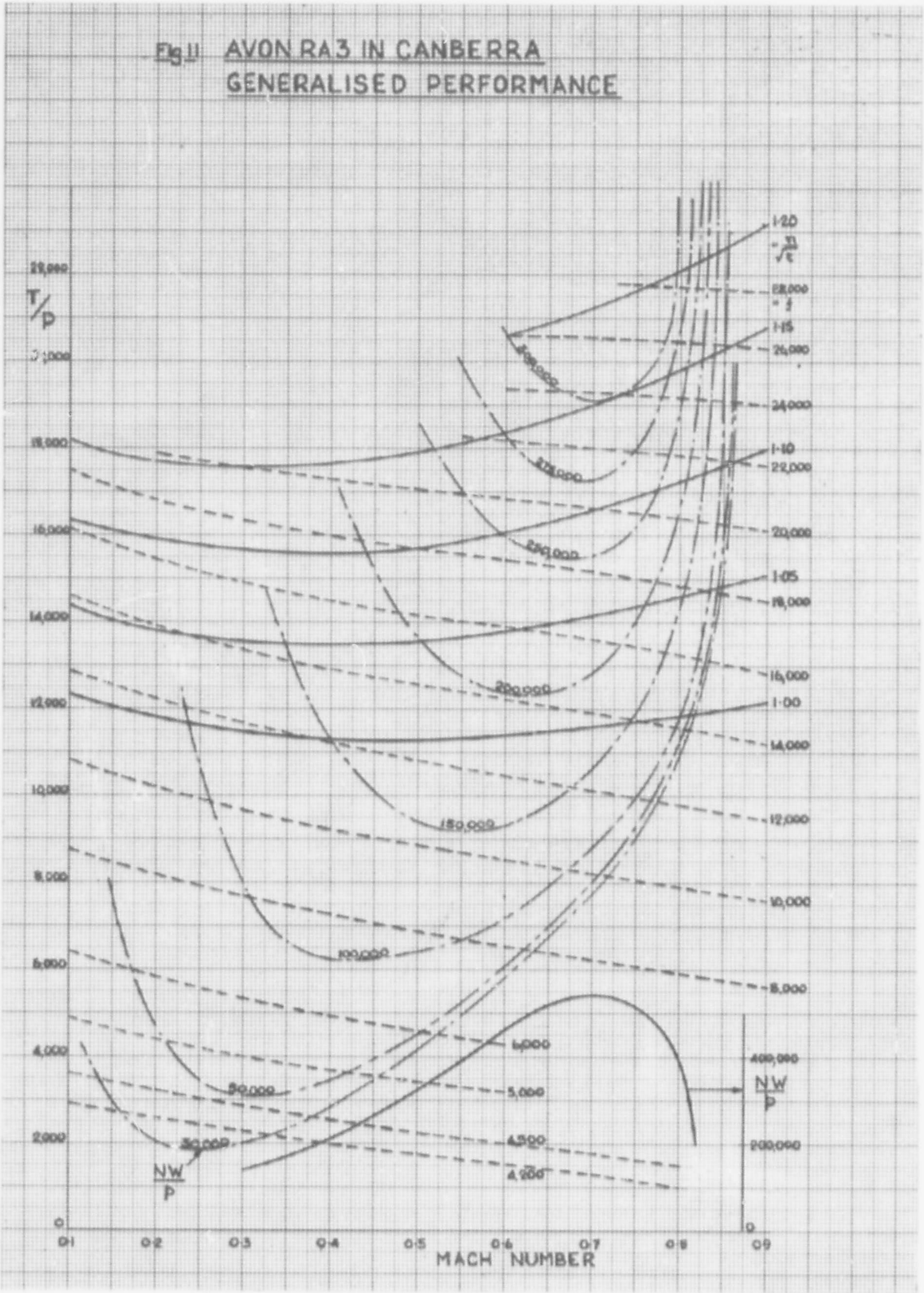


Fig 11 AVON RA3 IN CANBERRA
GENERALISED PERFORMANCE
T/P
MACH NUMBER
NW/P
300,000
275,000
250,000
200,000
150,000
100,000
50,000
30,000
1·20
1·15
1·10
1·05
1·00
28,000
26,000
24,000
22,000
20,000
18,000
16,000
14,000
12,000
10,000
8,000
6,000
5,000
4,500
4,200
400,000
200,000

FIGURE 11 gives performance under Ng. One little trade secret: the original well-proven curves are simply extrapolated to the left of the minimum drag speed; inversely with M^2 as described before. They are also re-labelled $\frac{NW}{p}$, as lift now equals NW instead of W.

We should strictly have done this from the start, since Lightning and later aircraft had climb gradients more than a radian. (We should have written $\sin\theta$ where we said radians before, and $N = \cos\theta$. One soon moved towards academic situations with vertical climbs at $N = 0$, and SEPs around 1000 fps at high subsonic speed.)

The Reynolds number correction in Appendix 2 could become more important, but the customer has never been interested enough to allow us to try and measure it. This is particularly true for the extrapolation to the left of the minimum drag speed. This implies the same K, which our general experience suggests would be higher near the buffet region, although there is general lack of reliable data.

But the lift at which the buffet occurs was estimated and measured as accurately as possible, and the resulting $\frac{NW}{p}$ is plotted bottom right. Copies of these charts held by people in contact with operators are dotted with cross-checks only near steadier conditions above the minimum drag speed.

This makes one somewhat sceptical attempts to make too precise a science out of the SEPs under g, or steady turn rates. All such values can of course be read from this chart, (and even more easily from computers) but it seems they have been used at the lower altitudes only to get a rough idea of the g that can be pulled. (As practical experience shows cockpit and other factors even more important, data for our combat simulators is discussed later.)

The situation was different in the days of high altitude missions, when performance margins, g's and turning radii were so much poorer, as well as fighter weapon systems. A small g advantage could then be decisive, and a lot of Canberra and other measurements were made. These checked very well with the results from Fig. 7. As expected from the discussion of p in Fig. 5, the ceiling (or altitude at given M) dropped by slightly over 1000 feet for every 5% increase in N, or by 14,400 feet if N is doubled. If the aircraft dropped out of the stratosphere, the rate of drop becomes greater. This was particularly true for steady turns, since $^n/\sqrt{t}$ and $^T/_p$ also drop.

Take-off drag resistance curves were also added. But airfields seldom proved critical for Canberra operation. It was sufficient to approximate the somewhat uncertain rolling friction, assuming $p = 0.95$ for a small range of airfield altitudes. The more important increase in $^D/_p$ for undercarriage and flaps was represented precisely, both in the air and on the ground. Take-off is simple longitudinal acceleration at $(\frac{T-D}{W})g$, up to a speed M at which there is sufficient lift and (T-D) to pull g and climb over the obstacle.

(Thence to approximations that ground roll $= \frac{1}{2}\frac{V^2}{\text{accn.}} = \frac{1}{g}(\frac{W}{T-D})\ \frac{W}{SC_L}$ etc.)

The aircraft is cleaned up after this, so that the drag merges quickly into the previous acceleration and climb calculations.

Fig. 13

Effective Aspect Ratio

The usual drag equation with constant K can be differentiated to show:

$$\max {}^{L}/_{D} = \tfrac{1}{2}\sqrt{\frac{\pi A}{KC_{D_o}}} \quad \text{when } C_D = 2C_{D_o} \quad \text{at } C_L = \sqrt{\frac{\pi}{K}AC_{D_o}}$$

Since C_{D_o} varies so much with wing area changes, it is more useful to re-write the above in terms of $C_f = \left(\frac{S}{S_f}\right)C_{D_o}$ based on the total aircraft surface area, S_f.

$$\text{i.e. } \max {}^{L}/_{D} = \sqrt{\frac{\pi}{4KC_f}}\,A_{eff} \quad \text{at} \quad C_L = \sqrt{\frac{\pi C_f}{K}}\left(\frac{S_f}{S}\right)A_{eff}$$

$$= 14.6 \; A_{eff} \quad \text{at} \quad C_L = .093\left(\frac{S_f}{S}\right)A_{eff} \text{ for } K = 1.15 \text{ and } C_f = .0032$$

on a clean design like Canberra. This also shows the importance of an effective Aspect ratio,

$$A_{eff} = \frac{b}{\sqrt{S_f}}$$

$$= \frac{64}{\sqrt{3400}} = 1.1 \text{ for Canberra, with } \frac{S_f}{S} \text{ only } \frac{3400}{960}$$

So that Canberra max $^{L}/_{D}$ = 16 at C_L = 0.36

It is interesting to check what happened on an extreme configuration like B47, with an effective aspect ratio of 1.4. Although much larger aircraft always have improved $^{L}/_{D}$ due to relatively smaller fuselage and so on, the measured $^{L}/_{D}$ at low Mach number was only about 18 at a very high C_L. This was with aileron reversal limits on low level operation, and despite resorting to a bicycle undercarriage with cross-wind landing limitations.

Canberra/Avon Preliminary Design Considerations

So that Fig. 11 (or even 6) had most of the important performance factors that matched Avon to Canberra. The original Rolls Royce offer to consider scaling the engine lasted very little time as other applications became apparent.

The engine and aircraft diagrams were later superimposed optically, so that the relative scales could be adjusted in various ways for those who still doubted. Few critics took up our offer 200 miles out of London! Many considerations have already been touched upon, and the overall match is essentially a compromise between conflicting requirements.

Rolls Royce started making the Avon so quickly that there was little serious chance for further consideration. Judging from my jet engine optimisation discussed later, they were pushing the pressure ratio and TET as far as they dare in those very early days. The Avon was Rolls's first axial jet, and gave large gains over the centrifugal compressor considered earlier.

The usual temptation towards scaling up a load-carrying aircraft, despite reduced numbers for given cost, was met by the V bombers. Engine size was itself limited by the fear that we might need to fall back on an existing centrifugal engine (one prototype was actually converted in a period of surge worries) plus Rolls Royce's confidence that the engine would eventually develop 15% more thrust, and probably much more. This would increase range and altitude particularly, like C on Fig. 14, but not if engine size were scaled from the start with other resultant weight and cost growths. We cannot show you a mass of parametric cross-plots typical of today. We saw them all optically in the early versions of this diagram. The one or two people I had on performance considerations were much too busy on detail design to prepare a massive paper justification for 30 years later. (A similar problem 30 years hence will be whether the right cabinets of computer print-outs have been kept!)

But we soon faced the more usual criticism that we had matched the wrong aircraft to a given engine! A government visit to America to see the B45 and later designs brought a hornet's nest around our heads.

Quite apart from these new American ones, all previous long range high altitude aircraft had Aspect ratio 6 to 10 or more. So that the Canberra's aspect ratio of 4.3 took some getting used to, and its wing loading seemed much lower than the general trend. Most people were convinced by our claim that every change in our optical system led to serious deterioration somewhere, after hearing that:-

FIGURE 13 defines an effective Aspect ratio that seemed more relevant to me than the usual definition. Consider the extreme case of halving the wing chord to double the usual definition of aspect ratio. Due to the smaller reduction in total surface area, S_f, this only increases the effective aspect ratio to about $1\frac{1}{4}$.

The combined effect is to nearly double the C_L. For a start, this would remove nearly all the buffet G margin we were struggling to obtain. Furthermore, the best high Reynolds number data we could find after the War (in Germany) showed that any increase in C_L would rapidly increase the value of K. Equally bad, halving the wing chord and thickness would no longer bury the engines, the "soft" undercarriage and part of the fuel. Since the drag of these and other items like canopy and aerials, etc. would be based on a reduced value of S_f, this would all increase the value of C_f significantly, and worsen the critical M.

The combined changes in K and C_f could more than cancel the gain in L/D from effective aspect ratio. So this gives poorer or more uncertain M, and most of the buffet margin is lost for certain.

FIG. 14

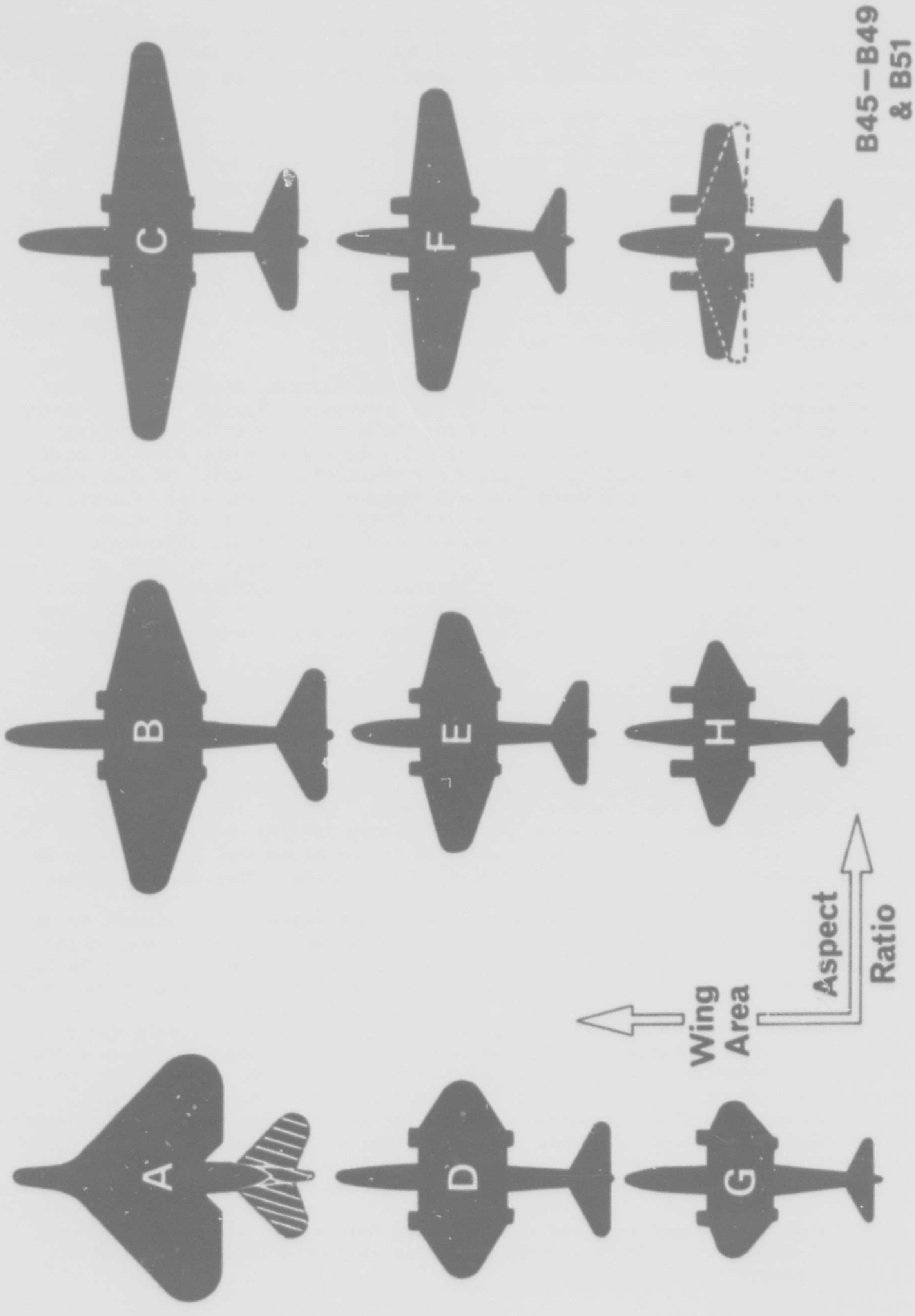


FIGURE 14: Canberra developments and V-bombers

For our own satisfaction, and then to join in the V-bomber studies of 1946, we analysed other large systematic changes from the basic design. Design E was the Canberra design we had chosen. The others changed aspect ratio and wing area in a systematic fashion, each using Avon engines.

As with the computer baselines discussed later, each design was fully studied in respect of all crew requirements including vision, plus all the needs of the main items of equipment, structure, propulsion and fuel systems etc. Similarly for stability and control, CG range, external tankage etc., and the other essentials for all aspects of realistic engineering design. Since it is the role of the other speakers to talk about airframe optimisation, with all the implied sciences or arts of final weight, drag, lift and cost estimation, I will be brief in mentioning just a few results.

Aircraft J was closest to the B45 and later designs. It proved to have much poorer buffet, ceilings and range.

We found it better to move towards A for long range high altitude performance, but this required three or more Avon engines. The Vulcan and Victor V-bombers eventually went ahead with new engines after our 4-engined Valiant, with the Short SA4 falling by the wayside. But the Canberra was much cheaper and better at low altitude, and had already gone ahead much more quickly. The Avon eventually developed more thrust, and take-off weight developed over 50%, to beyond 60,000 lb. Even greater thrust was developed for the PR9 high altitude version, but the RAF would not compromise on low altitude speed and G limits like the USAF.

In an early visit to Wright Field, I found they would accept severe structural limitations at low altitude to permit even higher altitude missions. C was best of all for such missions, and bears a strong relation to the RB57D and F.

If results are anything to go by, far more Avons have been built than any jet engine outside America. Canberra production far exceeded any other non-American aircraft since the War, apart from small fighter aircraft. The Canberra was built in Australia as well as America, and we are still pursued by sales enquiries!

In that first year or two of jet-propelled strike/reconnaissance designs, the circumstances described on the last two pages forced us to explore this wide range of possibilities. We are highly suspicious of computer programs that claim to embrace anything approaching this range today. Design margins of all kinds are far tighter and more complex, and vary so much across a range. Fortunately, nearly 30 years more experience also allows us to home in more rapidly towards the best design. The computer is one of several sophisticated tools described later for optimising the final design in detail.

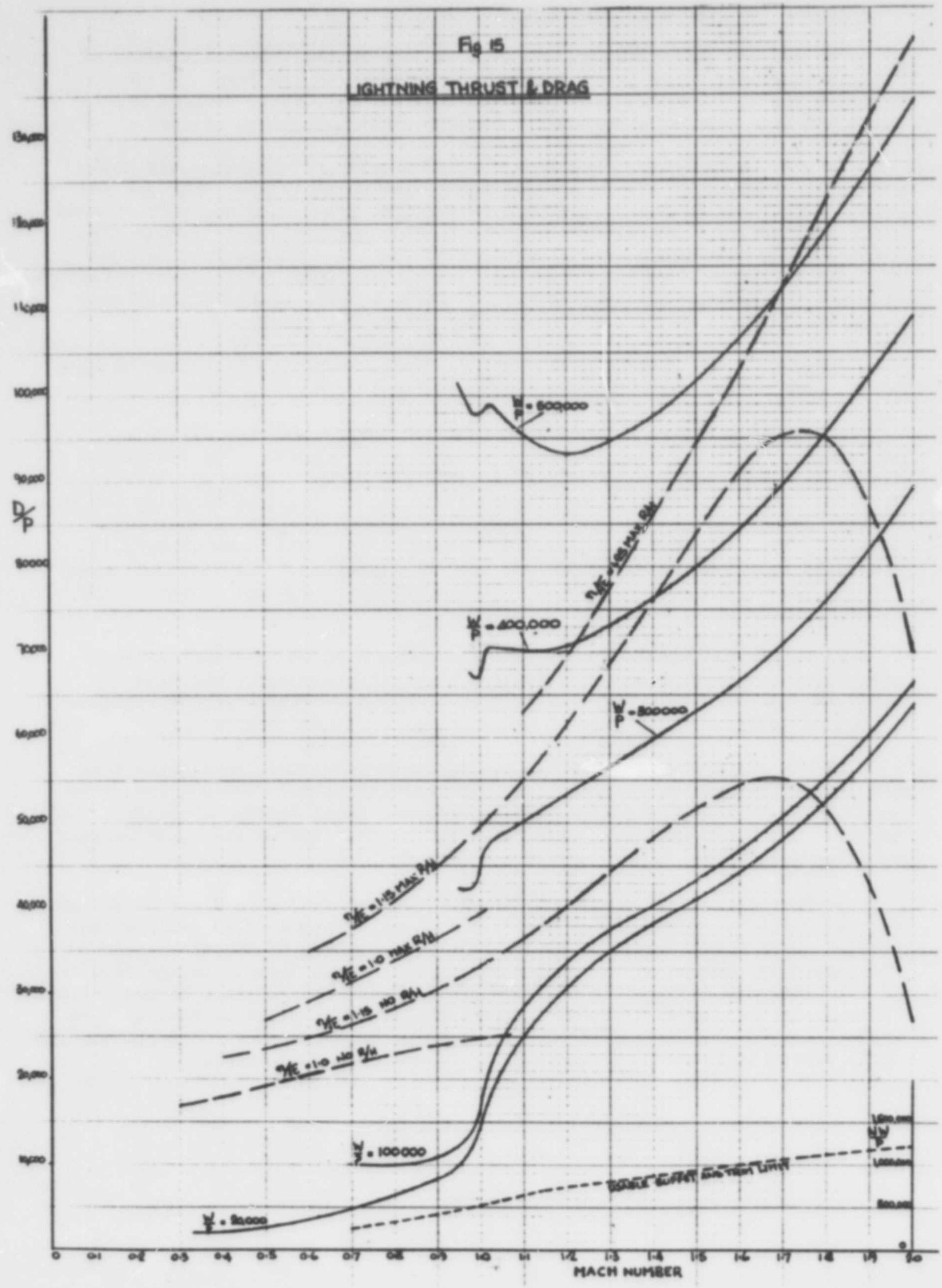


Fig 15
LIGHTNING THRUST & DRAG
D/P
MACH NUMBER
W/P = 500,000
W/P = 400,000
W/P = 300000
W/P = 100000
W/P = 20,000
BUFFET AND TRIM LIMIT

2. LIGHTNING

FIGURE 15 shows the corresponding drag and propulsion curves for the Lightning supersonic fighter, with its later version of the Avon. For clarity in extending the previous topics, only a few curves are shown. Examine the thrust curves without reheat first, since the initial design was in 1948, and the subsequent specification said we could not await the development of reheat. It was urgent to obtain a speed/height advantage over any Russian equivalent of the Canberra or V bombers. The major new design emphasis was on reducing transonic and supersonic drag.

Due to the propulsion installation described later, acceleration was very high towards M = 1. Looking at drag for W/p = 100,000 (corresponding to a final combat weight of around 22,500 lb. for the early aircraft near 36,000 feet), you will see that an acceleration of around 0.08g (0.16 M per minute) is maintained to beyond M = 1.1 at maximum rpm on a standard day. (Are there any aircraft with this performance after 25 more years experience, apart from very large aircraft like the Concorde and B58, which are also assisted by reheat?)

A subsonic bomber was quickly caught by a speed of 1.4 M say. The g capability of the Lightning at transonic and supersonic speeds was then important, as shown by the $\frac{NW}{p}$ curve at bottom right. (Since W less, N was much better than the Canberra on Fig. 11 - vastly so above M = 0.8). Excess speed could be converted quickly into altitude, since we have already seen that a climb angle of more than 0.5 radians can be obtained if we decelerate at 0.5 g (or 1.0 M per minute). This gives a rate of climb of about 700 feet per second, dropping to around 500 feet per second as speed drops off.

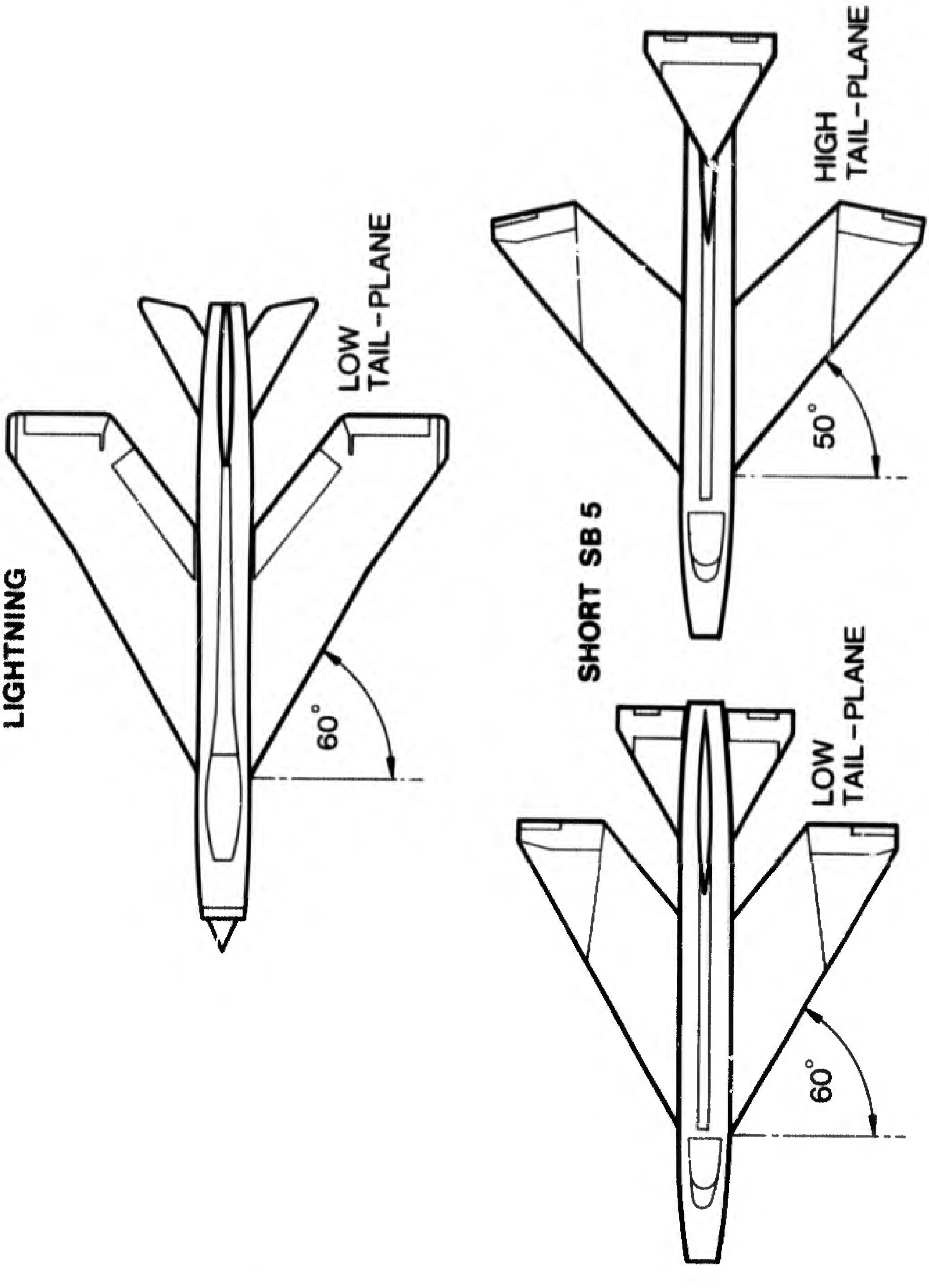
This means that well over 15,000 feet is climbed in under half a minute, to above 50,000 feet. During this time, Mach number has dropped by less than 0.5. This left the Lightning behind the bomber at gun or rocket aiming speeds around 0.9 M.

/This leads to even easier calculation of constant energy climbs, which were very useful to the pilots. Check that $V^2/2g$ changes by 15,600 feet as V drops from 1350 to 900 fps. Since $(\frac{T-D}{W})V$ = rate of change of energy height, varied from small negative to small positive as moderate g was pulled and then removed, the change of energy height in those quick climbs was negligible.

One could elaborate this by plotting lines of constant energy height and superimposing contours of constant SEP, or SEP divided by fuel flow. This showed that time and fuel is saved by climbing above 36,000 feet subsonic and diving down a constant energy line, etc. We even found a design with less sweep that had an "oasis" of performance at around M = $1\frac{1}{2}$ which could only be reached by such a technique. The computer can produce all manner of such results, but a quick look at the T and D curves shows that any such design is too marginal for practical operation on warm days, etc./

REHEAT: As the Russians developed atomic then hydrogen bombs, faster bombers and stand-off weapons, it became increasingly clear that an unreheated Lightning had insufficient margins for short early warning times, particularly under ECM conditions. Reheat was fitted into the first and subsequent aircraft as soon as it was cleared. We had pressed for it from the start, accepting the modest amount of 1600°K. Apart from the engineering problems of the time, the sfc of reheat (based on its extra thrust) is several times that of the basic engine. Current turbofan fighters that rely on 2000°K reheat thrust for most of their performance, also have enormous fuel consumption. This is sad for the present fuel crisis (December 1973).

FIG.16


LIGHTNING
LOW TAIL-PLANE
60°
SHORT SB 5
HIGH TAIL-PLANE
50°
LOW TAIL-PLANE
60°

We still preferred to put more emphasis on basic engine performance, and you will note that this version of the Avon had much more thrust than the original RA3 in the Canberra. Certain aspects of Lightning are still classified, so we must be a little vague about which Mark of Lightning and Avon. Although the reheat thrust was modest by modern standards, you will see that the aircraft had near-vertical climb capability at subsonic speeds - SEP around 1000 fps in modern parlance. Normal operation was a rapid climb at high subsonic speeds, followed by acceleration and turns near the tropopause (although diving lower could save a few seconds on an energy basis, pilots did not like the complication, and it used more fuel).

Although the Lightning achieves $M = 2$, up to over 60,000 feet in cold air, we could never see the "sales" value of designing for this round number on standard or warmer days. Quick acceleration up towards $M = 1.7$ was much more important in our operational analysis, carrying better weapons and other systems. Even with this controlled approach, there was growth in the internal fuel capacity problem that we know on all high performance aircraft today.

Because the RAF emphasis was on supersonic interception, we had to concentrate on the maximum possible acceleration without over-reliance on the far greater sfc of reheat. Many of the subsonic considerations given earlier to the Canberra were necessarily compromised; nevertheless, the Lightning's subsonic L/D of around 10 is still good for a supersonic configuration with a tailplane for large supersonic g. It is very easy to get the L/D nearer 8 (effective aspect ratio down towards 0.6) on extreme supersonic designs like the F104 and fully swept variable geometry.

FIGURE 16

We finalised the Lightning configuration as much in the wind tunnels as by aerodynamic theory. Considering that sweepback even of 30° was fairly new to British eyes, 60° was too much. The discovery that the tailplane had to be below the wing to avoid pitch-up was the final straw! Although this became familiar later, it was all against the fashion in 1949. Detail design and manufacture of the Lightning prototypes was slowed whilst the Government persuaded Shorts to build a low-speed version of our configuration. This incorporated "variable geometry in the hangar", and it first flew with the more conventional sweepback of 49° and a high tail. This gave the pilot a fright as our wind-tunnel tests predicted, and it was soon converted to the Lightning configuration. The Lightning flew very satisfactorily just after, and has done so ever since.

It is fairly obvious from this drawing how two Avons of much more power were installed with surface area and D/q that were little more than half the Canberra. The k was also quite good for a supersonic shape, since much attention was paid to leading edge rounding outboard where theory suggested. But there was no point in overdoing this when the same theory showed that the trimmed value of k increased rapidly with M at supersonic speed, up to a value nearly independent of aspect ratio (the consequences for designs dominated by more supersonic considerations will be discussed later on Concorde).

The reason why critical Mach numbers exceeded 0.9 are fairly obvious from the high sweepback. But the drag and propulsion problems at higher speeds still dominated preliminary design considerations, for reasons we saw on the previous diagram.

The remaining Lightning discussion can now turn to the engine matching aspects of all this.

FIG 17

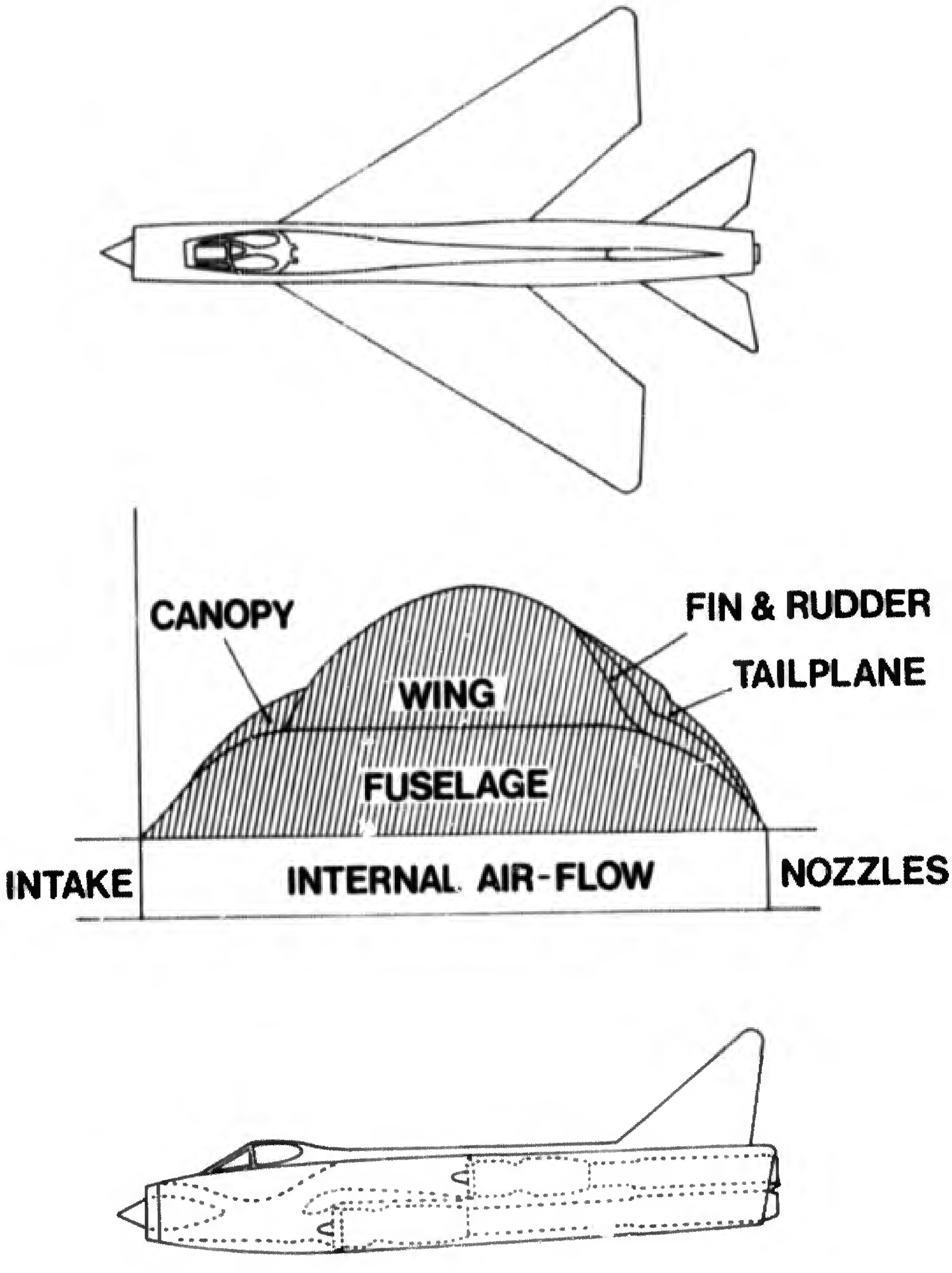

CANOPY
FIN & RUDDER
TAILPLANE
WING
FUSELAGE
INTAKE
INTERNAL AIR-FLOW
NOZZLES

FIGURE 17

This allows concentration on the unique propulsion matching of the Lightning, the subject to which I was asked to pay particular attention.

The engine people were already doing the best they could to increase engine rating and to try and make reheat practical. Their main problem was our new requirement for a variable nozzle, since the cold thrust penalties with a fixed oversized nozzle were much too great.

Our own problem was to install the engines in a practical shape which had acceptably lower drag at supersonic speeds. We had heard of Hayes' 1947 supersonic wave drag analysis, as our mathematicians developed similar theories. These all suggested that the fuselage should be slab-sided over the length of the slender wing. The smooth area plot later attributed to Whitcomb is gross use of the theory for M = 1. It is certainly one rule to be satisfied, but it cannot put extreme "coke-bottle" shapings on the same basis (e.g. cutting a hole on the other side of a fuselage to compensate an aerial!) There was the additional need for practical slenderness and smoothness in all components. Thus, although the canopy and tail surfaces help fill the dip fore and aft of the wing on a simple area plot, their own shapes were even more important.

With the wing structure as thin as would house the undercarriage and fuel, the absolute minimum cross-section of fuselage became of truly vital importance. The only way to achieve this in a slab-sided shape was to stagger the engines one behind the other.

Thence to the unique propulsion installation of the Lightning. The cross-section of one intake, or of a transition pipe forward of the reheat, was much less than that of an engine.

We were of course anxious to confirm the value of what we had decided. We converted our high speed tunnel to proper transonic operation with slotted walls in parallel to John Stack. After John won his first Collier trophy for the X-1 in 1947, I used to pull his leg about its terrible transonic drag and other characteristics. But all the aircraft for some while were the same, and only one or two could go transonic with the aid of large rockets or near-vertical dives. We were attempting to do this without any of the effective extra $T/p = W/p$ or more! We were further ambitious to accelerate, climb and aim guns without reheat or auto-stability. It all came out so well, and we are eternally grateful to John for mutual encouragement in that difficult period.

Although internal aerodynamic unknowns were also formidable, these were dealt with more straightforwardly. The jet pipes were straight, and there was already some experience of long pipes. The intake bifurcation and other shaping was developed in a water channel, and checked out in a large rig.

External intake aerodynamics were largely unknown at transonic speeds, and we frankly funked the boundary layer effects on side intakes. Those who didn't have probably paid a large penalty in efficiency or drag for many years, as discussed later. Even without the unknown boundary-layer penalties in the nose, we had to give a lot of thought to preserving our thrust-drag margin.

The remaining discussion will refer to other propulsion aspects of this.

FIG 18 LIGHTNING INTAKE PRESSURE RECOVERY

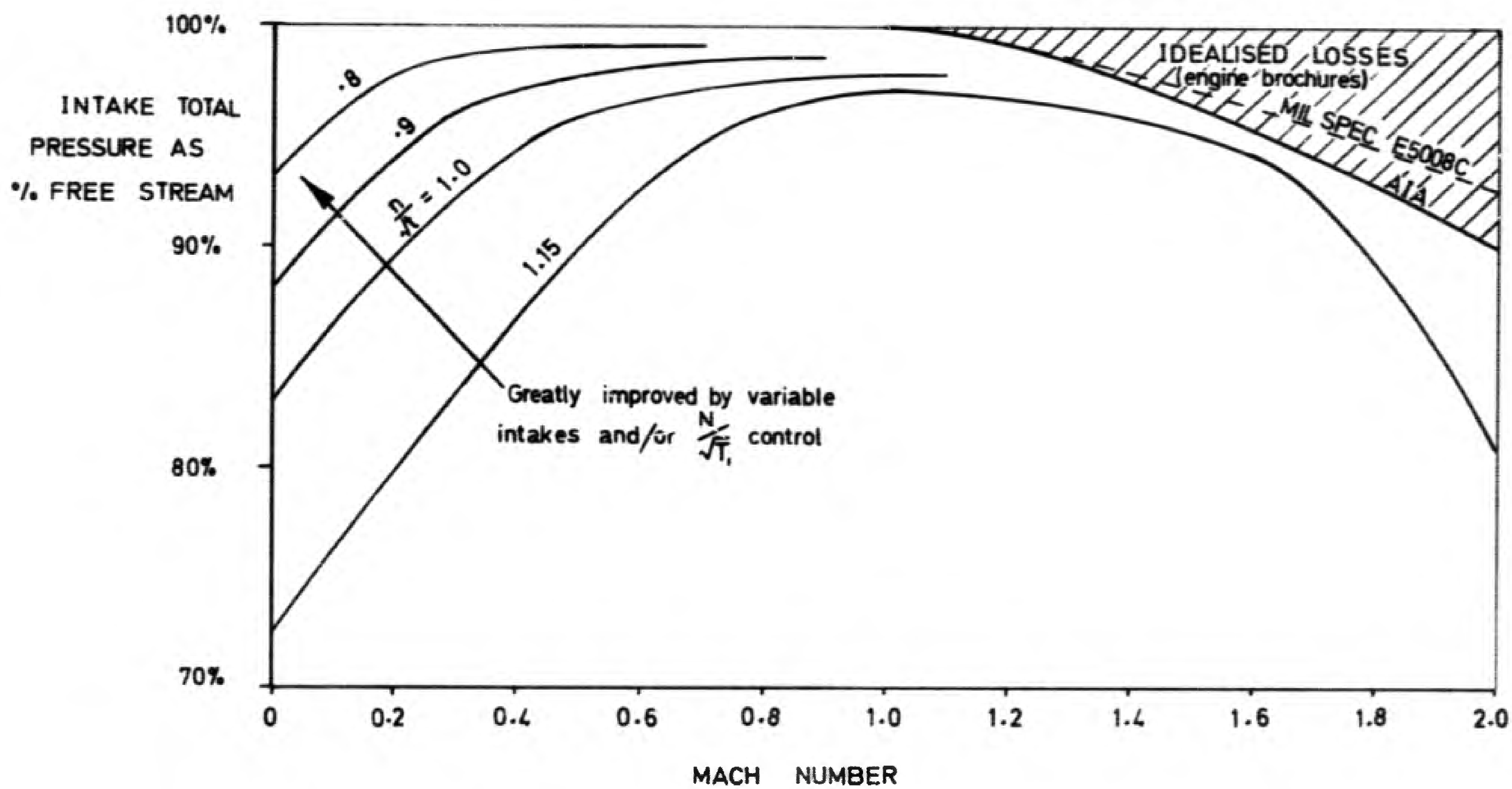


FIGURE 18

I have left discussion of intake effects until now, because they were small and uninteresting on the subsonic Canberra intakes; after we had done a lot of work on the proper shaping and rounding of the intake cowling. Such rounding would have made unacceptable inroads into the Lightning's supersonic margin. The pre-entry spillage "drag" on even a slightly oversized intake starts anyway at transonic speeds, due to the inability of the cowl to carry a forward thrust as aerodynamic suction. (All intakes had previously been oversized to reduce internal losses, and spillage "drag" gives graphic description of the basic $^{10}/_{6}\ (1 - {}^{1}/_{M^2})$ shock drag effect they give). But since all such drags or thrust losses are functions of mass flow, we treated them as thrust losses.

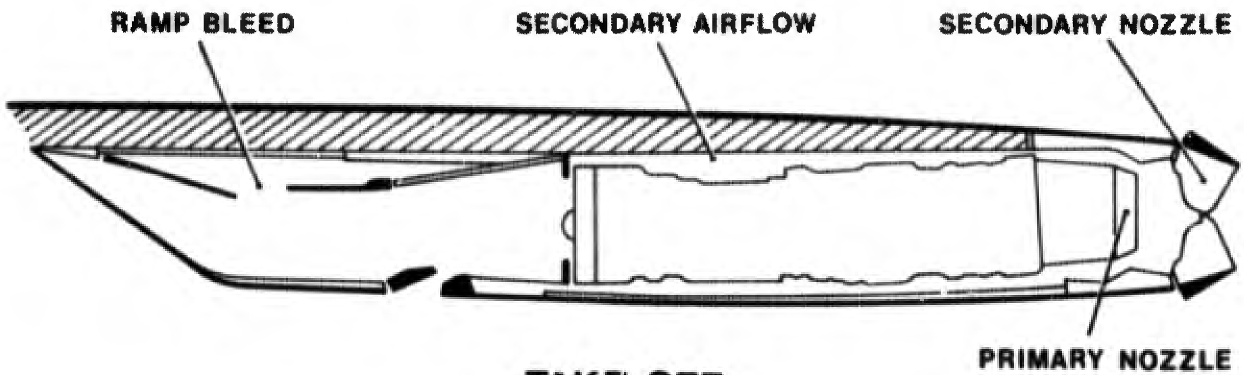
This not only stopped all academic argument over what was thrust or drag, but avoided the untidy (and perhaps careless) result that drag be a function of $^{n}/\sqrt{t}$ on top of all of its usual parameters. It ensured that the aircraft designer took a balanced interest in all parts of the propulsion system, since all penalties and alleviations were seen at once on our installed $^{T}/_{p}$ versus M and $^{n}/\sqrt{t}$ curves.

It was immediately apparent that it paid to reduce the spillage losses on our slender thrust margins by using a very undersized intake compared to previous practice. We took this to the point where the forward part of the internal intake was increasingly full of powerful shock waves down to zero forward speed at very high $^{n}/\sqrt{t}$, where thrust margins were unnecessarily high. These were intensified when the engine mass flow was increased after the intake design was frozen for manufacture. Appendix A summarises my method for matching intake and engine, and of improving the crude approximation that each 1% of intake loss loses 1½% of thrust.

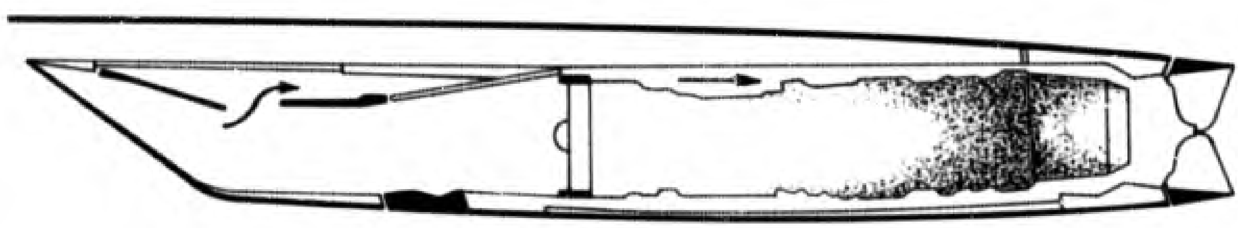
Fig. 18 shows the small losses at the all-important high subsonic/low supersonic climb and acceleration. Increasing shock losses are of course inevitable with increasing supersonic speed, and it is seen that the Lightning achieved close to the idealised engine brochure values up to the desired design M = 1.7. (We use engine $^{n}/\sqrt{t}$ control and variable geometry intakes to give better results at lower speeds today, or at higher speeds like the Concorde discussed later).

Fixed geometry was essential for installing early radar in the Lightning nose, and was also much simpler. The early prototypes for Mach numbers up to 1.4 or so had an even simpler pitot intake, with a small radar submerged in the lip. That was decided in 1948 as we have seen before the Russian nuclear threat. The Avon was then at its lowest ebb, with surge problems mentioned for Canberra, so we had changed to Armstrong Siddeley Sapphires for two prototypes.

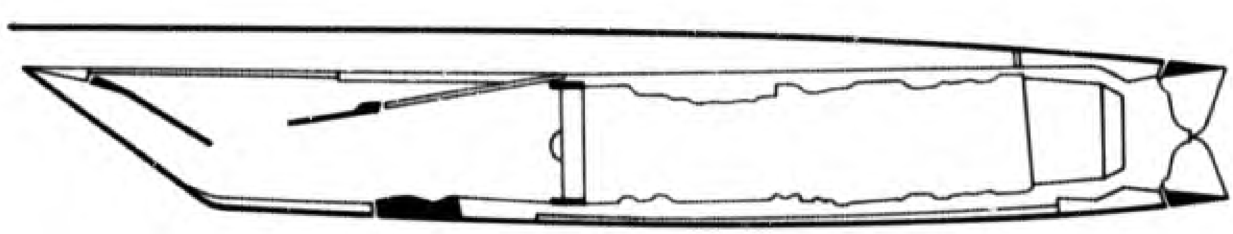
When we returned to the re-heated Avon with a variable nozzle, the only change to the whole aerodynamic configuration was the centre-body in the intake (apart from a larger fin for higher speeds and external missiles forward). This allowed a much better radar, and gave an oblique shock system for our new design emphasis towards M = 1.7 (later extended to M = 2).



TAKE-OFF



TRANSONIC ACCELERATION



CRUISE AT M = 2·05

FIG.19. SOME CONCORDE INTAKE AND NOZZLE POSITIONS

3. SUPERSONIC TRANSPORTS

Whilst preoccupied with this real Lightning and other supersonic experience from 1948, we worked only on the supersonic transport sidelines in the 1950s. We had already carried out our own studies, and appropriate engine technology seemed decades away. But the Supersonic Transport Committee soon involved the rest of the industry and RAE. We advised the latter against M wings and W wings in favour of slender shapes. Some were taking the latter too far for practical purposes perhaps, into completely integrated shapes. If one concentrates on $M = 2+$ operation, the design emphasis in Canberra on span is turned almost 90° onto length. But real operation is a compromise, as will be seen.

When BAC was formed in 1960, Sir George Edwards and Dr Russell pulled me into the study of the Bristol 198, which was moving towards the Concorde shape we know today. I was very fortunate to work for great structural designers in this period, since I was already working on the Canberra, Lightning and later military designs for Mr Page, now Chairman and Managing Director of all BAC aircraft.

Our Director asked us to talk today of the emergence of groups of senior people who respond with early estimates to a proposed requirement. The advent of jet aircraft, supersonic particularly, was finally making the task too much for the one-man efforts of much earlier days, discussed later. Greatly increased understanding of all aspects of aerodynamics and propulsion were required, but the "architects" find it a great relief to be able to turn to really experienced "constructors" to interpret the increasingly difficult requirements; and presumably vice versa. This analogy is far too simple of course. Thus, the aircraft architects must develop knowledge of all aspects of modern structure, systems and materials that affect costs and weights, to supplement their understanding of all sensible measures of the complete weapon or transport system effectiveness.

Propulsion matching was the problem for Concorde more than ever before. The desirability of a subsonic by-pass on a pure jet for $M > 2$ was seen from the start. But the technology budget for engine variable geometry still seems beyond everyone except the Russians. Even as developed for TSR2, the Olympus appeared to require at least 50% more thrust. I was withdrawn when the Anglo-French project of 1961 replaced the 1960 BAC study, so will touch on the matching of the present propulsion system, when the Olympus thrust has in fact increased around 50% in stages.

Intakes. Fig. 19 has been made self-explanatory for the Concorde intake. We saw how the need for fixed intakes forced all manner of careful aerodynamic compromises into the design of the Lightning. The inevitable penalty could not be afforded in the even more sensitive matching of the Olympus with Concorde. Variable geometry and controls just had to be accepted so that intake losses, spillage or other pre-entry nacelle drag losses could be largely things of the past. The large depth of the wing allowed the engines to be semi-submerged in nacelles, so that short ducts could eliminate most of the internal losses seen on the Lightning.

Nozzles seemed even more important than the intakes, with each 1% loss in gross thrust at $M = 2.05$ causing $2\frac{1}{2}$% loss in net thrust. Knowledge is still fairly ad hoc, so it is not surprising that the nozzle has been given further attention in recent times. We found on Lightning and other aircraft that jet

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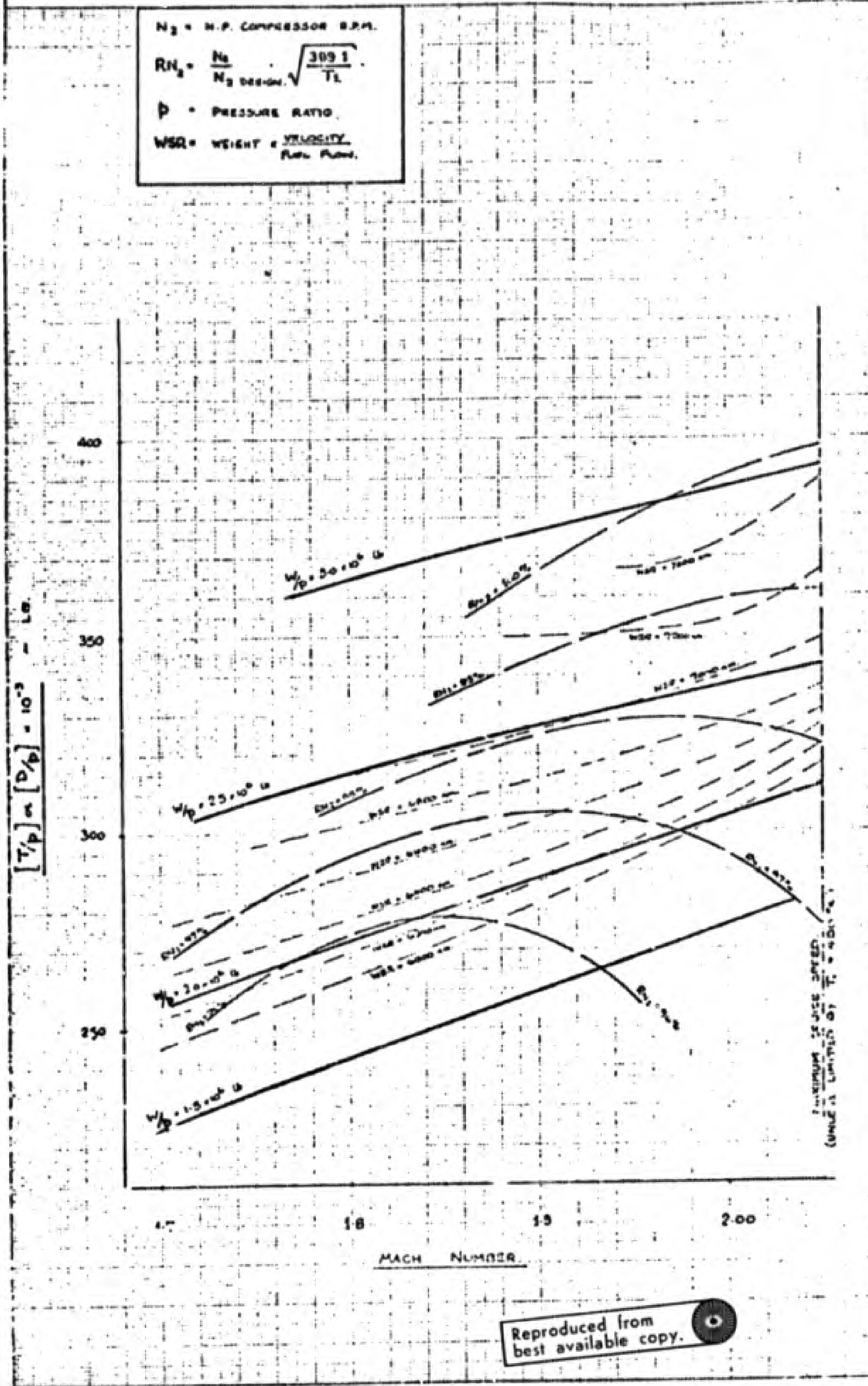
CONCORDE GENERALISED SUPERSONIC PERFORMANCE



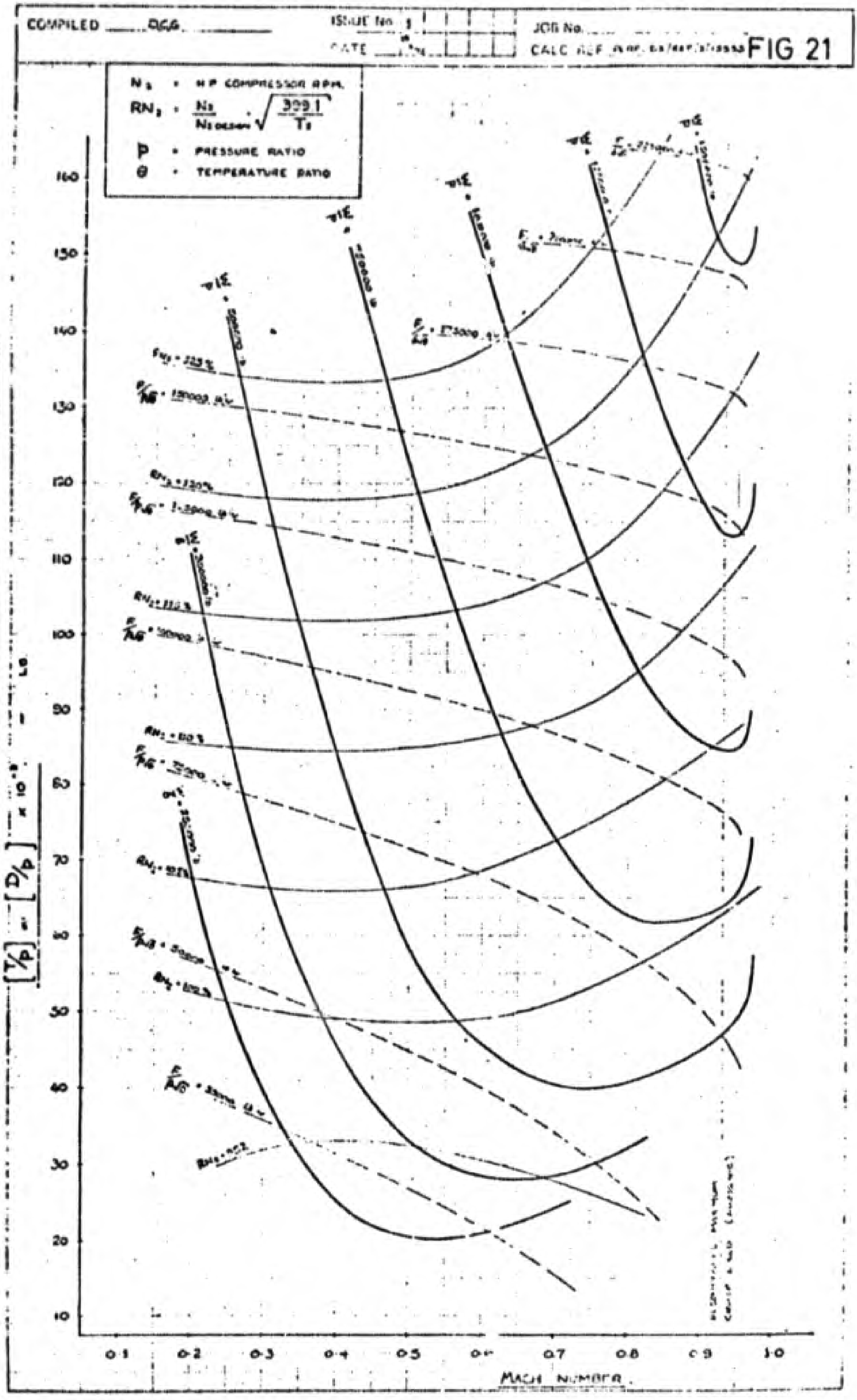
interactions with the flow over the after-body shape are very complex at high speeds. The after-body requires careful shaping, even for subsonic speeds, particularly in the "valley" between multiple nozzles. The importance and complexity of all this increases with speed, but effects are not all adverse. Favourable relief in after-body drag can occur from the powerful pressures feeding forward from a convergent nozzle - equivalent to up to 50% of the favourable effects from a convergent-divergent nozzle. The addition of a heavy con-di nozzle can sometimes give disappointing returns; even adverse at the lower pressure ratios. Ideal thrust is not achieved, and there is no reduction in boat-tail drag with full internal expansion, since this fixes the terminal shock at the end of the after-body.

The Olympus nozzle became increasingly complex, and Fig. 19 does not attempt to show all positions. It is varied even in normal flight without reheat. This allowed the aerodynamic "gearing" between the 2 engine spools to be tuned internally for each flight condition, respecting surge and other limits. It also allows lower jet velocity and noise after take-off by increasing low pressure rpm and airflow. (See also Appendix 2.)

Performance. Aircraft like the Lightning may fly only a few per cent of their missions at Mach numbers near 2. Nevertheless, this can be disproportionately important for fuel, propulsion, structure and equipment/heating. The Concorde is designed to fly 75% of its stage length at Mach numbers over 2, but the problem of getting up there (and back) quietly and efficiently is also important. Since most of the fuel reserves are for subsonic loiter and diversion, and many routes are restricted to subsonic overland (as expected from Lighting experience), more than 30-50% of the fuel is for subsonic flying. So there are many design matching problems similar to those we have been discussing, despite the great emphasis on cruising at Mach numbers over 2.

FIGURE 20 is kindly supplied by colleagues in BAC. It shows that the main objectives for cruise at Mach number above 2 have been achieved. An L/D of over 8 is measured up to $M = 1.9$, and is still well over $7\frac{1}{2}$ at the design $M = 2.05$. Since range is the main consideration in this region, fuel flows have been converted directly into specific range, as previously described. It will be seen that range is increasing with speed up to $M = 2.05$, and with altitude up to over W/p = 3 million lb. The specific range $\doteqdot$ 7500 miles is nearly 20% more than the Canberra, at nearly three times the speed. (Propulsive efficiency M/sfc increases almost as much, but L/D is more than halved).

Exactly the same cruise climb technique is employed, as fuel is burnt off to reduce W. Concorde currently climbs from 51,000 to 58,000 feet on a typical stage. It should be noted that maximum cruising thrust occurs slightly above RN_2 = 100% in these conditions. It can be realised that the control system for twin spools running between continuously varied intakes and nozzles has become pretty sophisticated, and I will postpone discussion of such systems until later.



CONCORDE GENERALISED SUBSONIC PERFORMANCE

$\left[{}^{T}\!/_{P}\right] = \left[{}^{P}\!/_{P}\right] \times 10^{-3} = LB$

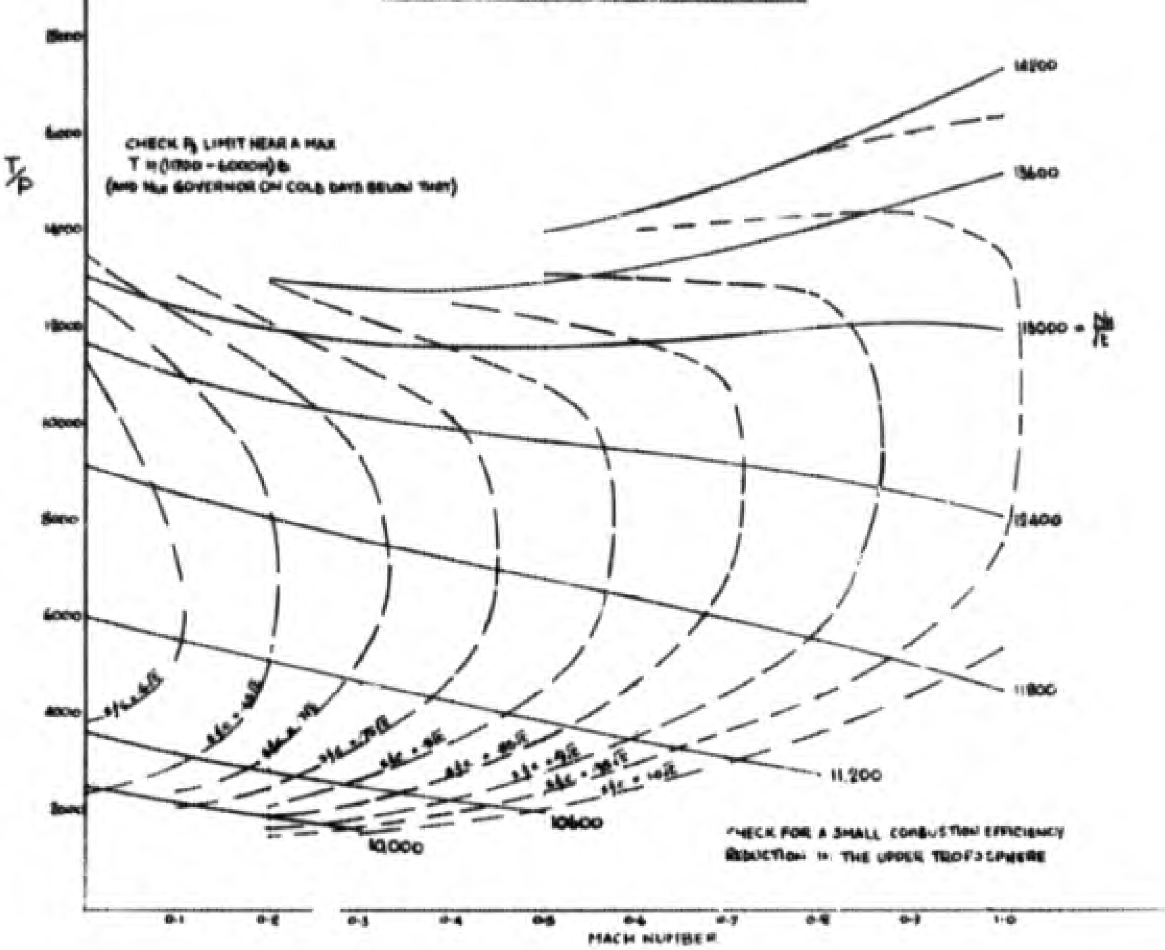
It is satisfying that the data still reduces to my standard form at such extreme conditions, to an accuracy of $\frac{1}{2}$% on fuel flow with most of the secondary corrections discussed later. The main reservation is that other cut-offs appear due to the greater rate of spillage thrust loss above ISA + 5°. It is interesting that this correction is still treated as a thrust loss.

FIGURE 21 similarly covers all subsonic cruise and loiter conditions, and is even more general. The fuel flows have been left in the original "f" form, so that the optimum or other range and endurance conditions can be seen at a glance, just as we did on Canberra. A specific endurance of around 9 hours occurs just below the minimum drag speed at all altitudes up to around W/p = 1 million lb., and then drops slowly at M = 0.9. This is due to compressibility drag effects on the low speed L/D of around $12\frac{1}{2}$. The specific endurance is less than two thirds of Canberra due to this, and also to poorer sfc at the respective speeds (although the sfc at the same M are similar). But the twin-spool Olympus does not have the equivalent of the swirl vane penalties of the Avon for low level loiter - very important for a civil aircraft.

The maximum range at each altitude again occurs at somewhat higher speed. The optimum occurs at around W/p = 1 million, down in the normal air traffic lanes in the troposphere. This specific range is within 10% of Canberra, due to the beneficial effect of a much higher cruising speed (M = 0.93) than Canberra. If required, the subsonic L/D could be somewhat better for a large tailless aircraft, and part of this can be found in the nozzle region discussed earlier. Rather more span and leading edge rounding would also help, further away from the original integrated M = 2+ emphasis. This would also help take-off and noise. But the largest improvement would still occur with the subsonic by-pass originally considered, which would further improve subsonic range and loiter particularly. But the best could never become the enemy of the good.

I have shown that novel matching areas for first and second generation supersonic transports fit into the methods previously discussed, so need not extend the Concorde curves or analysis further. Reheat is turned off soon after take-off for a climb at 400 knots CAS through the troposphere up to transonic speed. Climb acceleration to over 500 knots is then with reheat, since the effects of restricted span are still important. Although the previous rules show a flat optimum for turning off reheat at about M = 1.5, the system has been cleared to M = 2.05, to speed up flight testing and other purposes.

FIG 23 SPEY 163-25 MK 511 PERFORMANCE



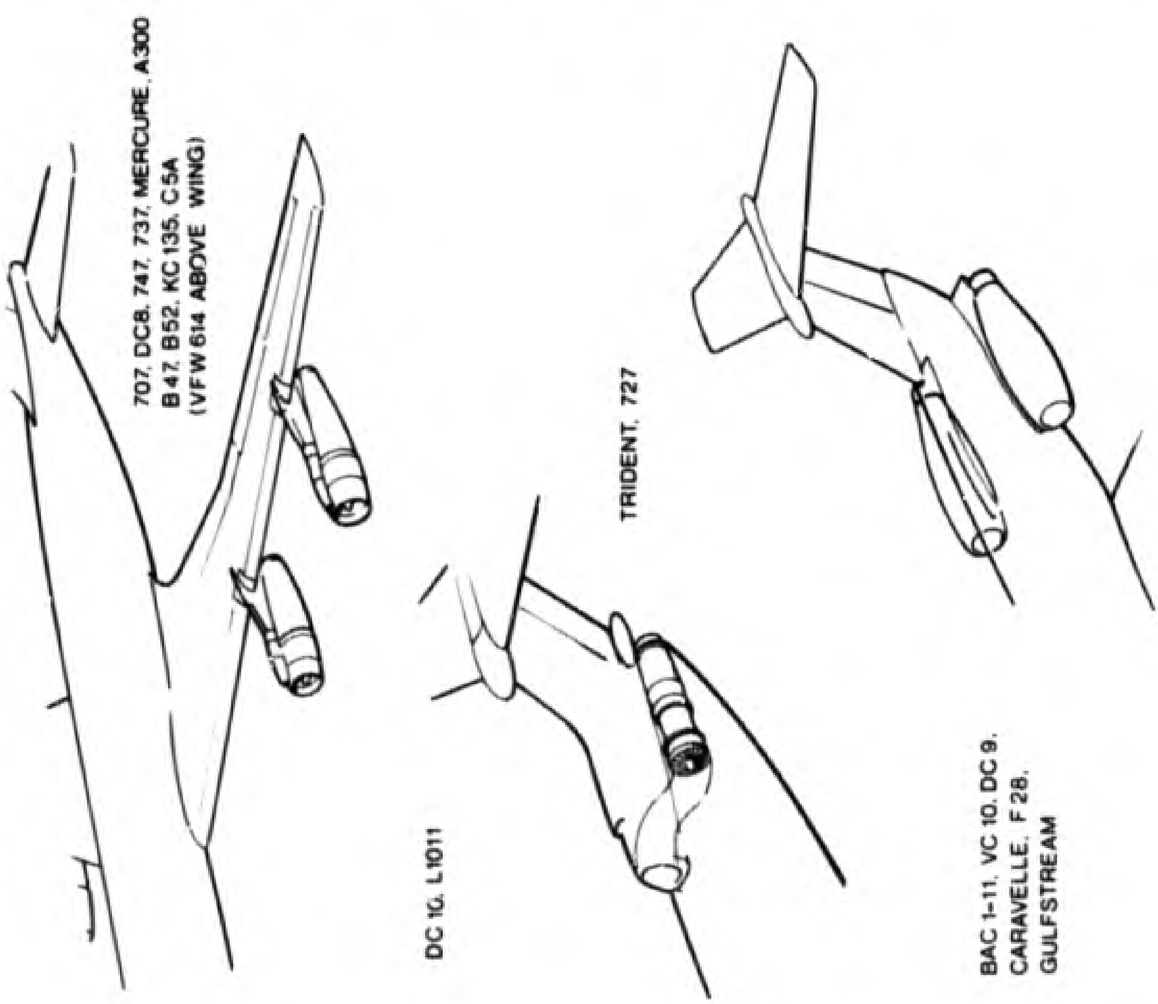


FIG. 22

4. SUBSONIC TRANSPORTS

As discussed already, the engine matching process is very similar to the Canberra, ignoring its extreme design matching points at very high and very low altitude. Optimum climb and descent techniques are similar, although speed at low altitudes may be more restricted by structural or rough air consideration. The rates may also be restricted by cabin pressure changes, easing engine climb ratings and air brake size. More attention has to be paid to matching take-off and landing, particularly with engine failure.

Some interest was shown by operators in converting the Canberra to civil use, including the more recent role of the Gulfstream II. But we were always too busy with military requirements.

Sweepback has shifted the drag diagram of current transports to slightly higher M. Wing chord can be reduced without the high altitude emphasis, leaving much higher aspect ratios and wing loadings. But the saving in Do/W is offset by nacelle drag, etc., so that the largest change is occurring in the propulsion diagram discussed below.

Engines are now installed in nacelles on all transports, apart from the early jet engined Comet and Tu104. The nacelles are carried on wing or rear fuselage pylons, sometimes with a third engine low in the fin. Fig. 22 summarises the pattern that has appeared to date. Boeing set the fashion under the wing with the military B47, B52 and the KC 135. The latter developed into the 707, and increasing by-pass has brought it to stay. Similarly, there has been little variation in cowling. All pure jets were completely cowled of course. Similarly for by-pass ratios up to 1 or so, with flow mixing internally in front of the nozzle, as on all British aircraft and engines, except RB211. Higher by-pass ratio fan air exits through an annular nozzle, normally from a short cowl at the front (the Convair 990 with the fan version of the J79 being an expedient).

Propulsion diagram. The Spey was the best engine available for the last civil project market studies by English Electric Aviation. This was before our merger into BAC, who later developed the fairly similar 1-11. Rolls Royce were persuaded to produce data in our matched non-dimensional form, long after the worst of the high-altitude test bed corrections discussed later. Fig. 23 shows an early diagram, which is still fairly similar to the Avon. Even though the Spey has by-pass from a multi-stage fan, the fall off in thrust with M is only slightly more noticeable. The two shafts help avoid swirl vane or other discontinuities at low rpm, but high engine pressures limit rpm at low altitude at forward speed or cold days.

Due to more efficient technology, sfc is reduced over 30% at low hold speeds etc., and about 25% at higher subsonic speeds. But maximum L/D is only about the same as Canberra, even in aircraft of two or three times the size, having two or three engines of about the Spey size. Although specific range should be more than Canberra due to much better M/sfc, it is often somewhat less in actual operation. This is because short haul operation has been occurring at excess speed, well below the cruise ceiling, with still lower L/D. It is doubtful if this can continue with increasing fuel prices and shortage (December 1973). Specific range is not only a measure of range, but is a direct measure of fuel use and costs.

Future Trend. The Specific Range of longer haul aircraft has long equalled propeller aircraft and now exceeds 10,000 nautical miles, operating near the tropopause. Part of this comes from improvement in L/D and M with much larger aircraft, as discussed on Fig. 13.

Journal R Ae S Jan 1945
(1943 - Oct 1944)

FIG. 24

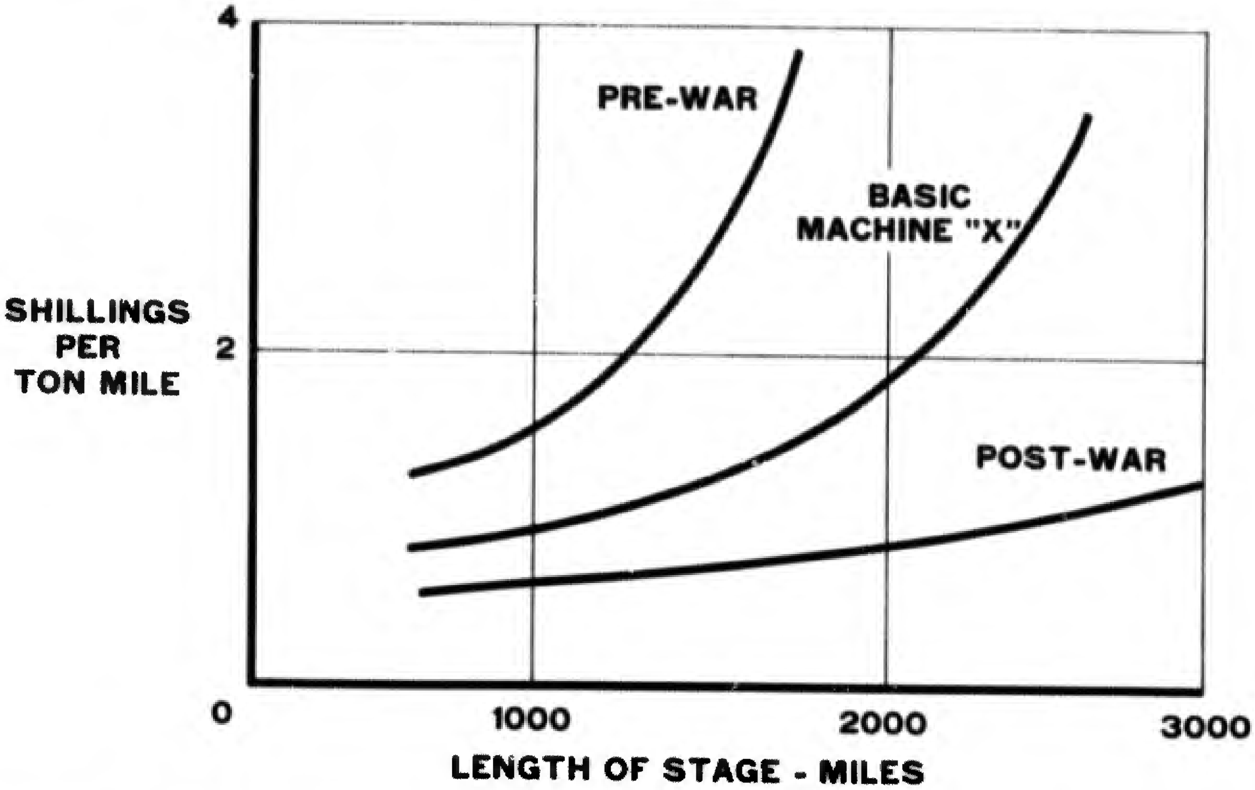


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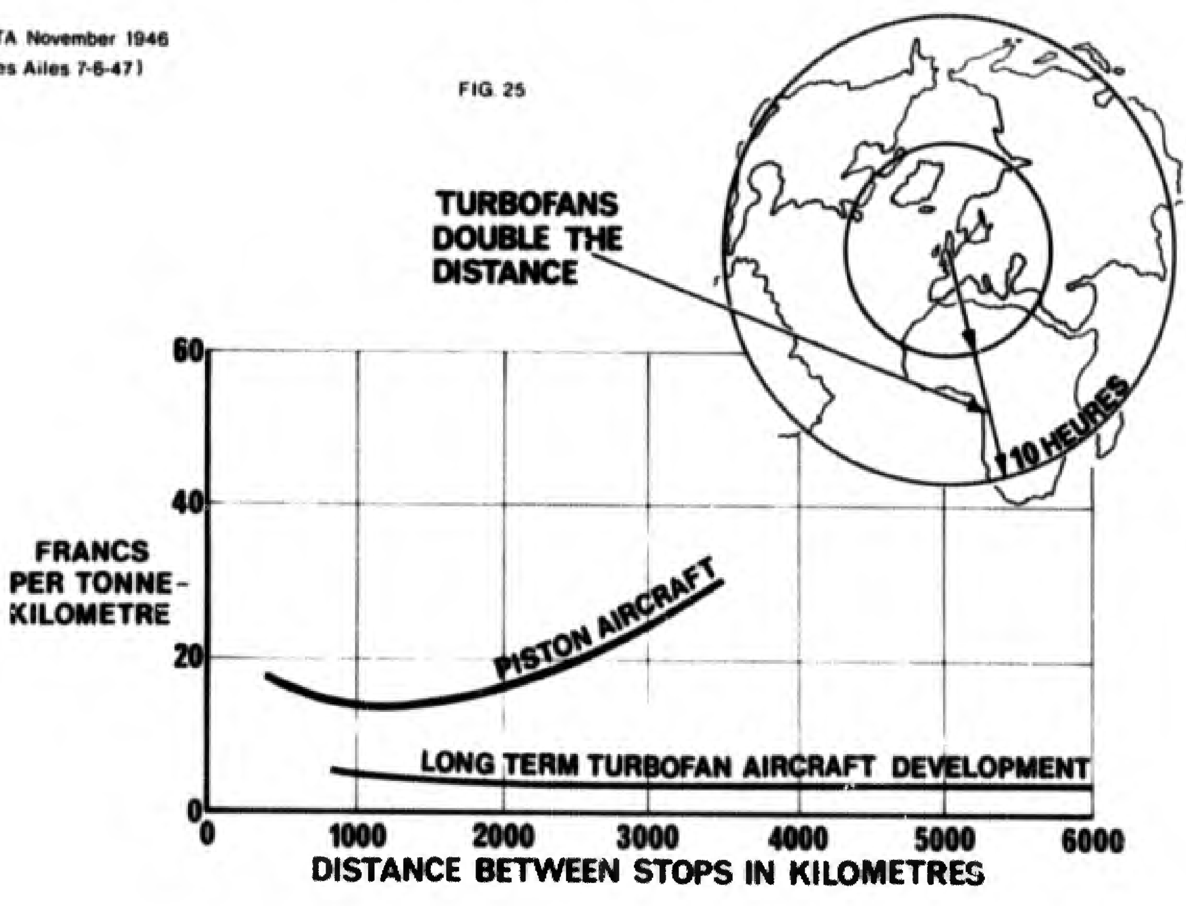
FIG. 25



But the much higher by-pass single-stage fan is improving sfc by 20% or more, partly offset by extra nacelles. The latest engines have by-pass ratios around 5 for low noise as well. This is now divergent from combat needs, except for the special AX mentioned earlier (cf Spey in Buccaneer, Nimrod and British F4). As discussed later, a common engine core will sometimes be possible. With a trend to even higher by-pass, Appendix 1 explains why fan gearing and variable pitch need consideration.

Subsonic transport design with turbo-fans is approaching a plateau analysed 30 years ago, discussion of which will lead into modern computer methods. The "one-man-band" designers of the time encouraged you to look well ahead:-

FIGURE 24 is from a study carried out in 1943/44 for R K Pierson, erstwhile Chief Designer of Vickers at the time. Parts of this study were disguised for a paper in the special edition of the R Ae S journal for January 1945. Costs were always our yardstick, but note the old-fashioned shilling and other units to show the large improvements over the pre-war designs that were still operating. Reference to turbine engines was prohibited by security, so Machine "X" was a decoy for developments like the Viscount we were studying, beyond the Viking we were already building.

But my real interest was in the even vaster improvements from the turbo-fan that we had to call "post-war". Most of the paper had to be devoted to the optimisation of aspect ratio, wing loading and so on to divert attention from the war-time revolution in propulsion. All the engine and aircraft calculations were single-handed, with advice on pre-war operating costs from an airline consultant. This study taught me more about the fundamentals than many of the computer studies of today, and it helped confirm the change of Vickers from its all-military history. But RKP's interest turned from turbofans, and I was tempted up to Preston on the promise of a civil version of the Canberra. The nearest I actually got to that was to write a paper that was given in Paris in November 1946 by Teddy Petter, Chief Designer of English Electric at the time.

FIGURE 25 is one of many from that paper. Security restriction on turbine engines had been partly lifted, so that the case for turbofans could be better revealed (it was partly reproduced some months later in the French technical journal). Note the metric units and unstable currency on that occasion. Operating costs and range were again used to measure transport efficiency, but the vast impact of speed could now be shown. To emphasise the backwardness of the pre-war piston engined aircraft of the time, Teddy Petter decided that we must go to Paris by train and boat!

To further simplify the understanding of my 1943 calculations, I converted all my engine results into efficiencies. It should be understood that, with a fuel of 10,599 CHU/lb.,

$$\text{Specific range} = 10{,}500 \times 1400/6080 \text{ etc.} = 2430\ {}^{L}/_{D} \times \text{Engine efficiency}$$
$$\text{where engine efficiency} = \text{True speed in knots}/(2430 \times \text{sfc})$$
$$= \text{Thermal} \times \text{Propulsive} \times \text{Mechanical efficiency}$$

The last term can be taken as nearly 1 for a gas turbine without gearing, so that all the real changes occur in the first two terms. The first term includes combustion efficiency, but this was already near 1, due to the excess of air to restrict turbine temperatures. Compressor and turbine aerodynamic inefficiencies had more complex effects on the cycle, making it necessary to increase the temperature to run the engine.

FIG. 26
AFITA NOVEMBER 1946

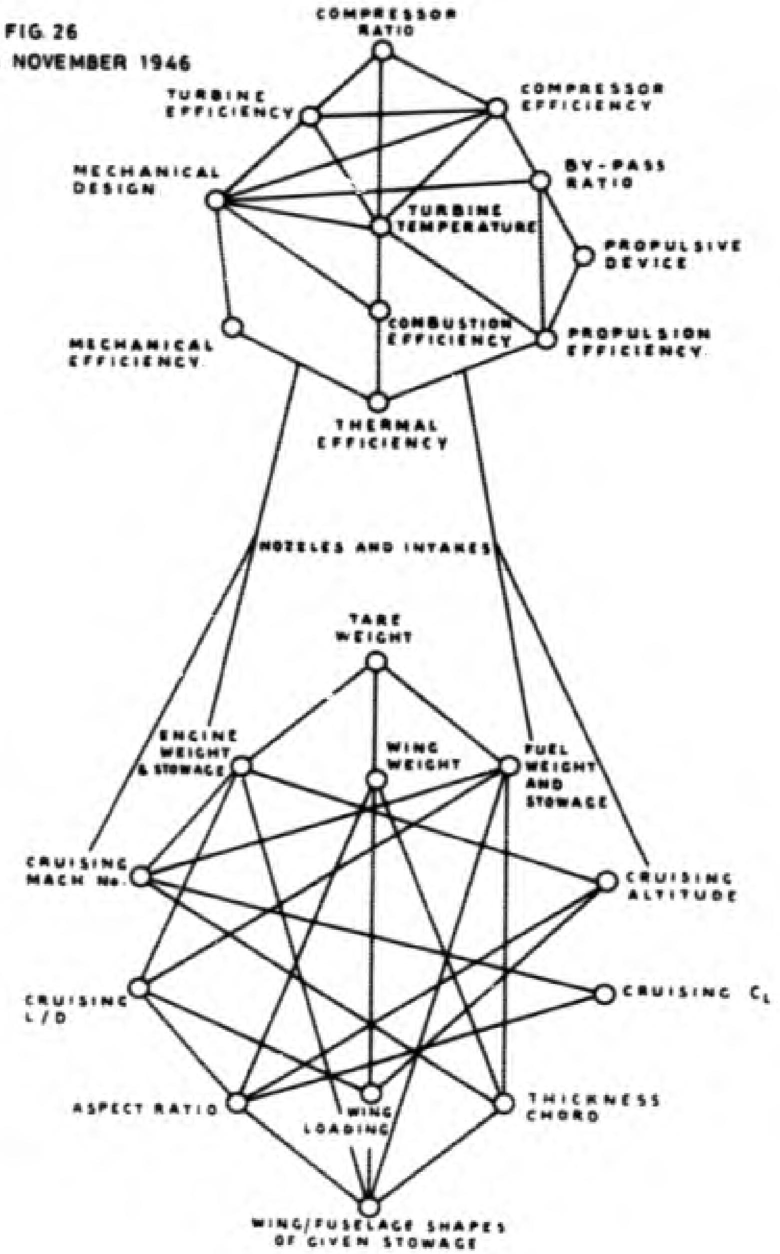
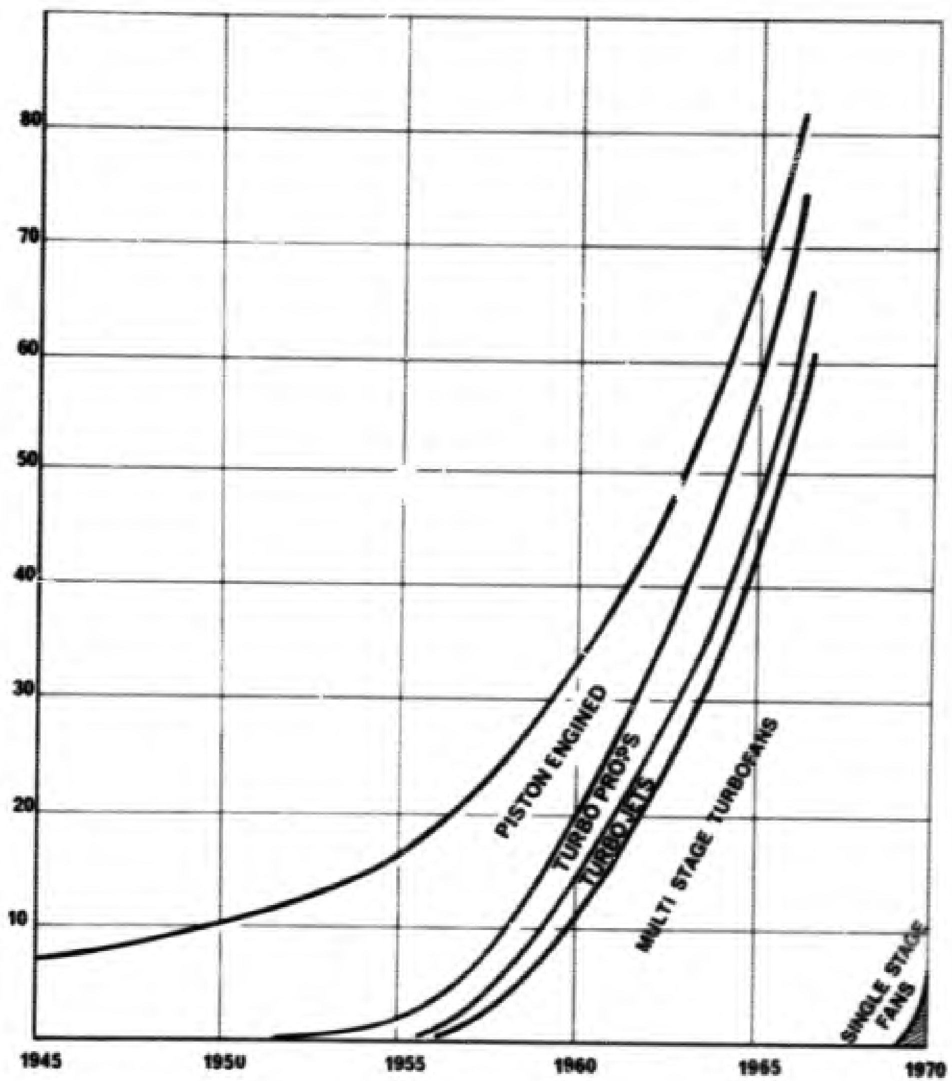


FIG. 27

SCHEDULED
AIRLINE
TONNE-KILOMETRES



So long as turbine temperatures were severely limited by materials, the propulsive efficiency of a pure jet was not too bad. Increased temperatures allowed higher pressure ratios and thermal efficiencies, but propulsive efficiency fell away badly due to increased jet velocity. Although the resulting thrust per weight or volume was nearer the right way ahead for the high performance military jet, much lower jet velocities were desirable for noise reduction and transport propulsive efficiency.

The geared propeller-turbine had already gone ahead for the latter purpose. This could be regarded as an unducted fan of very high by-pass ratio, suitable for operators in the Viscount and F27 role. Due to company connections with Napiers, we were also obliged to discuss clumsy piston gas generators and heat exchangers of the 1946 era. Although these might yet be interesting, my aircraft and engine matching diagrams showed the overriding importance of installed aerodynamics at high speeds. Fig. 26 marked some interactions that were taken into account.

The early way ahead at high subsonic speeds was seen to be the intermediate ducted fan, and the market is now proving this true (Fig. 27). This diagram is taken from my review of the civil and military aircraft markets in the R K Pierson memorial lecture. This was on the occasion of the 50th anniversary of the pioneering Atlantic and Australian flights of his Vimy design. Compared to the Avon, engine efficiency has nearly doubled, but L/D is often little more than Canberra at larger size, despite the simpler design point.

FIGURE 26 showed many engine matching interactions with overall aircraft design, as were also considered for the Canberra. I never regretted my early understanding of the engine man's problems. Glimpses of this continued into the recent rapid growth in obscurity of computerised engine data. Much of the rest of this lecture is concerned with attempts to restore some mutual understanding of the complete problem.

5. COMPUTER OPTIMISATION

These examples illustrate what can be done with a single slide-rule. Even in those earliest days, it was possible to predict some trends we see 30 years later. But subsonic transport design matching is relatively simple, and transport requirements are fairly clear. Boeings were developing a computer synthesis program for subsonic transports some years ago, and RAE etc. more recently. Full support from the top is essential for general acceptance.

We have been edging likewise on high performance aircraft at the other extreme over many years. I was rather pleased that the Director is shouldering the task of how operational analysis is used to formulate requirements, as it used to be felt that this should be left mainly to government employees! Since parts of this are so integrated with design, this takes up increasing time on our computers. Oh for a measure of operational effectiveness as simple as transport speed and passenger miles! I will be content today with illustrating aircraft design trade-offs in relation to total costs, which is becoming accepted as part of the collaboration that industry should undertake.

Even this can be very complicated compared to the subsonic transport that operates on a relatively simple flight plan, often regarded almost as a towed glider. (The only recognition that the propulsion pods are nearby is in structural "relief" and aerodynamic interference factors, variations in which are not fully accounted in some computer programs.)

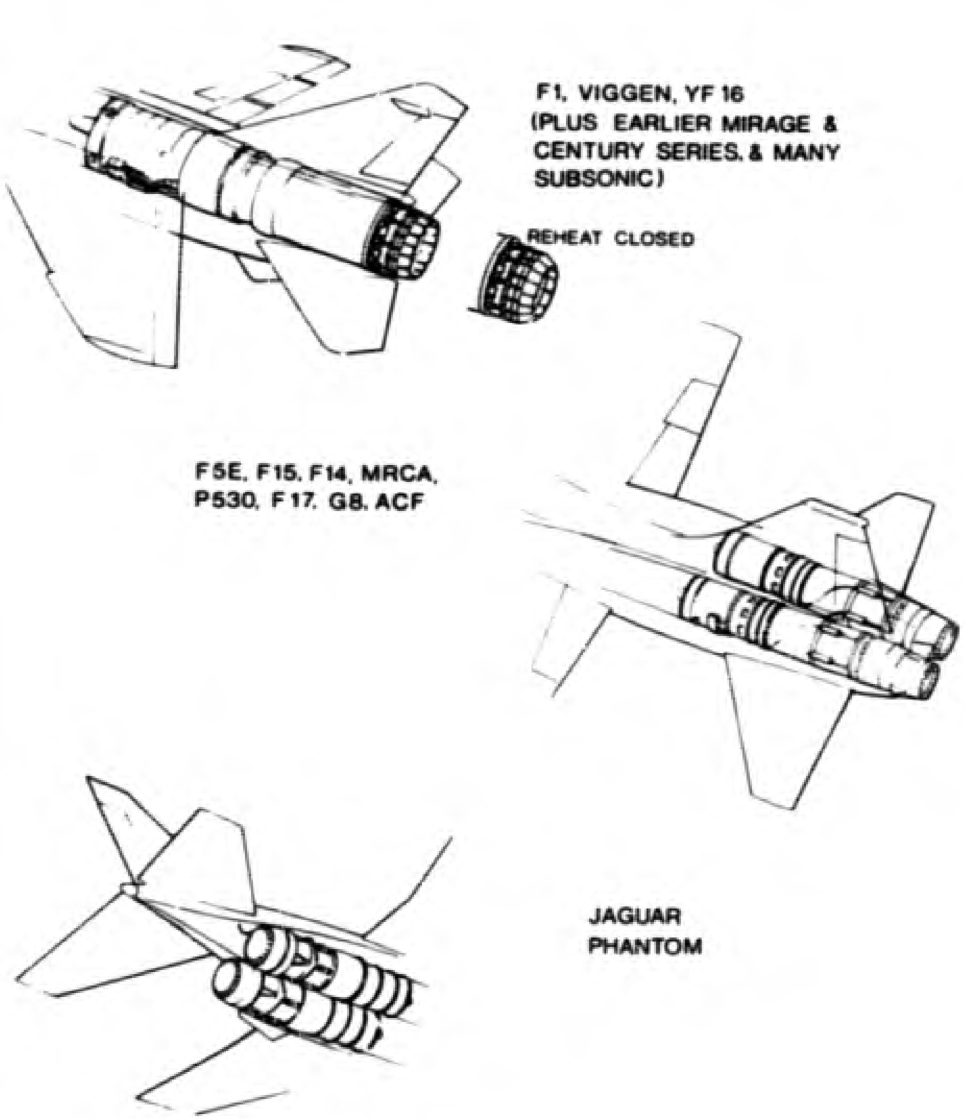


FIG. 28 ENGINE INSTALLATIONS

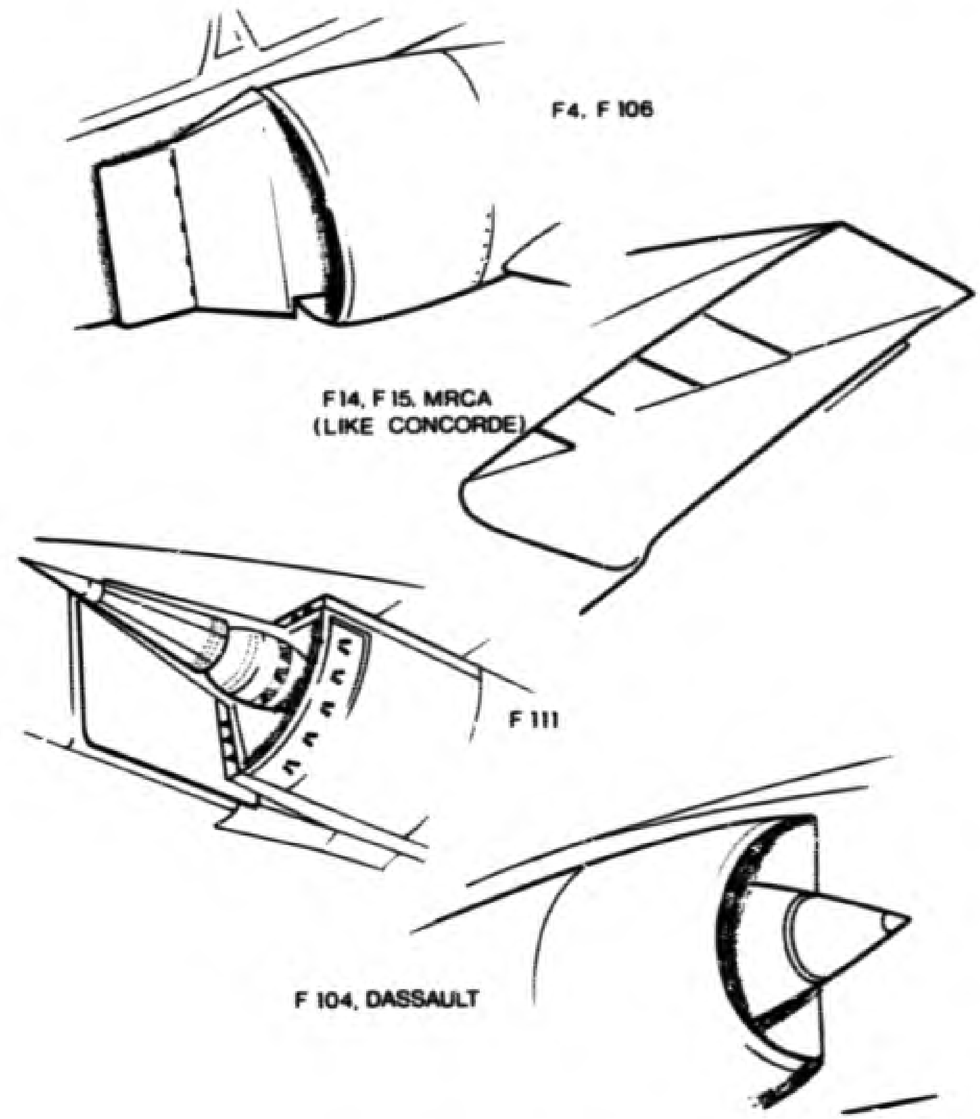


FIG. 29 SIDE INTAKES

The general case can have a more integrated propulsion system, so it is first important to understand all the interactions before discussing computer processes.

Engine installations

Discussion here must be very short and simplified. Aircraft with transonic or supersonic performance have far the most difficult range of Mach number design considerations, including the critical sensitivity of wave drag to integrated shaping.

Large supersonic aircraft like Concorde and B1 have the nacelles semi-integrated as we have seen. Fig. 28 shows that smaller supersonic aircraft now have the engines integrated into the fuselage, in simpler ways than Lightning. They rely on a lot of reheat, with fuel in the fuselage as well as the wings. Structural cross-sections near the nose are usually dictated by equipment and crew; further back by intake ducting, fuel tankage and undercarriage. We deal first with the engine installations that dictate the cross-sections at the rear.

The simplest is of course the single-engine, which can approach a circular body of revolution towards the rear. Around half the smaller aircraft were of this type at one time, but the increasing cost of supersonic aircraft has put more emphasis on engine-out considerations.

Twin engines are now all side by side, unlike the Lightning. The Phantom and Jaguar at the bottom are unique in having the engines tilted downwards forward of the tail surfaces (the older F101 and Buccaneer were nearer to short nacelles on the side).

The largest number of new aircraft (centre) have engines side by side at the rear. Most are fairly closely spaced, but the F14 has them so widely spread that they appear almost as nacelles on a horizontal pylon towards the rear (but not for reasons like the original weapon delivery system of the A3J).

Since all installations share the problem of large reheat nozzle variation, this can be the source of base/afterbody drag that helps increase low speed C_{D_o} to more than 0.003 based on surface area. Some of the more complicated interactions at higher speeds were discussed on Concorde and again below.

Intakes (Fig. 29)

Only very early aircraft like Lightning now have the intake right in the nose. The F8/A7 and YF16 have them rounded underneath the nose, and the F101 and 105 had them in the wing root. The great majority now have them on the fuselage side, which will be taken as typical to illustrate further variations.

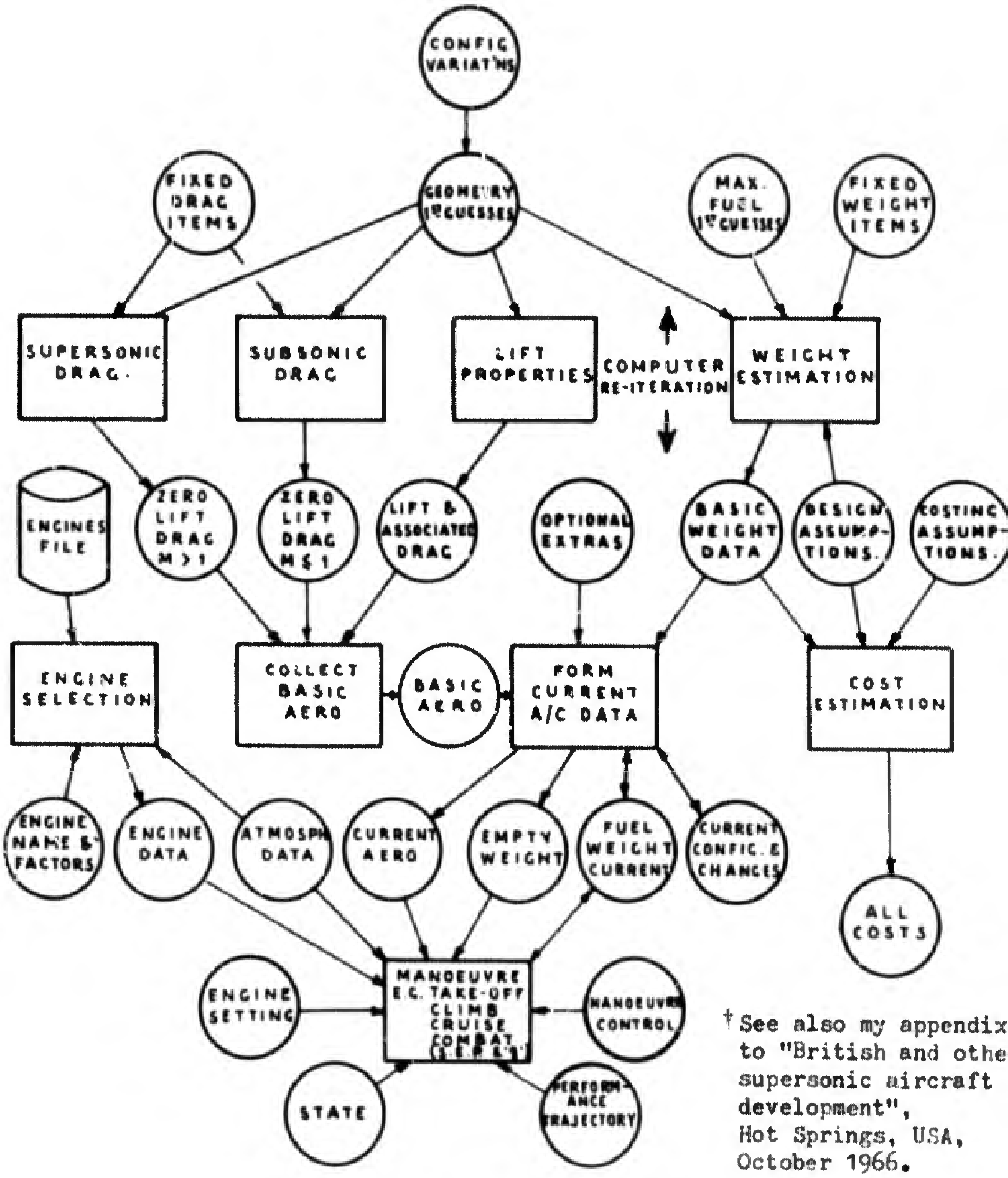
Most recent supersonic aircraft do in fact have variable intakes, apart from the simple pitot intakes on the Swedish fighters and Jaguar, with blow-in doors for take-off like Lightning. Otherwise, one has to go back to the F104 for a fixed intake on the side. This was semi-circular like most of the Dassault family. The latter have the bullet moving fore and aft "like a mouse from its hole." This F1-11 wing cuts this further to a quarter-cone.

Most other side intakes have flatter ramps. The Concorde has already been illustrated, and several of the very latest military aircraft are similar. The ramp was effectively turned through 90° on the Phantom and F106. All systems have numerous pros and cons that can be argued interminably.

As expected in 1948, intakes aft of the nose have boundary layer penalties that increase with speed. Bleeds or diverter plates are used in most cases to reduce this, but this can be the other source for increasing C_{D_o} above 0.003. These and other interactions can be even more complex than Lightning at higher speeds.

FIG. 30. COMPUTER INTEGRATION - PROJECT STAGE.

FOR CLARITY SEVERAL DATA BLOCKS & PROGRAM MODULES HAVE BEEN OMITTED OR GROUPED UNDER A SINGLE HEADING (E.G. GEOMETRY)



† See also my appendix to "British and other supersonic aircraft development", Hot Springs, USA, October 1966.

This led to parametric wind-tunnel testing of the kind used later on Concorde. We have had as many as six tunnels in operation at Warton, plus many realistic flight and other sophisticated simulations. These became accepted tools of the trade before the digital computer. We had the first of these around 1950, before anyone we know in the business. Equally important is a continuous flow of new aircraft to keep us in line with realities.

Since propulsion systems have such large interaction problems with the aircraft, it is important to maintain a powerful design team throughout, like our friends at Dassault. But having established firm "baseline" drawings, the computer can be a relentless and impartial inquisitor as to whether any combination of small changes gives better results. If it does, a new baseline design is established to start again. It is essential to find the best design before anything is built in this expensive age, so let us just mention one of these tools.

Computer Integration

FIGURE 30 grossly simplifies the parts that the computer can play in integrating our human and other experience. Each of the modules represents life-times of experience in specialist areas, rationalised into empirical laws or graphs and tables of relationships, or fairly exact theories (e.g. wave drag integration for slender shapes). Cost output on the right has been extended from R & D and procurement into all life cycle operating costs, based on our wide experience. Computerised empiricisms can be used for weight changes until more and more structural elements are optimised and analysed. A recent MOD survey called ours "the most advanced European aerospace capability".

Typical modes of operation in the project stages include:-

1. All cost, performance and effectiveness changes due to small variations in wing, body and engine. This is simple but often fruitless.
2. Cost optimisation of possible variations, maintaining specified aspects of performance or effectiveness eg scaling new engines and keeping range constant with small pockets of extra fuel.
3. All benefits from variations (including fleet size) within specified cost limits.

To ease the organisation of such tasks in English, Bill Crookes* and his people developed a program language called PROTRAN. Someone improving the cost modules for example, need not understand the workings of the other modules, but is allowed full say in all the design interactions affecting his responsibility. Results appear in tables of data and graphs as required.

Considerable effort has gone into developing this system and keeping it up to date. It can saturate you with results if not planned in conjunction with graphical and other understanding. A big advantage is that the computer never tires of being up-dated from the experience in all quarters of the organisation; or of including all interactions, however small they might seem until particular situations arise. Growth factors are very different to what were once believed, because all interactions are fully included and vary through the life cycle.†

One limitation has been dependence on data for particular engine families. Due to the problems of extending this to other engine parameters or even installations (Appendix 2), MOD kindly gave us their engine synthesis program to integrate into our system. Although not as elaborate as some now used by engine companies, MOD insist that it is good enough for the complete matching problem. We look forward to increasing collaboration to ensure consistency of overall conclusions on future engines.

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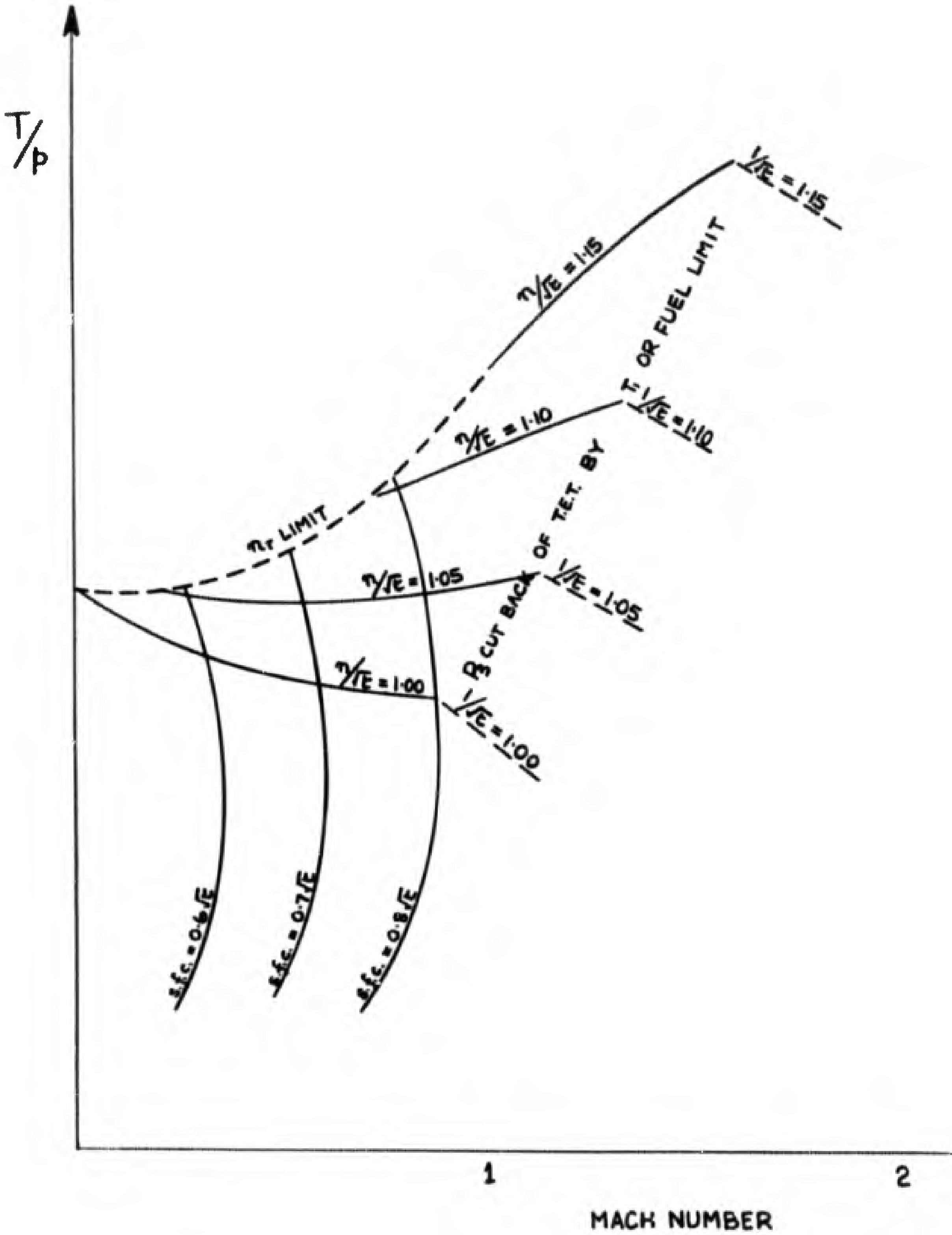
*To round off on organisational trends, Bill is an experienced user of the Ae 21 methods. My other seniors had longer experience almost from the start of our jet projects, plus recent civil at Boeings. Throughout an organisation, across to the final Programme Directors, it is important not to abdicate basic understanding to people who tend computers without a lot of other recent experience.

Fig 31 MAX. DRY THRUST & s.f.c. OF POSSIBLE FUTURE TURBOFAN



6. FUTURE MILITARY TURBOFAN MATCHING

Over a hundred times more jet engines have been produced than military turbofans to date. Many of them had single shafts, but even this will now be typical only in SNECMA. Military data on future turbofans is classified, and very complex, but the appended methods give some warning of new possibilities and problems that must be faced.

All new engines have by-pass ratios up to 1 or so. Even those nearest to a pure jet have a "leaky" airflow to cool the nozzle systems, equivalent to a by-pass ratio of around 0.2.

It is a fallacy that the thrust of a by-pass engine always falls off more rapidly with aircraft speed, due to the lower average jet velocity. This was true for the earlier by-pass engines that were governed near constant rpm. It can still be true for the military fan without reheat at low altitudes as on Fig. 31. We will see later that this is where rpm are kept approximately constant by the control system, or even brought down at the higher Mach numbers.

With the trend towards two or even three shaft engines, etc., rpm is no longer as accurate a measure of temperatures in the hot end of the engine as it was. The high pressure compressor/turbine rpm usually gives some measure, and is often retained as a pilot instrument (the larger units on the LP shaft run at lower speed with an aerodynamic "gearing" that increases with $^{n}/\sqrt{t}$, as discussed in Appendix 1.) Most civil engines measure Engine Pressure ratio, which we found on Canberra to give a good measure of gross thrust.

The prime instrument on most British military engines is now some measure of TET, which has always been a major factor in the predictable life of the engine. n^2 was usually the most reliable guide to TET on early engines, and is the reason why all engine performance was in terms of n. From now on, we should really replace n by the more significant $\sqrt{TET}$, where TET is the measured TET divided by its maximum value.

Engine control systems

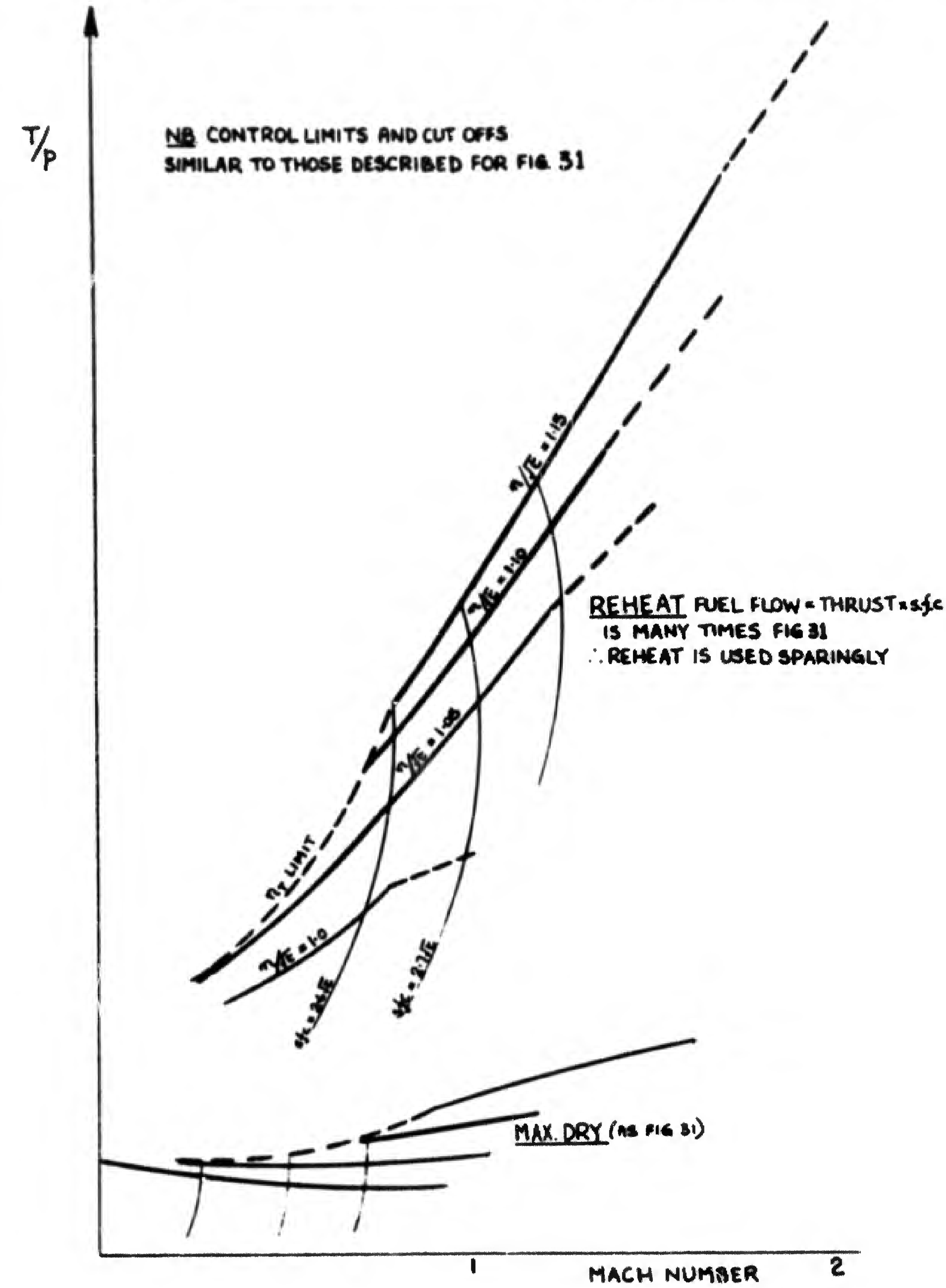
With the availability of these and other measurements, the relation between the pilot's throttle and the fuel control system are increasingly complex, even via miniature computers. Consider some typical limits for engine safety or life reasons:-

(1) Over a broad range of subsonic speeds/low altitude and supersonic speeds/stratosphere, Fig. 31 (and 32) shows we still have the old types of n (or TET) limits and laws. Since maximum continuous n is still around .95, and desirably lower for engine life, this still defines over most of the usable flight range when the engine is throttled back.

(2) The cut-off at lower speeds can be at constant $^{n}T = {}^{n}/\sqrt{T_1}$. This will usually be matched to allow maximum n (or TET) down to a T_1 somewhat below ISA static, i.e. it will not operate at all in ISA conditions below an altitude of several thousand feet, but will then cut-back from increasing speeds, up to around $M = 1$ in the stratosphere. Since $T_1 = 288\ t\ (1 + 0.2M^2)$, this speed is increased on a cold day, and it can cut-back at sea level static on a very cold day.

Thrust curves merge towards a single curve with idealised ^{n}T control. More recently, TET may be cut back as a function of T_1 instead of n. If it were cut back proportionately, it would again approximate constant ^{n}T. It may not be cut back quite so rapidly in practice, so that the $^{T}/_{p}$ curves do not quite merge.

Fig 32 MAX REHEAT THRUST & sfc OF POSSIBLE FUTURE TURBO-FAN



This thrust loss at low speed at altitude is rarely important, particularly on cold days. Intake air flow is also cut back, which would otherwise embarrass intakes that are now more sensitive than Lightning. Even more critical conditions would occur in the front stages of the compressor or fan, including surge. I've talked of cut-backs for simplicity of description. If one did not have the cut-back, one would have to re-match the engine to avoid surge at low speeds/high altitude on a cold day, so that ^{n}T would be much lower at more normal flight conditions. It is fairer to regard such controls as a method of getting more efficient use of the aerodynamic performance of the fan compressor at normal flight speeds.

(3) The cut-off at much higher speeds is for engine weight and life reasons, limiting pressures as well as temperatures and rpm. If properly selected, it can help the pilot reduce aircraft design speeds. If set for example to start at the T_1 corresponding to high subsonic speeds at sea level, this M will rise towards about 1.7 in the stratosphere, or slightly more on cold days. If TET or n is reduced as a function of T_1, our old engine laws still apply. But it is necessary to drop out n or TET from the labelling of this part of the curves (since they are no longer at maximum) and leave $1/\sqrt{t}$ to interpolate this thrust limit for any atmospheric condition.

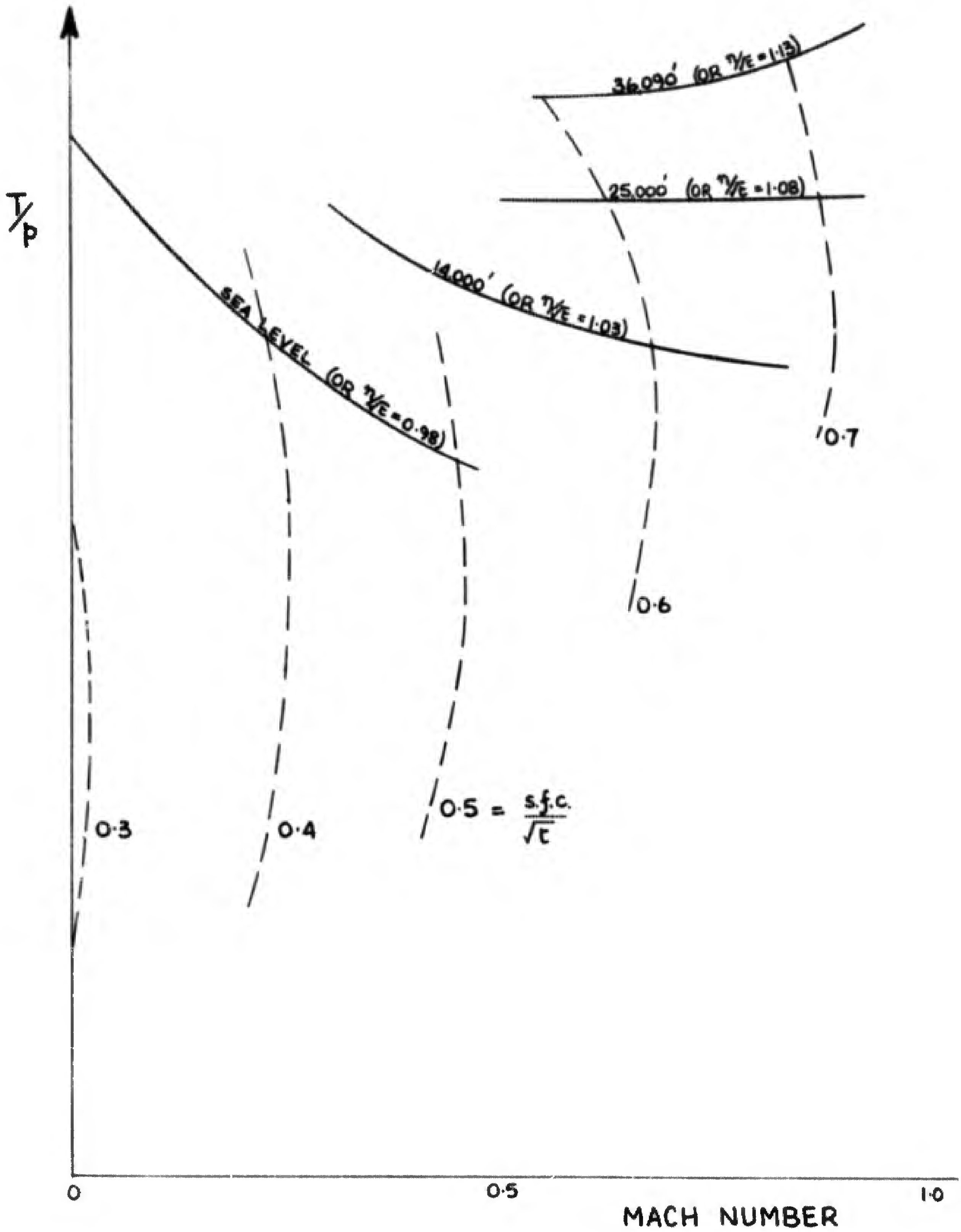
(3A) An alternative cut-off at high speeds is by limiting the engine fuel flow as some function of altitude, and possibly speed. This has the performance advantage of greatly reducing this old $1/\sqrt{t}$ interpolation for decrease in thrust on a hot day. Thrust $\propto 1/\sqrt{t}$ in such case at the higher speeds (since sfc $\propto \sqrt{t}$). Such control has the disadvantage that it can leave little or no speed range where maximum TET and n can be used on cold days. Vice versa, it increases the range of speeds over which the pilot can use maximum TET still further on hot days. A pilot using the throttles in what he believes to be the same way can greatly reduce turbine creep life in a hotter climate. How many people on the ground are fully certain of such conclusions without hours of thought and computer running?

(3B) The importance of engine life and operating costs is becoming even more important versus performance in different climates. Now that the control possibilities are so great, we can expect even more sophisticated laws. In order to exploit these, or even fully comprehend the significance of what we already have, we require non-dimensional or other simplified graphical print-outs of key features in the mass of computer data we now have. The Appendices suggest that the scatter on such plots (unlike some real test data) will be due mainly to reasonably predictable corrections for specific heats and to more dubious corrections for "Reynolds number". It would be a pity if the latter crowded out most understanding of the main engine laws and possibilities, now that most critical combat operations are concentrating near sea level (and economic transport operations around the tropopause).

CONCLUSION

If engine/aircraft matching becomes ever more complicated, it is because the possibilities for improving efficient performance or engine life are ever increasing. This challenges us all to understand what we are doing, before abdicating too much responsibility to the computer operator. This will encourage "judgment, innovation and originality", as requested by the Director. Correspondence on this paper will be welcomed, particularly from engine people.

Fig 33 THRUST & s.f.c. OF A FUTURE CIVIL TURBO-FAN



POSTSCRIPT ON OMISSIONS

With time limited, I was sorry to end on the control matching problems of future military turbofans. This is essential for consolidating aircraft development, and for matching common engine cores. Any understanding would have been far more complicated without my simple methods in the Appendices, and almost impossible to discover in recent computer programs.

I regret that time does not allow discussion of the even more complex V/STOL and RPVs to which our computer systems have been extended. Only take-off and landing matching is often much affected, but V/STOL particularly can lead into endless discussion of alternative propulsion configurations. Many of these have been flown, but only one has gone into production so far. We have studied endless permutations even on the Pegasus configuration. V/STOL requirements inevitably lead to increased engine size and fuel, and the overall matching to minimise such penalties is even more complex. Computer matching with millions of fairly inexplicable numbers can be very dangerous without easier engine understanding.

If rocket boost is used for RPV take-off at the other extreme, this is the simplest propulsion problem of all. Often carried and jettisoned as a complete package, it usually supplies constant net thrust only for a relatively short time. This is because sfc = 3600/SI is rarely less than 15 lb/lb hr., except with the hydrogen fuel in space rockets. This reminds me to mention our very early use of the computer to optimise space Shuttle and other systems, and our studies of hydrogen or other fuel ever since it was flown in a B57. (AAS70-059). An impetus for studying other fuels is more obvious in an oil crisis (December 1973). We must meanwhile put back the emphasis on fuel economy discussed under Lightning reheat and transports.

FUTURE CIVIL TURBOFAN MATCHING

To end on this note, examine the characteristics of new turbofans, with by-pass ratio over 5 as already discussed. Fig. 33 shows that these have very much simpler control patterns than the military turbofan. In fact, most of the latest civil engines are now flat-rated up to around ISA + 10°C by pressure and other fuel controls. So that if T/p is plotted for ISA + 10°C, this represents actual thrust versus altitude in temperate conditions (thrust may even fall away slightly at low temperatures).

The curves are labelled with $n/\sqrt{t}$ only for T/p in hotter conditions, and to give TET in throttled conditions. One should really replace n^2 by TET again, since all such engines are 2 or 3-spool. It will be seen that the thrust of such engines usually falls away with M, except at high subsonic speeds near the tropopause. This is because thrust is usually sufficient in such conditions, and the fan is more important for a better and quieter take-off and climb.

But sfc is improved 20% or so for cruise, as already discussed. It will be seen that fuel consumption is even more greatly improved for holding and other low speed operation. Such engines, including the RB 211, are already flying in the latest American aircraft. It is regrettable that this is still not the case in British aircraft.

APPENDIX 1

Intake - Engine - Nozzle matching

As promised, I struggled to avoid gas-turbine theory in the main text, even of the simplest kind. It is still desirable to have a little practical understanding for many purposes.

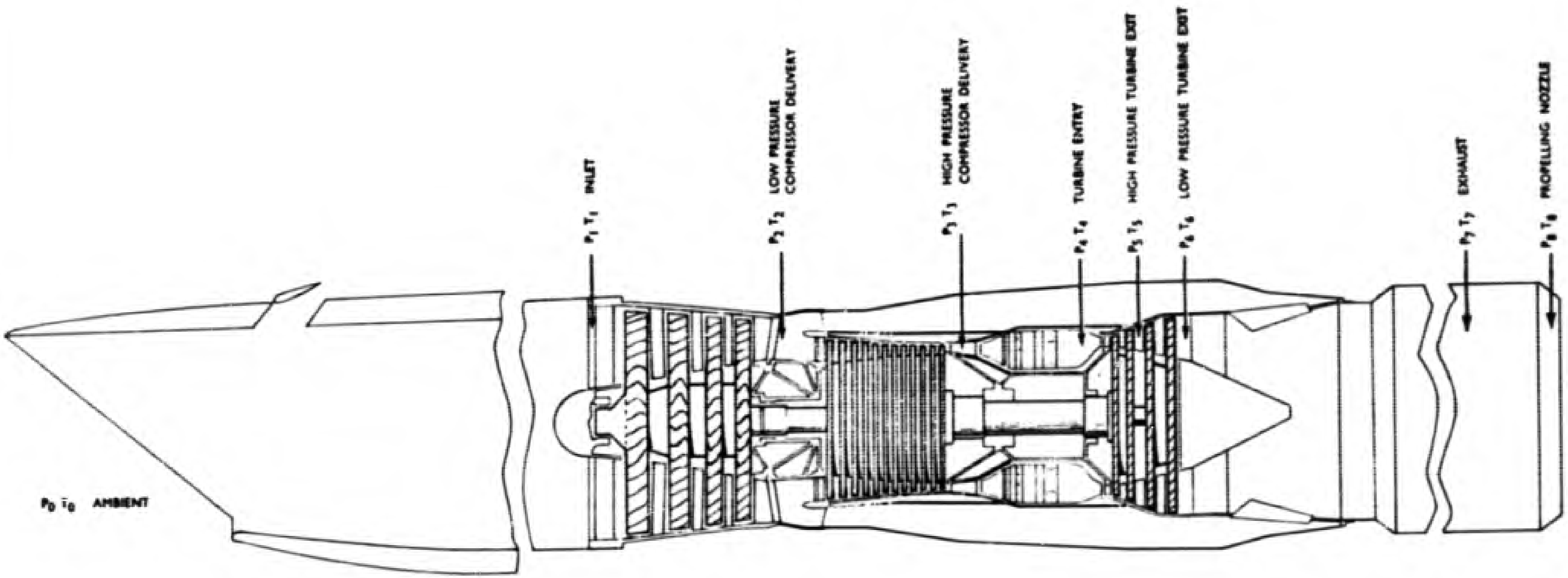


Fig A Engine Pressure and Temperature Definitions

APPENDIX 1

Intake - Engine - Nozzle matching

Ever since the early 1940s, it has been important to understand how various aircraft intake factors should be matched to the engine. It is not just a matter of ensuring that the airflow in any one-sixth of the intake is not too different to the average of the rest. Most engine companies now have numerical criteria even for that. Average intake pressure recovery characteristics are critical to engine performance, and are increasingly critical functions of aircraft speed, intake size, etc. (Figure 18 showed the importance of η_I):-

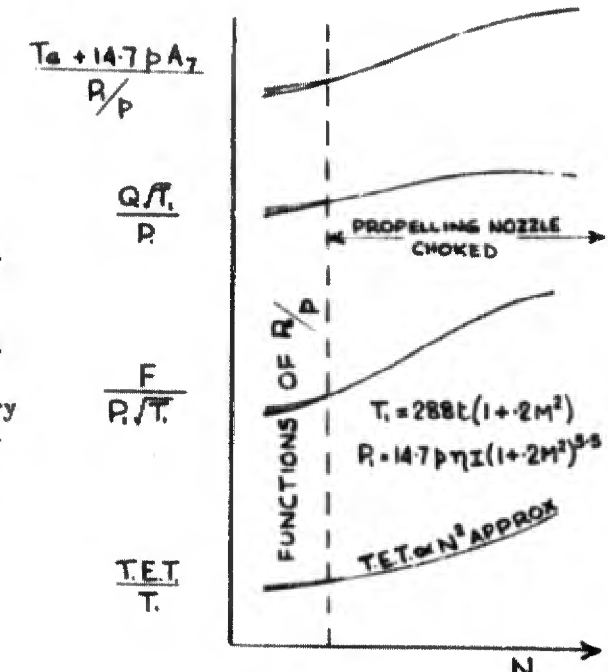


FIGURE B

Engine test bed non-dimensionals

Engine pressures and temperatures inside the engine are defined on Fig. A, but we will make minimal use of these today.

Engine test bed results of interest to us are usually analysed in terms of the ram pressure (P_1) and temperature (T_1) at the intake. Most engine companies eventually made results like Fig. B available to us in terms of psi, etc.

To give some crude understanding of what is going on inside the engine, $N/\sqrt{T}$ is a measure of the blade tip speed, expressed as an internal Mach number. Call it n_T, and note that it is directly proportional to $n/\sqrt{t}$ at any given M. Increase in this can produce more than pro rata increase in the Mach number of the air entering the engine, until it begins to choke. This gives the $\frac{Q\sqrt{T}}{P}$ airflow curve, whose final flatness is dependent on the amount of transonic flow.

Gross thrust is effectively airflow times fully expanded jet velocity. It therefore increases with n_T faster than mass flow does. Because the constant $14.7\ pA_7$ will be more than one third of the gross thrust at low M, gross or net thrust can increase much faster than n_T^4 before the intake chokes. Deducting momentum drag at the higher M, net thrust can increase even faster (away from zero value at an increasing value of n_T).

The turbine entry temperature to run the engine, divided by T_1, will be close to a curve of n_T^2. The resulting approximation $TET \propto N^2$ is usually much closer than the early compressor temperature rise assumption, $\Delta T \propto N^2$. The further temperature rise that must be produced by fuel burning will therefore increase more rapidly than n_T^2, particularly when it is the small difference of large quantities. So that fuel flow = (air flow x combustion temperature rise) typically varies faster than n_T^4. It is more sensitively related to thrust by the efficiency considerations touched on in the main text. There is no time today to deal with compressor/turbine matching and other internal considerations in my early theory.

Neglecting Appendix 2 scatter due to extreme Reynolds numbers or other secondary effects, curves like Fig. B are unique for given engine geometry, whatever their slope and shape. The propelling nozzle unchokes on throttling back to lower n_T, and the curves spread into a small family as a function of $P/p = (1 + .2M^2)^{3.5}$. More usefully, they are a function of M, (see Fig. B) since intake efficiency increases with reducing n_T as a function of M. This is illustrated overleaf.

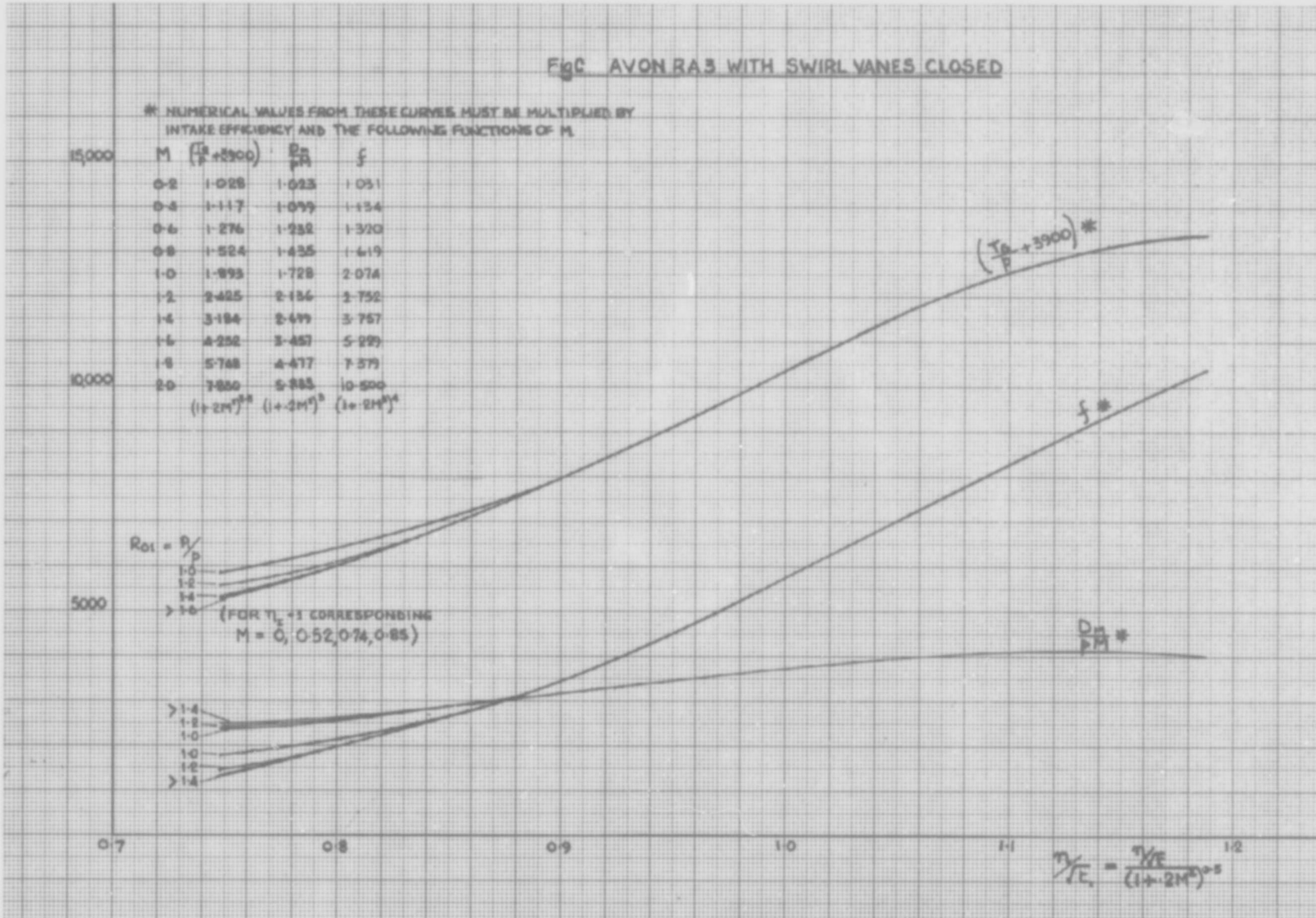


Fig C AVON RA3 WITH SWIRL VANES CLOSED
* NUMERICAL VALUES FROM THESE CURVES MUST BE MULTIPLIED BY INTAKE EFFICIENCY AND THE FOLLOWING FUNCTIONS OF M
M | $(\frac{T_R}{P}+3900)$ | $\frac{D_m}{pM}$ | f
0·2 | 1·028 | 1·023 | 1·031
0·4 | 1·117 | 1·099 | 1·134
0·6 | 1·276 | 1·232 | 1·320
0·8 | 1·524 | 1·435 | 1·619
1·0 | 1·893 | 1·728 | 2·074
1·2 | 2·425 | 2·136 | 2·752
1·4 | 3·184 | 2·699 | 3·757
1·6 | 4·252 | 3·457 | 5·229
1·8 | 5·768 | 4·477 | 7·379
2·0 | 7·880 | 5·885 | 10·500
$(1+\cdot 2M^2)^{3.5}$ | $(1+\cdot 2M^2)^3$ | $(1+\cdot 2M^2)^4$
15,000
10,000
5000
$(\frac{T_R}{P}+3900)$ *
f *
$\frac{D_m}{pM}$ *
$R_{01} = P_1/p$
1·0
1·2
1·4
> 1·6
(FOR η_I =1 CORRESPONDING M = 0, 0·52, 0·74, 0·85)
> 1·4
1·2
1·0
1·0
1·2
> 1·4
0·7
0·8
0·9
1·0
1·1
1·2
$N/\sqrt{t_1} = \frac{N/\sqrt{t}}{(1+\cdot 2M^2)^{\cdot 5}}$

Figure C shows the actual curves for the Avon RA3 with swirl vanes closed. It will be seen that, in the unchoked nozzle region at low n_T, the air flow (converted into momentum drag for the performance purposes below) increases with M or ram pressure ratio. This is due to the fall in pressure ratio on the compressor matching, away from the main surge/stall peaks. This fall in pressures and turbine temperatures to run the engine causes a fall in the gross thrust and fuel flow with increase of M. This direction of change applies to other engines, even though the numerical detail differs.

If reheat is used, it will normally be at higher n_T. It rapidly increases fuel flow and gross thrust, as discussed later, but the nozzle area is usually increased such that air flow is little affected. Intake matching is then relatively straightforward:

INTAKE matching

(1) The three main functions are directly proportional to P_1, so that 1% loss in intake total head pressure reduces air flow and fuel flow by 1%. Gross thrust is reduced by $(1 + \frac{pA_7}{T_G})$%, which is about $1\frac{1}{3}$% at high $n/\sqrt{t}$. More usefully, since momentum drag is reduced by 1%, net thrust is reduced by $(1 + \frac{pA_7}{T})$% which is usually around $1\frac{1}{2}$% at high $n/\sqrt{t}$ (so that sfc increases $\frac{1}{2}$% - although greater throttled back, intake losses are then falling rapidly).

This halved the thrust of the Lightning at very low M in extreme high altitude $n/\sqrt{t}$ conditions. These could not therefore be approached, avoiding the surge problem with incidence and weapon firing encountered on other fighters with the Avon. The compressor performance was matched to the more useful high subsonic and higher speeds, which explained the rapid increase in thrust on Figure 15.

(2) The intake performance was originally calculated and measured in terms of $\frac{Q\sqrt{T_1}}{P_1}$ and M. Matching this to the engine requirement on Figure B gives $N/\sqrt{T}$ for each point. Thence to $n/\sqrt{t}$ at each M.

Conversion to aircraft non-dimensionals

The equations on Figure B quickly converted such curves into aircraft M, etc. to give Figure C. These are the curves for the Avon RA3, assuming swirl vanes closed. The mass flow has been converted into momentum drag, because that is more useful for flight performance.

This immediately shows the reasons for the RA3 Canberra speed effects at nearly 100% intake efficiency. All the $(1 + .2M^2)$ terms are near 1 at low M, so that the first order M effect on momentum drag causes the initial fall in net thrust. This fall is then slowed by the $(1 \times .2M^2)^{3.5}$ ram pressure effect on gross thrust. As this term becomes dominant at high M and $n/\sqrt{t}$, nett thrust actually increases.

Broadly similar effects occur on most engines with high intake efficiency. The Avon in the Lightning had a new compressor, matched much deeper into choking with higher n_T. Although the momentum drag term increases with by-pass ratio, forward speed effects are now dependent on the control systems discussed at the end of the main text.

Nozzle matching

The basic size of nozzle is designed so that the engine is matched to run without reheat to give the best relationship between maximum TET and the aerodynamic $n/\sqrt{t}$ of the compressor-turbine units. Turbine guide vanes usually choke before the nozzle.

The burning of reheat fuel behind the engine lowers the gas density by increasing its temperature. If the nozzle area is increased proportionately, the pressures and temperatures through the engine remain about the same. This also applies to the fuel to run the engine, but much more fuel is of course used in the reheat burners (which used to be plotted as a curve against ^{n}T for each degree of reheat), and the gross thrust is increased almost proportionately - mostly from the increase in A_7, since a gross thrust/A_7 curve usually dropped slightly.

Having accepted the weight and complexity that goes with a variable nozzle, this facility can of course be used for purposes other than to maintain the above relationship, e.g. without reheat lit.

Thus, if the nozzle area is increased above the design value, the compressor delivery pressure and temperature may drop a little at given ^{n}T, but there will certainly be a large drop in hot end temperatures. The effect of this of course was to lower the curves for fuel flow and gross thrust, with negligible effect on the mass flow or momentum drag curve. Since engines were usually designed to run hotter than that giving minimum sfc, this often produced some improvement in sfc at given thrust (higher rpm) but with much reduced thrust at maximum ^{n}T. Closing the nozzle to the design value will restore thrust, leaving only the usual surge margins for acceleration.

Thus, opening the nozzle can often be useful for more economical subsonic cruising, particularly when base drag and spillage is allowed for. As noted under Concorde, it can also be useful for reducing take-off noise of a multi-spool engine by keeping up low pressure rpm and airflow. But opening the final nozzle will move the LP running line up towards its surge line, with little effect on the HP running line, as discussed below.

Current engines

The above 1½% or other approximations still apply sufficiently well over most normal operation. Where the engine is fuel limited however, thrust cannot be much affected either. Similarly, if the fuel flow is limited by a pressure in the engine, the fuel flow etc. will not be reduced until the intake losses reduce the pressure sufficiently. A further engine pressure/P_1 curve was usually supplied to define any cut-back in ^{n}T.

With more than one spool, the high pressure set effectively acts as a gas generator, with choking of the low pressure turbine equivalent to choking of the final nozzle in a single spool. Over this wide range, TET increases fairly closely to n_H^2, almost independent of the final nozzle and LP rpm. But the latter is increased by opening the final nozzle, and the Concorde control system keeps this nearly wide open for cruise and noise abatement, closing it only for maximum thrust and surge margins.

More generally with fixed final nozzle, the aerodynamic "gearing" of the LP shaft increases fairly linearly with ^{n}H towards a maximum value crudely inverse to the relative diameter of the fan (limiting its turbine speed; thence the trend to mechanical gearing or variable pitch in large civil fans). As the fan dictates most airflow velocity, the powers of ^{n}H discussed at the start can be much increased.

Whatever the slope and shape of the curves, engine data for a given geometry reduces to a simple set of curves like Fig. B, even for a 3-spool. This is neglecting the small and doubtful effects of Reynolds number, specific heat variation etc. discussed in Appendix 2. These "corrections" must be simplified for many purposes, if we are not to lose much understanding of the increasingly important but complex possibilities from variable geometry and control.

APPENDIX 2

Non-dimensional "corrections"

I promised to throw more light on my warning near the start about "approximations". The next pair of pages explain the very difficult situation that has developed in understanding and using engine data, as it becomes entirely computerised.

I discuss practical corrections to non-dimensionals on the final pair of pages to give sufficient accuracy for preliminary design work. People in the later program stages are now finding these of value as well.

CORRECTIONS TO ENGINE NON-DIMENSIONALS

It is important to try and understand why the complexity of some engine performance data has grown to almost unmanageable extent over the decades, out of all proportion to our knowledge of the matched aircraft drag discussed overleaf.

The 1940s and 1950s had no high altitude test bed data, and as much scatter of flight test data as today. This was probably due as much to changes in engine or aircraft during the programme as to instrumentation. My theory was mainly of value in checking whether scatter on the non-dimensionals was systematic deviation due to one or more of the following:-

(a) Specific heat variations with gas temperatures. Effects in the intake compressor air, and even from the lower value assumed behind the turbine nozzle, appeared negligible in comparison to the increase in fuel required at high intake temperatures to the combustion chamber. Intake temperatures on the test bed correspond to those at low supersonic speeds/stratosphere, but a few per cent more fuel flow was expected above $M = 2$/stratosphere, or low supersonic speed/low altitude, (often compensated by the reduction in compressor blade clearances). Vice versa, a few per cent reduction in fuel flow was expected at the lowest useful speeds in the stratosphere. A simple correction factor for most of this was possible, and one still wonders if knowledge of the aerodynamic effects of reduced γ in the hot section (confused below) justify much better than this.

(b) Reynolds number reduction in the engine. The reduction with P_1 was greatest with altitude at the lowest speeds (only partly offset by lower T_1 and higher $n/\sqrt{t}$). This had disastrous effect on engines with small poorly shaped blades which might never have left the altitude test bed today. On larger engines, the effect was often insufficient to even compensate the small favourable specific heat effect at the end of the previous paragraph. Much Canberra measurement was made when extended flights at 50,000 feet or more were of great interest. Rolls Royce at first suggested 5% deterioration in T/p and 2% on sfc. We later modified it to a more likely 2% and $4\frac{1}{2}$% or so, but it was difficult to find in the scatter between engines and aircraft. Remember that any thrust instrumentation calibration was at sea-level, so its own corrections would be suspect today. But our most important and reliable data was the fuel used in the many long distance records and other flights of the Canberra. Reynolds number can now equal ignorance factor, as with aircraft discussed overleaf.

(c) Combustion efficiency reduction due to Reynolds number/combustion parameters were inevitably included in these net results. Rolls Royce believed that they kept combustion efficiency near 99% in those unreheated days, except near idling conditions. The latter were never considered on any non-dimensional basis, as the thrust and fuel were relatively small anyway.

(d) Power or other offtakes. Assuming selection of the most efficient method of taking off P horsepower (0.746 kW), simple analysis showed a loss in thrust up to the order of $\frac{1}{2}$P lb or so. It was usually more difficult to be sure how the crew varied P, but the power that did not fall off with altitude usually had negligible contribution to non-dimensional scatter. Although engine sizes have increased, I don't doubt there are aircraft today which try and use them as small permanent power stations. One hopes not at high altitudes, because by-pass engines can be more sensitive to off-takes, and the corrections can justify an even larger computer program!

(e) Specific heat variations with humidity were found to be important. I don't know what has happened to such corrections in many large computer programs today. Perhaps it is part of a new operational scatter, as most sophisticated altitude test beds are in temperate climates!

The 1960s saw the introduction of sophisticated altitude test beds. In addition to all the previous engine parameters, altitude M temperature could now be varied effectively; over a range that normal aircraft do not fly, or so briefly that the last per cent of performance is unimportant.

Whilst prospects for using an engine in a number of types of aircraft still seemed good, there might always be the "designer" who would wish to use the engine for Mach 1.5 at sea level, or with "glider-like" wing loadings. Since much extreme testing was useful for handling limits, and the performance instrumentation was feeding into the computers anyway, it would otherwise be wasted.

It was clearly difficult to match this even partly to possible applications, so it was easier to start from what I called the test-bed non-dimensionals in Appendix 1. Famine to feast! We were soon saturated with a mass of P_1 and T_1 corrections over every range of combinations of parameters.

These were presented for a long while in ways that could produce our matched non-dimensionals fairly readily. Even as we turned increasingly to the computer for performance ourselves, we could still check a few points by hand to be sure of what we were doing. The people doing aircraft performance in the engine companies had adopted methods similar to our own, with needs for matching still in mind. But as corrections grew more and more complicated, we wondered what scatter, interpolation or extrapolation lay behind the vast families of beautifully smooth curves.

The 1970s sees nearly all engine data on computer tapes, and abandonment of the test-bed non-dimensionals that gave us most of our confidence in what we were doing. Engine performance people have undoubtedly done wonders in matching computer programs to the running of a complete engine on the altitude test bed, or a synthetic estimate from components prior to that. (But Reynolds number and other scatter is becoming obscured as discussed overleaf.) With government tightening of fixed price/performance guarantees, perhaps this is becoming an end in itself, with the added complication of getting things agreed internationally. (US and other contracts with SAE ARP 681).

Although the computer can be run to give selected points for a given installation, it is becoming too expensive to up-date complete sets of data, even on the most appropriate computer. Data is either continuously modified or out of date in the development stages. In the flight test stage, the atmosphere is rarely kind enough to be ISA or any other standard.

We are beginning to defeat the real object of the exercise. We often dare not make aircraft performance estimates over the full range, or analyse flight test data as we get it. Collaborators may be tempted into rash use of out-dated data, or out-dated corrections of new data. Either way, the errors may be much greater than the corrections which the very expensive altitude testing and computers are trying to give us. Ironically, these corrections may be a lot less than the practical scatter in actual flight conditions.

The Future. After the simplified discussion of main areas of precise matching overleaf, perhaps our engine friends can give us simpler laws or corrections for those areas. Since we are all interested in understanding how to improve joint performance in those areas, this should also be of great interest to customers.

PRACTICAL CORRECTIONS TO NON-DIMENSIONALS

Perhaps it will encourage our engine friends if they first appreciate the degree of approximation inevitable on the aircraft side. Compared with the wealth of possible engine corrections (only partly summarised in the last two pages) we largely worry about Reynolds number for each configuration:-

1. **Aircraft** Reynolds number

$\propto \frac{\rho V}{\mu}$ for each aircraft or part thereof

$\propto \frac{pM}{t^{1.3}}$ since $p \propto t^{0.8}$ approximately in the aircraft atmosphere

$\propto \left[\frac{M}{t_I^{1.3}\frac{\overline{W}}{p}}\right] \frac{\overline{W}}{(t/t_I)^{1.3}}$ where t_I represents ISA value, and $\overline{W}$ = a mean Weight.

The square bracket can vary up to 20 to 1 from the top left to the bottom right of a drag diagram. If roughness drag and transition shift were negligible, skin friction coefficient would be increased over 50% by this, particularly if compressibility is allowed for. Local pitot rakes have confirmed this. With subsonic C_D mostly in friction drag, large variations should surely be measured with pitot rakes removed?

But measurement on dozens of operational aircraft usually finish with the simple conclusion that the scatter of data justifies little or even no systematic variation in effective C_{D_0} below the critical Mach number. "Effective" here means the value of C_D extrapolated to zero C_L^2 for each Mach number. Part of the explanation lies in the variations of Reynolds number thus involved for a given aircraft. If one goes to all the bother of correcting for that (and getting a small variation in C_{D_0} like some research laboratories), one has to reverse the whole process and get back to the original result for the actual aircraft.

Most of the explanation must lie in the disappearance of practical roughness drag and the moving back of transition points at low Reynolds number, however reluctant we are to rely on it at the design stage. The true picture possibly resembles those elegant experiments in rough pipes we remember from our college days, where the curves are broadly flat over vast ranges of Reynolds number, but delicate changes of slope can only be detected by the most sensitive instrumentation of a simple uniform pipe. And a fly then spoils the experiment!

Where we do obtain reliable information on a small variation of C_D with aircraft Reynolds number, it is accurate enough to build it into the drag diagram through the term in the brackets - in simpler and more accurate terms, just assume a mean aircraft weight. This immediately gives the value of p and altitude for each W/p. Thence the exact ISA coefficient of viscosity and speed, giving the Reynolds number for C_D variation.

Conclusion on secondary corrections

The terms outside the brackets have negligible effect. W typically varies 10% or so from a mean value, and t up to 5% from ISA. So that Reynolds number variation is only about 1% of that discussed above, which itself has small or uncertain effect. Although sudden shifts of transition or separation can occur at around a given Reynolds number, it will vary from day to day in production, if drag is that sensitive.

Meticulous computer accuracy cannot make up for ignorance of the basic laws of nature. It must certainly not obscure such ignorance. Can we be assured that some engine corrections are not similar?

2. Matched Engine

Aircraft secondary corrections opposite were negligible, unless obscuring something wrong with the aircraft. This is because up to 99% of known effects can normally be built in by matching on a mean weight and drag ISA basis.

In regard to engines, we have already discussed the important extreme case of the Concorde Olympus. Although an accuracy consistent with our knowledge of buoyancy and gravity variations has been demanded, Reynolds number effects have been matched in with negligible variations through the flight range. (Only transonic acceleration at greater altitude could introduce a small effect). Can lower precision examples be treated similarly for areas of prime importance, with simple demarcations as reminders of exceptions? Consider a few preliminary ideas for particular cases, before discussing the simplifying requirements of future matching:-

Steady flight. Most predictable fuel is normally used at steady (or near steady) flight, often throttled well back. The mean ISA basis should be used to define the sfc loops to the right of the minimum drag speeds. This defines p and altitude at each point as opposite, but also $\frac{1}{t}$ and any P_1, T_1 or other parameters required for test bed data. ISA data is usually good enough, since the $sfc/\sqrt{t}$ law seems pretty good for matched conditions. n itself is now seldom of prime interest, but values of $\frac{TET}{t}$ can be cross-plotted if engine life is of concern. If TET is defined relative to the maximum allowance, it is usually sufficient to define these curves up to around $\frac{TET}{t} = 1$, extrapolating beyond afterwards where necessary. Most combat aircraft now have few critical operations near maximum TET in the stratosphere. Most subsonic transports are now cruising economically around the tropopause.

Acceleration/climb is usually with or without reheat at or near n = 1. The curves for $\frac{TET}{t}$ above 1 should therefore be drawn for the maximum climb rating in ISA conditions, i.e. near the tropopause at $1.153^2 = 1.33$, and sea level at 1.0, extrapolating afterwards if necessary down into the small region of overlap without reheat. Any discontinuity here gives warning of the size of corrections, if any. (Opposite signs in recent engines, so perhaps some will cancel). Below the intersection with the minimum drag speed, a further curve around $\frac{TET}{t} = .9$ can be drawn as a further guide, and to cater for accelerations on a hot day at low altitude.

For the exceptional case where great precision is required in the stratosphere, an extra curve for 45,000 feet or so will give warning of any large corrections at the highest $\frac{TET}{t}$. True error in important operations is the main criterion.

Deceleration/descents are usually near idling. The associated thrusts and fuels are still sufficiently small to be treated dimensionally for most purposes with sufficient accuracy.

Future engines

TET may be cut back by the engine control system at low and/or high speeds, as discussed at the end of the main text. Together with the matching requirements of the corresponding aircraft, this reduces the variation in engine Reynolds number etc. much further. Despite very much greater variations of temperature inside an engine, one hears of empirical correlations of actual test bed results where even its first order effects on Reynolds number have been neglected. If "Reynolds number" corrections do not correlate with changes in dimensions between engines, one suspects that the computer program is obscuring scatter that may otherwise be insignificant in the important flight range.

Further trends to low level operation following the latest Middle East War could eliminate important secondary corrections almost entirely. So it is ironic that the engines which have required the largest computer programs could end up almost purely non-dimensional for most practical purposes! We are seriously considering a return to non-dimensional methods, even in the detail design, flight test and operating stages. It is already proving useful for assessing Russian competitors etc.

Although the accuracy of such corrections is unproven for all applications, it should be good enough for most preliminary design. If not, you perhaps ought to reconsider the engine before losing sight of the basic laws!

ADDENDA ON SPOKEN PRESENTATIONS

From various omissions mentioned in the Postscript, the Director asked me to illustrate some prospect for Mach numbers much more than 2 particularly. Western experience would suggest cost problems more than prospects, but we need simpler methods for keeping all possibilities under review.

PROSPECT FOR HIGHER SPEEDS

Although the Second International Symposium on Air Breathing Engines a few weeks ago was still dominated by Americans and Europeans, there were surprising numbers of Japanese, Russians and Chinese. The second in the field can always benefit from the experience of the pioneers.

America spent a fortune on manned vehicle R & D around 10 years ago, but only the secret budgets of the CIA could afford much more than prototypes. This cost experience is producing a Mach number limit of 2 point something in Western minds, almost as strong as 1.0 was 30 years ago.

EXPENSIVE TURBOJETS FOR M ≈ 3

2 off XB70

DOZEN OR TWO YF-12 or SR-71

MIG-25 FOXBATS

ROCKET VEHICLES

3 off X-15 for M ≈ 6

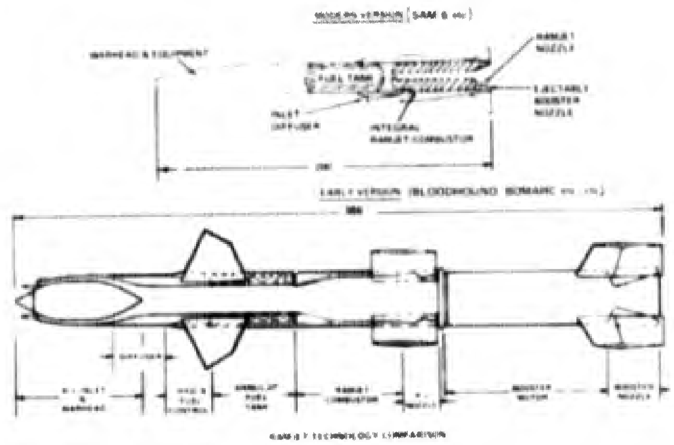
4 off MERCURY for M ≈ 26 } Parachute recovery

10 off GEMINI for M ≈ 26 } Parachute recovery

4 off APOLLO for M ≈ 26 } Parachute recovery

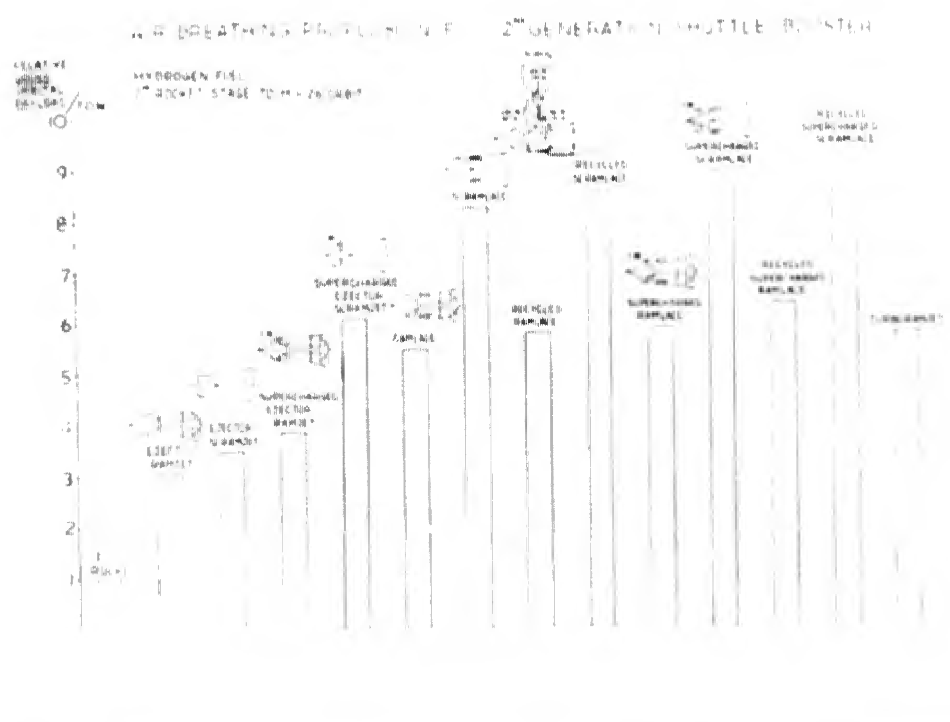
9 off APOLLO for M ≈ 36 } Parachute recovery

+ RUSSIAN SPACE EFFORT



Is it time to start a complete re-think? The Ramjet technology in the Russian SAM 6 surprised most people, except perhaps LTV missile systems. Their drawings dramatise the recent improvement. Many of us already thought the Ramjet could prove cheaper down towards Mach 2 at very high altitudes, or even around Mach 1 over shorter distance at low altitudes.

Ramjets are better still for a wide range of speeds up to Mach 12 or even more. Efficient operation beyond Mach 6 or 8 requires Supersonic Combustion in the Scramjet. Ramjets require little more than an efficient intake and nozzle with something extra for low speeds. Marquardt's have examined many unusual possibilities, a few of which are compared below. These show several times more percentage payload than hydrogen rocket boosters, which themselves are much better than the solid rockets on the present space shuttle. Interpret:



Another factor is suitability for terrestrial operations. Steadier cruise applications would be slower, cruising at very great altitudes for radiation cooling. Air breather enthusiasts may eventually admit that they could be over-stating the prospects for a single-stage into-orbit aerospaceplane. But they'll argue fiercely that other applications are now almost within reach. Get back a basic understanding by using good approximations for things like Reynolds number corrections, and you'll find far more important preliminary design possibilities and unknowns to argue.

Don't wait for computer programs or other data on such engines, or you may wait for ever. What I said earlier about massive computer programs only seems to apply to the gas turbine machinery that established companies have to sell for Mach numbers up to 2 or so. Engine computer programs could get the West into "sound-barrier" thinking about anything except circular fixed geometry engines up to Mach 2 point something. Given the usual specialist company advice on combustor design and nozzle materials, an aircraft man is just as capable of designing a Ramjet for example. Since it doesn't have to be circular in section, he has far greater possibilities for integrating the cheapest complete design to a given requirement.

I think this gives sufficient hint of how to re-introduce "innovation and originality" back into preliminary design, as requested by the Director. Without much better ideas for reducing cost, prospects for Mach numbers much above 2 look pretty bleak.

INTERNATIONAL COST LIMITS AND AGARD

The first 70 years of manned flight experience produced the bad habit of expecting governments to help project funding to increase exponentially. An American government agency analysis of transport aircraft shows a typical doubling of unit price around every five years, and development costs around every three years:-

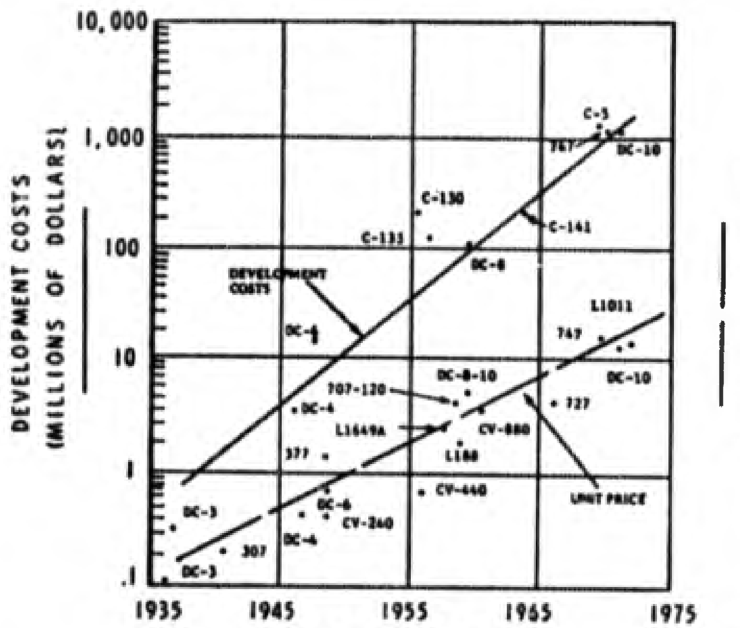


Figure 5. Transport Aircraft Development Cost and Unit Price RADCAP 1972

It is interesting that the Concorde plots reasonably near such curves for an in-service date of 1975/76 (American SST R & D expenditure is almost as great, with the most visible remains in Disneyland!) The Space Shuttle development is not too far adrift nearer 1980, but unit price has risen well above. These rates of increase are much more than the general rate of inflation, due to the increases in size or performance sophistication that seemed worthwhile up to now.

Whilst development costs were well below $1 million, redoubling was generally accepted. This continued through WW2 and the Cold War that followed on a national basis, often on a company equity basis. But where development costs are now measured in $ billions, increasing numbers of taxpayers in the Democracies are questioning such base for supporting an aircraft industry, even without redoubling. If the Concorde had continued as a purely British project, there seems little doubt it would have been cancelled long ago, however much cheaper we could have made it. The Space Shuttle has carried on in the wake of NASA budgeting established by President Kennedy for the Moon, helped along by various efforts at European collaboration. The vestiges of this remain in the Spacelab module, but it is a pity that the pressures to keep close to the wake did not allow more time to establish closer collaboration on proper examination of all the economical possibilities for the Shuttle itself.

Inter-European co-operation is already much closer, and we ourselves are fortunate to work with all the larger European countries, and with America from the B57 onwards. This has all been ad hoc, but AGARD now provides a forum where we can talk frankly about our experiences on cost and other problems, not only on NATO aircraft and weapons, but now on Space and civil aircraft as well.

DESIGN TO COST PROTOTYPES, etc.

American speakers at this Conference have put much store on Design to Cost and Prototypes. These are processes with which we were very familiar on Canberra and P1 for example, where two prototypes apiece were designed and built for £1 to £2 million respectively. Dassault particularly have continued the experience with many prototypes, some of which have carried on into production. This is at European prices which are well below $\$37\frac{3}{4}$ million for the size of aircraft illustrated with such beautiful colour films today. We congratulate the designers on a fine technical achievement, particularly in the face of the previous American paper system. The films brought back much nostalgia because the "new" features of inclined seat, strakes, fixed intake for supersonic speeds, and "air superiority" at $T/W > 1$ and $W/S = 50\ \text{lb/ft}^2$ were all featured in the P1 design of 25 years ago. (Although we flew what is now called CCV some years after in TSR2, we still await operator reactions to completely electronic pitch stability).

We then remember that the total cost of getting the P1B Lightning into production was many times greater again. The extra weight of equipment, weapons, fuel etc. had then rather spoilt the T/W and W/S, although these were more than adequate for "air defence". The cost of getting the Harrier into production was also many times the P1127 prototypes, and similarly for French experience. This is not to say that the total is any greater than going straight to a development batch, so this approach may still have merit in getting started at all with risky new features. But it will be at the expense of overall time, which becomes infinite if the risks prove too great. Many governments can no longer afford such risks, and must certainly include all costs in any process leading to eventual operational systems.

Something nearer a development batch approach has merit in getting more thorough preliminary design and ground testing in a shorter overall programme. A mixing of the best features of both systems may be best of all, as partly carried through on Jaguar. (This is proposed in my 1966 reference on page 3-46)

The top half of the next slide is taken from the excellent lead paper at the Florence conference by Bill Lamar of Wright-Patterson. Many Americans now look back on several items as little more than fashionable slogans:-

SOME AMERICAN COST REDUCTION METHODOLOGIES

· LIFE CYCLE COST	· FLY BEFORE BUY	· SYSTEM INTEGRATION
· DESIGN TO COST	· COMMONALITY	· MILESTONING
· PROTOTYPING	· "ILITIES"	· SKUNK WORKS
· SHOULD COST	· TOTAL PACKAGE	· ADAPTIVE MANAGEMENT
· FLY-OFF	· KISS vs ADVANCED TECHNOLOGY	· CONCURRENT DEVELOPMENT

EUROPEAN EXPERIENCE SIMILAR, MOVING TOWARDS A BETTER BALANCE OF:
FIXED PRICE PROTOTYPES — FIXED PRICE PRODUCTION
FIXED PRICES FOR INCREASING AMOUNTS OF TOTAL DEVELOPMENT & PRODUCTIONISING
MORE THAN A FRACTION OF 1% LIFE CYCLE COST ON TRULY PRELIMINARY DESIGN

It is interesting that, less than two years after YF16 and 17 design had started, "prototyping design to cost" was already being eclipsed by emphasis on "life cycle cost". This is one time that we have been years ahead, rather than behind an American fashion. Life cycle costs can be well over 100 times the prototype cost, and too tight a fixed price at the early stage can cost an operator dearly over years of later operation. Our BAC paper to the Florence conference showed this to be at the root of the current budgetary crises in many Western Air Forces.

I am sure that Bill Lamar would agree with the objectives I have written below his slide above, stripped of all slogans. Perhaps he will forgive me for putting his KISS ("keep it simple stupid") in the blank space on his slide. It still seems a problem after more than 10 years, due to the early needs of research establishments to improve performance with much less emphasis on cost. The problem of advancing technology and systems to improve cost-effectiveness is much more difficult, which is why I took on a new title many years ago.

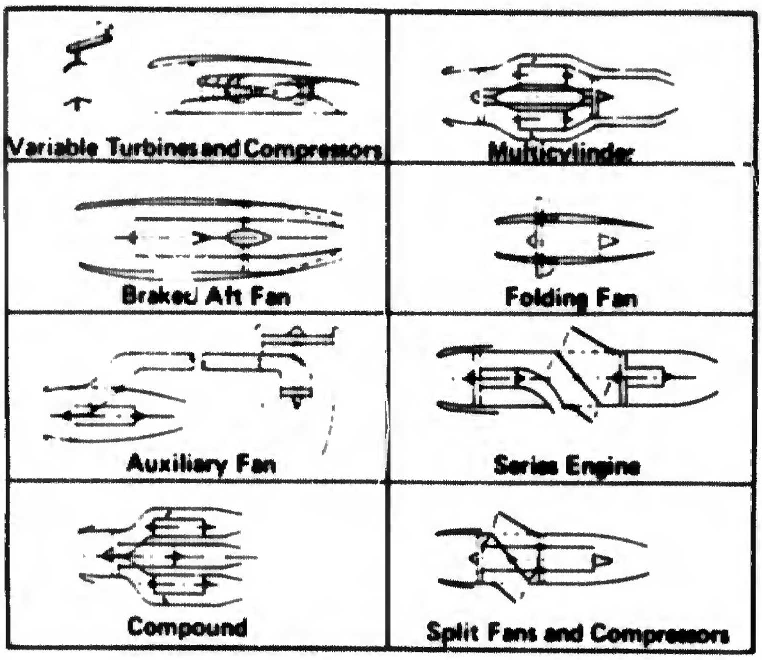
Bill Lamar tried to get everyone in Florence to agree that we should aim to devote nearer one per cent of Life Cycle cost to Preliminary Design. This must be right in most situations from now on - despite all the temptations to get something in the air more quickly, e.g. to make use of an existing engine; pressures to get into the wake of favourable political climates; sudden new requirements; the continuing desire to keep all parts of an organisation fully employed.

Old habits die hard, and most people find it difficult to see how one per cent could possibly be spent on Preliminary Design; before trying to sell a finally frozen configuration without all-out attempt to find better alternatives. If so, let me illustrate one hint at the end of my Appendix 1 on how to extend "innovation, originality and judgment" into preliminary design even up to Mach 2.

ENGINE VARIABLE GEOMETRY

The only major form of engine variable geometry is often in the final nozzle, as is virtually essential with reheat. Long before the Concorde, we found it only natural to adjust the nozzle to improve economic fuel consumption on the Lightning. But that was with our simpler understanding of the Avon performance. With massive computer programs even for a single max dry thrust position, we now find it increasingly difficult to get it considered as a preliminary design factor. (For example, unless it gives a bonus with multi-spool by-pass, perhaps we should also examine the Wright-Patterson plot of Specific Range in AGARD 124. They suggest that by-pass is responsible for the fall-away in combat aircraft specific range, and only the F111 with fully spread wings compares with the old Canberra!)

But there are many other possible forms of engine variable geometry. Some fairly extreme examples were shown at the Air Breathing Symposium by Walter Swan, who is Head of Project Technology at Boeings.



PERFORMANCE PROBLEMS RELATED TO INSTALLATION OF FUTURE ENGINES IN BOTH SUBSONIC & SUPERSONIC TRANSPORT AIRCRAFT

There have been fairly modest examples of the first case in the inlet guide vanes or stators of Avon and later axials. Although some of the others appear more extreme at first sight, one might ask if the engine is keeping pace with the aircraft? Control surfaces, flaps, airbrakes and retractable undercarriage are on all our aircraft. Many have increasingly sophisticated variable intakes, extending flaps and slats, plus variable sweep. These have all been added to reduce the overall cost of meeting particular requirements. Have all engine possibilities been examined likewise in appropriate permutations? Much of the cost, weight and drag of some of our variable intakes and nozzles is due to the fact that we are having to cater for engines of essentially fixed geometry!

The Russians and NASA now believe it feasible to convert a pure jet for supersonic speeds into a by-pass around 1 for subsonic speeds, without undue matching complications. But is the weight and cost of other engines cycles, such as those I mentioned earlier, now likely to prove more attractive? There are more modest forms of variable geometry even with gas turbines. The Pegasus nozzles for example, and we have examined many variations on that theme. Fortunately, this is an engine whose performance is still available in the older non-dimensional form.

RESTORING UNDERSTANDING

I trust that I have now begun to get across the warning that the latest engine computer programs are already inhibiting our ability for "innovation, originality and judgment" in the Preliminary Design and later stages. We have already encountered this in the following respects:-

1. The economic balancing of aircraft/engine matching considerations in relation to the importance of many problem areas or possibilities. (Compare the single matching diagrams of Canberra, Lightning etc.)

2. The importance of many engine installation factors for which we used to have immediate feel (Appendix 1). There is even a trend at the time we are still doodling round the Director's drawing of a standard pilot, for the engine people to ask us for all variations in required power and air off-takes versus speed and altitude, to be concreted into their computer programs.

3. The importance of many engine factors to the aircraft designer and operator, rather than hazardous extrapolation of experience. This can apply even to overall results, let alone the finer design points, e.g. without curves like Fig. 32, it may not be obvious if sfc and thrust are each increased several times by reheat, such that fuel flow is well over 10 times greater.

4. The possibilities as well as problems of modern engine control systems (read Section 6 and its Conclusion); e.g. all manner of thrust curves like Fig. 31 are possible, instead of the familiar flat concavity against M plus fairly standard rates of fall-off with altitude and temperature.

5. Appreciating all the possibilities for engine variable geometry.

6. Possibilities for more economical flight, high speeds, V/STOL, RPV's etc.

7. Difficulties in understanding scatter of measured engine and aircraft performance due to the approximate nature of corrections like "Reynolds number" in these computer programs.

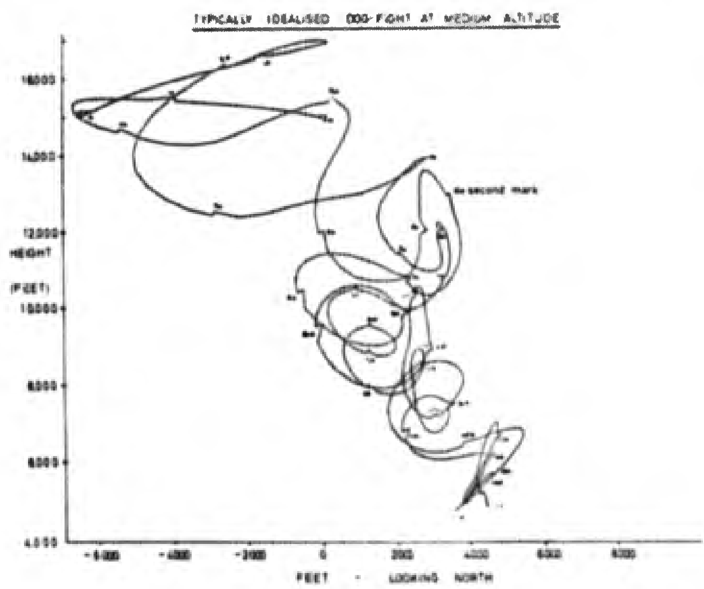
The problems we are beginning to encounter made me ponder the following recent news extract:-

> "Abacus Beats Electronics
>
> "In eight sets, each requiring very complex mathematics, the abaci again won every time. The leader of the Chinese team, Chang Chuan-shiang, 72, said that the use of the abacus stimulated the mind, while Western-style electronic calculators led to mental decay."

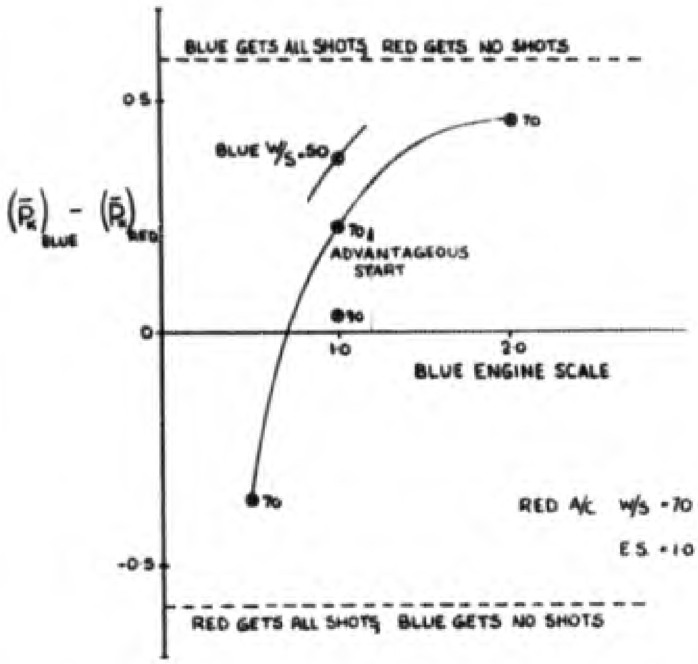
I thought these results might be partly due to the other team being unfamiliar with electronic calculators. Further check shows that the latter were used by properly trained non-Chinese! Although we have come to rely almost entirely on computers in recent years, Section 5 and other parts of my paper showed the need for appropriate graphical or other outputs to aid full understanding and prevent such mental decay.

PRELIMINARY DESIGN OF COMBAT AIRCRAFT

We established earlier that the F15, YF16, YF17 and P530 have all ended with a combat $T/W \approx 1.4$ and $W/S \approx 60$. More extreme computer designs and technology are being studied to the SEP under g and turn rate criteria of Major John Boyd. These date from the mid 1960's, when medium altitude dog-fighting was occurring occasionally over North Vietnam. The dramatic importance of such criteria appeared to be confirmed by digital computer simulation. We all carried out such work with ever-increasing sophistication of assumptions like "if the opponent is within certain limits of range, angles off, relative directions, range rates, differences of speeds and altitudes and sight line spin rates, the optimum manoeuvre is a perfect YO-YO, Split S, or whatever", regardless of the split-second demands on pilot eyesight, judgment and airmanship. This also assumed isolated 1 vs 1 combat well away from the ground, as in our example on the right. By starting high enough, it was possible to show that one ran out of ammunition or fuel before having to worry about the lethality of the ground or stall, as Helmut Langfelder indicated.



IDEALISED EFFECT OF FIGHTER PARAMETRIC VARIATIONS ON KILL PROBABILITIES



By accepting these and much other idealisation of pilot behaviour, it was possible to come up with devastating advantages from low W/S and increased engine size well beyond $T/W = 1$, such as shown on the left. But the introduction of pilot "domes" in the simulation has shown the importance of the first three assumptions below. ($T/W > 1$ "takes charge of many pilots", as compared to WW2 or high altitude combat thereafter, when small g's and SEP's gave pilots more time to make the best tactical moves.)

1. Continuous and completely precise information, instantly available at pilot's brain.
2. All decisions are immediate, perfectly clear-cut and repeatable.
3. Perfect piloting, often with zero imbalance and near perfect weapons.
4. Perfect CAP/other organisation to produce an advantageous start to combat in all conditions.
5. All of combat totally isolated from all other aircraft, ECM and ground defences (up to several minutes).
6. Insignificant ground collision or other worries.

The last three assumptions are difficult to simulate at all on the ground, and are particularly important in a future NATO context. There is little question that all assumptions exaggerate the importance of traditional dog-fighting and its beneficial design features, compared to those against future ground defences. If only four such assumptions are doubled in importance, the total exaggeration of kill probability is sixteen-fold. Wayne Huff's paper showed the difficulty in obtaining clear-cut confirmation that a reduction in drag of his leading-edge flap system was a good thing, even after discarding 200 of his 300 simulations with carefully selected pairs of pilots. No such difficulty was encountered with the original "Peas of S" criteria, or idealised digital computer simulations, but real wars have confirmed that variations in pilots and cockpits are more important than many other design features.

As the flap drag reduction was not associated with important weight and cost penalties, the choice was obvious anyway. But what about the cases where clearer numerical penalties must be justified? Helmut Langfelder mentioned the range/payload penalties of high T/W and low W/S, and I can again illustrate the order of importance of this on the blackboard, in terms of fraction of design take-off weight, W.

Engine weight, $W_E/W = \dfrac{T/W}{T/W_E} = \dfrac{1.2}{8} = \underline{0.15}$ for $T/W = 1.2$ at take-off $= 1.4$ at combat

This assumes advanced engine technology with $T/W_E = 8$. The latter can be improved only by even more expensive and fragile engine technology, without book-keeping deduction of weight items that must be included in the

Total propulsion weight, $W_p > \underline{0.20}$ W even with advanced technology, if all items that vary with T are accounted, including intakes, installation structure, etc.

Wing weight, $W_W > \underline{0.10}$ W for a combat wing loading below 60 lb/ft^2, particularly if all items that vary with S are accounted, including tail areas, fuselage joint structure, etc.

Fuel weight is increased on different parts of a mission. The wing can account for around one third the drag at high subsonic, transonic and supersonic speeds. Larger T means that the engine must be throttled back to poorer sfc, particularly for economic cruise at low altitude. Installed propulsion drag, including intakes and nozzles, is generally increased.

Thus, the Total weight affected can exceed $\underline{0.40}$ W for dog-fighting emphasis, even with advanced technology.

It is interesting that the other new USAF "air-superiority" aircraft in the last decade is the F111. This has $T/W < 0.6$ and $W/S > 130$, so it is easy to consider at least 2 to 1 on these parameters when range/payload and gust response are emphasised as much as dog-fighting. With equal technology configurations, this can save more than $\underline{0.20}$ W.

This is of the same order as the total payload in most aircraft. If put into extra fuel, the reduced sfc and drag improves the range more than pro rata. (Detailed studies confirm this large order of difference, despite many other factors in the less important wing difference, e.g. loss in space eventually offsets the fact that $\frac{1}{3}$ S can get buried as far as drag and weight are concerned. Variable sweep can have more than three times the aspect ratio of a delta say, and this is nearly as important as $W/_S$).

Whatever the balance of mission requirements, design optimisation requires more realistic numbers for the value of dog-fighting characteristics, particularly at low altitude. To obtain more consistent understanding of the human factors, BAC have deliberately started with a single pilot "dome" against a variety of repeatable electronic opponents.

<u>CONCLUSION</u> So don't leave this lecture series thinking that the only job left for us all is to link up our computers to pick out the winning SEP's under g, and then retire. There are many other tasks to prevent decay in our thinking than plotting out endless computer results for fighters with the highest $T/_W$ and lowest wing loading.

Given some ingenuity in configuration, that kind of $T/_W$ or wing loading is sufficient for both VTOL <u>and</u> STOL. The latest fighter technology can reduce the penalty of either of these. Without on-board pilot restraints, RPV's allow a wider choice of V/STOL configurations.

I'm afraid there wasn't time to prepare suitably unclassified slides on the scope for producing cheaper V/STOL and RPV's. Hints for improving fuel and other economy were mixed amongst the history of the Lightning and current subsonic transports numbers. Despite the limitations on time, I trust that I have met the Director's requirements to give some hints on how to re-introduce "judgment, innovation and originality" back into the preliminary design process.

DESIGNING FOR MANEUVERABILITY - REQUIREMENTS AND LIMITATIONS

BY
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SUMMARY

Based on the technology expected to be operationally available in the seventees, the design requirements and relevant design trades to ensure adequate maneuverability of military aircraft will be discussed. The basic contradiction of range / pay-load and maneuverability requirements will be treated from the point of view of preliminary design. Maximum lift and buffet penetration as well as maneuvre devices will be presented. Maneuverability consists of many factors, the main ones related to installed and available thrust-to-weight and usable lift, but such aspects as the reasons for extreme maneuvres for optimum attack and evasive action, the maneuverability for low-altitude penetrators (automatic TF-flight) and for air-to-air missions including crew toleration of G loading as well as cues to initiate extreme maneuvres also need consideration. Reference will be made to cost implications and some remarks on a point design for extreme maneuverability are intended to give more insight into the characteristics of balanced design of useful fighter aircraft.

NOTATION

SL = Sea Level

TF = Terrain Following

SEP = Specific Excess Thrust

$C''_{B(M,\alpha)} = \sqrt{C'^{2}_{B(M,\alpha)} - C'_{B0\,(M,\alpha=0)}}$

$C'_B = \dfrac{1}{K \cdot C_B}$

$C_B = \dfrac{\text{Wing Root Strain Signal}}{q}$

C_L = Lift Coefficient

M = Mach Number

b = Wing Span

1.0 Introduction

In the design of combat aircraft, maneuverability has always been an important consideration, but speed and range / pay-load have until recently been the overriding criteria. Speed in itself is now no longer a primary design goal and the increase of thrust is rather viewed as a means of achieving maneuvering performance in a wide sense. It is, therefore, now the problem of obtaining a proper balance of maneuverability on the one hand and economic range / pay-load performance at proper operational speeds on the other, which represents the fundamental design consideration of such aircraft. There is no generally valid answer for deciding this design compromise and each configuration must be assessed against proper requirements and operational utilisation. A review of basic characteristics of single and twin-engined fighter aircraft, expected to be operational in the current decade, shows that thrust-to-weight ratio, wing loading and combat weight vary considerably. Thus Figure 1 presents thirteen such designs, indicating that the thrust-to-weight ratio varies from about 0.5 to just over 1.4 and the wing loading between 250 kg/m^2 and about 500 kg/m^2. In general, better maneuverability is obtained at the lower wing loadings and with more thrust, while more range requires a higher wing loading, particularly for low altitude penetration, and a closer match between cruise thrust required and maximum dry thrust available. The sizing of these aircraft shows them to have combat weights between 7 tons and up to more than 30 tons, the majority, however, lying between about 10 and 20 tons. This is shown on Figure 2. The designer of combat aircraft is generally not quite free in choosing the thrust, since either the capability of existing engines or those in development will be specified for the design. Figure 3 shows that military engines available in the seventees have a thrust-to-weight ratio between 4.5 and 8.5 and a sea-level static reheat thrust between 6000 and 12000 kp. It is not only of course the engine size and weight, but also the engine cycle, which has a predominant influence on the aircraft performance qualities. The contrary requirements of cruise economy and maneuvering thrust must be related to priorities of the mission.

2.0 Preliminary Design Selection of Basic Parameters

In the preliminary design process a large number of configuration is generally investigated on the basis of point designs to examine the influence of the variation of the basic design parameters on mission performance. In the example shown on Figure 4 fixed engines are used for a given pay-load and radius of action, while other parameters, such as specific excess power (SEP), take-off and landing performance, acceleration time and load factor are plotted as boundaries to display their effect as wing loading and thrust-to-weight are varied. In such plots, the critical performance requirements, which will have a decisive effect on the design and determine the choice of its basic parameters, are shown. The case illustrated is such that the SEP requirement and the touch-down speed are critical and lead to minimum thrust-to-weight and maximum wing loading, as shown by the shaded area in the diagram. On Figure 5 a fixed wing geometry design is compared with a variable geometry configuration to illustrate the effect of the proper choice of wing loading, to satisfy the requirements, on the resulting take-off weight. It is seen that the variable geometry design will result in a slightly lower take-off weight. The diagram also shows that wing loading has a very strong effect on all-up weight and a decrease of wing area is a powerful means of decreasing the weight and as a consequence the cost of the design. It is often assumed in preliminary design, that cost and weight are directly proportional and this, indeed, may be correct as a first approximation and in a small range of variation. In In extreme cases such an assumption needs verification. Nevertheless, the strong dependence of weight on the choice of wing loading is very significant from the point of view of cost.

Once the general size of the aircraft has been chosen, more detailed variations can be studied. For instance, in Figure 6 for combat aircraft in the 15000 kg class, the direct influence of the maneuverability requirements expressed as SEP at sea-level and 1 g flight at M = 0.9 on the mission radius for a range of wing loadings, thrust-to-weight ratios and at different overload conditions is presented. Again typical effects are illustrated, namely the big influence overload conditions have on possible mission range, the large effect of thrust available and as a consequence the critical importance of the SEP requirement. This requirement, of course, essentially quantifying maximum thrust available minus drag at the actual flight condition, divided by weight and multiplied by forward velocity represents a speed of climb and is not the only maneuverability parameter. Another is certainly the steady state turn rate at any speed. On Figure 7 both SEP and turn rate at M = 0.9 are shown with thrust-to-weight and design take-off weight as parameters. This, however, is generally a less critical requirement. Finally, before leaving this general discussion of the basic design parameters and turning to the question of maximum lift, Figure 8 summarizes the direct effect of the SEP requirement on the design take-off weight with the overload factor required for each of the point designs presented to achieve a given mission radius. Each group of designs was made for a fixed thrust-to-weight ratio. It is clearly seen that for a given design take-off-weight, an increase of SEP obtained by more thrust, carries a strong penalty of overload to obtain the desired mission radius. The optimum overall design can only be developed by iterative adaptation to properly understood overall requirements within the possibilities of available technology, much like the evolutionary process in nature, operating by selection and subsequent survival of the fittest. An interesting example of relevant adaptation in nature are the large vultures of east Africa, a study of the soaring flight of which is published in a recent issue of "Scientific American". A powered soaring plane was used as chase aircraft to follow the birds and observe their flying techniques. These vultures fall into two groups of differing wing loading. The low wing loading type makes use of feeble early morning thermals and soars in these thermals staying in a small area searching for carrion. The requirement here is the ability to circle tightly within the narrow thermal currents. The other type, with about 40 % higher wing loading, require the capability to cover distance using slope lift and wave lift, flying from one thermal to another by gliding over considerable distance in-between, achieving glide ratios better than 60 : 1 by speeding up between thermals. These birds are also an example of variable wing geometry in that they alter wing area and aspect ratio to adapt to the contrary requirements of tight turning or fast gliding.

3.0 Maximum Lift and Buffet Penetration

The flow régime in which maneuvering fighter aircraft operate is characterized by flow separation and extreme non-linearities of the derivatives. This type of flow leads to severe restrictions of the maneuvering capability due to buffetting, loss of lift, asymmetries of behaviour and increase of drag, all of which lead to a loss of controllability. The investigation of these phenomena has received very much attention in the last ten years, because their suppression would considerably improve the maneuvering ability of the vehicle. Various methods have been developed in order to have available wind tunnel techniques which can be used for the prediction of the onset of the critical flow régime and define the increase of severity as the aircraft penetrates into unsteady flow. Figure 9 summarizes the techniques and comments on their usefulness. The most obvious way to determine the existence of flow separations is to judge it by means of the "kinks" that appear on the plots of various aerodynamic derivatives. In the judicious application of this method the type of flow which is characteristic for the particular configuration must be borne in mind i.e. whether it is classical attached flow of high-aspect ratio lifting surfaces, or stable vortex flow of low-aspect ratio shapes. Generally a first "kink" will indicate buffet onset, but for vortex flow this does not necessarily mean that the flow régime has changed. In Figure 11 for 35^o wing sweep reasonable correlation exists between light buffet predictions from such "kinks" and the C_B''-parameter of Mabey which is derived from unsteady wing root bending R.M.S. strain signals. The "kink" compares well with the higher level of C_B'' associated with moderate buffet. But for a 50^o wing sweep case shown on Figure 12, which is associated with vortex flow, the correlation is not nearly as good, in fact for lower Mach numbers the Mabey-method indicates much higher buffet C_L-values than would be assumed from an assessment of the "kinks".

Another method consists of a measurement of static pressures close to the wing trailing edge at various spanwise locations. The divergence is defined by an increase of the pressure coefficient of 0.05, which is taken to correspond to buffet onset and an equivalent value of 0.004 for the C_B''-parameter. Figure 13 shows the correlation obtained which suggests that an appropriate location on the span exists which would match the buffet onset as defined by the Mabey criterion.

Hollingsworth and Cohen have published results which compare the various methods discussed with flight test results for buffet onset. This data is presented on Figure 14 and would tend to show that both trailing edge pressure divergence and wing tip accelerometers agree well with flight tests, but the unsteady wing root bending moment is too conservative and is not well correlated with flight test. However, it must be admitted that none of the methods has such a clear superiority over the others that it could be exclusively relied upon to give reliable results in model tests to quantify the behaviour of the configuration in terms of buffet penetration.

Flight test is still the only sound means of completely defining the unstable flow characteristics of the aircraft, indeed it itself has difficulties of interpretation and correlation of pilot judgements. A certain level of buffet penetration might well make it impossible for the crew to make full use of their instruments and displays and impair the qualities of the aircraft as a weapons aiming platform, while in evasive maneuvres the degree of instability experienced might not prevent the pilot from controlling the aircraft adequately for the purpose.

Using the technique of unsteady wing root bending signals, NASA has made a systematic investigation of the influence of basic wing geometry on buffet onset. This is shown in Figure 15 and Figure 16. In the transsonic region, buffet C_L is improved with decreasing thickness, more rearward location of maximum thickness and increasing camber. Some of these trends, however, are reversed for a lower Mach number e.g. $M = 0.6$. This shows that the design optimum will depend a great deal on the precise requirements of the optimization. This applies even more for the planform effects. Increasing sweep will decrease the buffet onset C_L at subcritical Mach numbers. Similarly aspect ratio must be decreased if maneuverability at Mach numbers greater than 0.8 is desired. An attempt to extend these results to an assessment of buffet penetration is made on Figure 17, which shows the application of the Mabey criterion to a 45^o wing sweep configuration in the subsonic speed range. The considerable margin between buffet onset and C_L max, in particular at lower Mach numbers, and the importance of defining acceptable buffet penetration is apparent. An overall summary of the effect of wing sweep on buffet levels at $M = 0.7$, both in terms of C_L and angle of attack, is given in Figures 18 and 19. The increase of sweep which at this Mach number appears beneficial for increasing the maximum acceptable C_L is accompanied by a corresponding increase of angle of attack, which must always be taken into account for an overall maneuverability assessment in terms of viewability from the cockpit and weapon aiming requirements.

Finally the question of maneuvre devices, both on the leading edge and on the trailing edge, requires consideration. For angles of sweep up to about 45^o, such devices can give a considerable improvement of C_L at moderate buffet. Figure 20 shows, for example, that a leading edge slat can result in an increase of C_L of more than 0.2 over the whole speed range at 45^o wing sweep, while a trailing edge flap will give a similar order of improvement, though its effectivity seems to disappear at Mach numbers above 0.75 or 0.8. In general, such maneuvre devices have in many cases been found attractive, but their application must be decided only when overall systems implications are also taken into consideration. They will increase the complication of the hydraulic and electrical system of the aircraft, with a certain weight penalty, which could offset the gain in buffet-free lift which they produce.

4.0 Maneuverability for TF-Flight

The maneuverability required for terrain following flight is closely associated with the mechanization of the system, both for automatic flight with the autopilot and manual flight with flight director and other monitoring displays. An important aspect is also the performance of the sensors and the probability of false pull-ups and other errors. It is a basic feature of all practicable TF-systems to have sufficient warning time or looking ahead ability, compatible with the maneuvering capability of the aircraft, in order to avoid ground collision either in the case of system failure or degradation of sensor information. Normal terrain following uses only a small part of the vertical g-capability of the aircraft because the crew cannot be subjected constantly to excessive vertical acceleration, especially negative g's. A typical case would be operation of the system between -1g and +2g, but additional authority is required for the system for emergency pull-up, making use of the full g-capability of the aircraft in the pitch axis to avoid collision. Depending on the type of back-up sensor (whether it measures altitude above a point immediately below the aircraft or looks forward) this will limit the minimum clearance after low altitude warning in the emergency case. Simulations are carried out to obtain an understanding of system behaviour and assess the probability of crashing when specified malfunction occur during flight over typical types of terrain.

During automatic TF-flight heading and track acquire autopilot modes are in use and, therefore, the lateral axes maneuverability also becomes important. For navigation flight the roll rates and turning rates are, however, modest compared to the case of target acquisition, where the aircraft may have to maneuvre violently to correct a misalignment between itself and the target. Lateral maneuverability is limited by the following factors:

Autopilot computations which reflect into the pitch axis because vertical acceleration demands from the TF-computer must be converted into normal acceleration demands, lift compensation in turning flight must be provided and turn coordination is necessary in order to minimize side forces during turn entry, steady state turn and turn exit. In addition, a pitch priority system is required for the turn rate to ensure priority of the terrain clearance demands over the navigation demands.

Sensor problems, which are related to the probability of radar altimeter unlock at high bank angles, the monitoring of altitude for g-sensitive erection mechanisms, e.g. gyros in turning flight and errors in barometric information due to the sensitivity of the pitot-static sensor to angle of attack and sideslip. Further aspects of the problem relate to the radar, which must allow adequate warning time during execution of the turn. The antenna scan pattern is changed to rotate it into the turn and the antenna must be stabilized in roll up to maximum bank angles.

In summary, the maneuverability requirements for automatic terrain following are predominantly either emergency cases of pull-up to avoid ground collision when a failure has occurred, this limiting the minimum safe height at which TF-flying can be carried out, or the case of target acquisition, when the aircraft has to carry out strong lateral maneuvres, while simultaneously requiring a priority for vertical maneuvre for obstacle clearance. Typically turn rates of 3°/sec with roll rates at 40°/sec within bank limits at $\pm 45^\circ$ or $\pm 60^\circ$ will occur with TF-radar scan coverage looking into the turn, giving warning times of about 10 seconds. Both of these requirements are shown on Figure 21, which plots the kinematic relationship $\dot{\chi} = \frac{g}{V_T} \tan \varphi$ and simulation results of the maximum value of pull-up g required for certain minimum clearance heights, when a low altitude warning signal gives rise to automatic pull-up within a representative range of flight speeds for TF-flight.

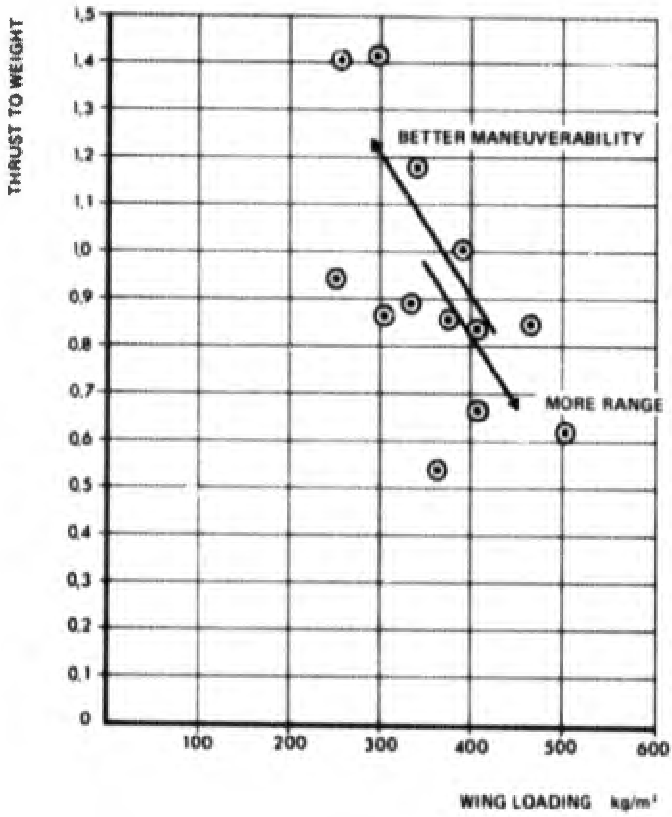
5.0 Maneuverability for Air Combat

The question of maneuverability requirements for air combat and, indeed, an understanding of the basic features of such maneuvres to be able to successfully design around such a requirement, are the subject of much discussion at the present time. In the sixties it became usual to define SEP and regard it as the key to overall maneuverability. Of course the other, older parameters, such as steady-state and dynamic load factor achievable and turn rate, were still of interest, but it was difficult to use them to compare designs or aircraft to decide which was the better maneuvering vehicle. For example, Figure 22 compares the maximum dynamic load factor of two fighter aircraft at comparable combat weights at sea-level and at medium altitude. "Comparable combat weight" here means that both aircraft, when engaging in combat, have sufficient fuel available to fly the same distance home to base. The one aircraft, while having a slightly lower structural limit, has a slight edge on the other in terms of the maximum load factor it can develop at any given speed. When this aircraft, however, is considered with full internal fuel, its advantage disappears. Which of the two is preferable, in view of the fact that in a practical case the distance away from the base will not be the same? Furthermore, it is not at all clear whether the load factor available is really the one and only decisive factor in combat. Figure 23 presents a chart of specific excess power of a combat aircraft at a low altitude with lift limitation and structural limit. A large part of the chart shows negative SEP-values, which mean that only dynamic flight states are possible in that area. Again two aircraft can be compared on the basis of a large number of such charts for various altitudes and combat configurations but in a practical case it will be found that one is superior in certain cases, while the other is better in other cases. How to decide which combination of requirements (altitudes, Mach numbers, positive or negative regions of the SEP chart) is relevant for success in the air combat? Computer simulations have been attempted to program the encounter and to decide this question by direct confrontation. Such computer studies either program the whole encounter by using a presumed model of air combat behaviour or, more ambitiously, aircraft simulators with visual or even motion representation programmed with the aircraft characteristics are used by human pilots to simulate the combat. The former are much appropriate for preliminary design investigations, while the latter are presently hardly available in Europe and represent a degree of sophistication which is very expensive. Indeed, all such simulations have not yet produced universally accepted results, if thereby is meant a complete answer to the air combat problem in terms of a precise definition of the priority of its requirements. A typical computer simulation result is shown on Figure 24. Here a chart of SEP at low altitude is used to plot the maneuvres as produced by the simulation of two opposing aircraft engaged in air combat, according to rules set for the encounter. Typically, the aircraft engage at high speed in steady state flight and both quickly pull high g's entering the dynamic region, progressively lose speed and tend to approach the C_L max limit, maneuvering there at a speed low enough not to lose too much altitude, because they are close to the ground already. The SEP as such is apparently not the most important characteristic, but rather low speed load factor and extreme tightness of turn at that speed. The result, therefore, just means that the lowest possible wing loading is the most desireable feature. Now this may not be an unexpected result, but it can hardly be said that it improves our insight into the real problem very much. Of course the difficulty of such simulations is that they produce a foregone conclusion, inasmuch as the encounter is programmed according to certain definite principles. The non-predetermined aspect of the combat situation is difficult to introduce. Most fighter pilots agree that the initial conditions of the encounter, namely the surprise element due either to luck or superior vision from the cockpit, the effectivity of warning devices and pilot technique and level of training and experience play a large part.

6.0 Conclusions for Maneuverability in Combat Aircraft

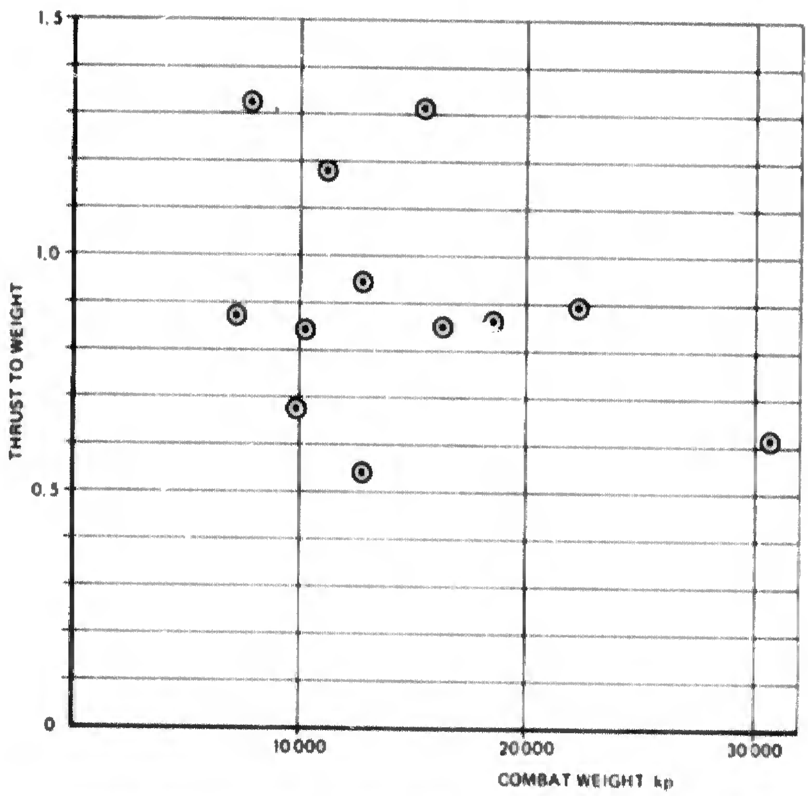
It appears, that it is not yet possible to define the maneuverability requirements for combat aircraft in various mission applications in so clear a manner, that the basic design trade between maneuverability on the one hand and range / pay-load performance on the other, can be handled in any other way then to respond with a design of overall flexibility adapted to the operational requirements of the user in their totality. Any exaggerated emphasis on any one parameter can lead to severe penalties in other just as important aspects of a balanced design. It still remains very true that a useful combat aircraft will result only with such a flexible approach. The military user has a tendency to shift his emphasis from one such aspect to another under the influence and pressure of events. There is a certain swing of the pendulum effect in this. The basic design constraints are summarized in Figure 25, which clearly shows the danger of going too much to one extreme. None the less, it is obvious that any combat aircraft design will in fact tend more in the one or in the other direction. To combine the best of all worlds in one design is not feasible, a truth which is hardly surprising to engineers.

All the choices enumerated on Figure 25 have considerable cost implications. In same cases this applies more to the R and D costs involved, in others there is, in addition, a significant increase of procurement cost of the production aircraft. In both cases this merits serious attention, since R and D costs of new combat aircraft programs can well amount to the equivalent of a 100 or even 200 production aircraft procurement. Some indication of such cost effects of combat aircraft size and thrust-to-weight ratio, two very relevant parameters as we have seen, is illustrated on Figure 26. The fly-away cost of production aircraft is shown in hypothetical cost units, indicating a reduction of about a third when going from the most expensive to the least expensive aircraft. This data is taken from a study in which a response to an essentially defined requirement was made. Design to cost is becoming more and more the order of the day, however difficult it might be to treat cost as just another parameter of the engineering problem. But this approach does tend to favour the balanced design and thus makes a salutary contribution.



TWIN AND SINGLE ENGINE FIGHTER AIRCRAFT PRESENTLY OPERATIONAL OR EXPECTED TO BE OPERATIONAL IN THE SEVENTEES.

Fig.1 Thrust and wing loading



TWIN AND SINGLE ENGINE FIGHTER AIRCRAFT PRESENTLY OPERATIONAL OR ESPECTED TO USE OPERATIONAL IN THE SEVENTEES

Fig.2 Thrust and combat weight

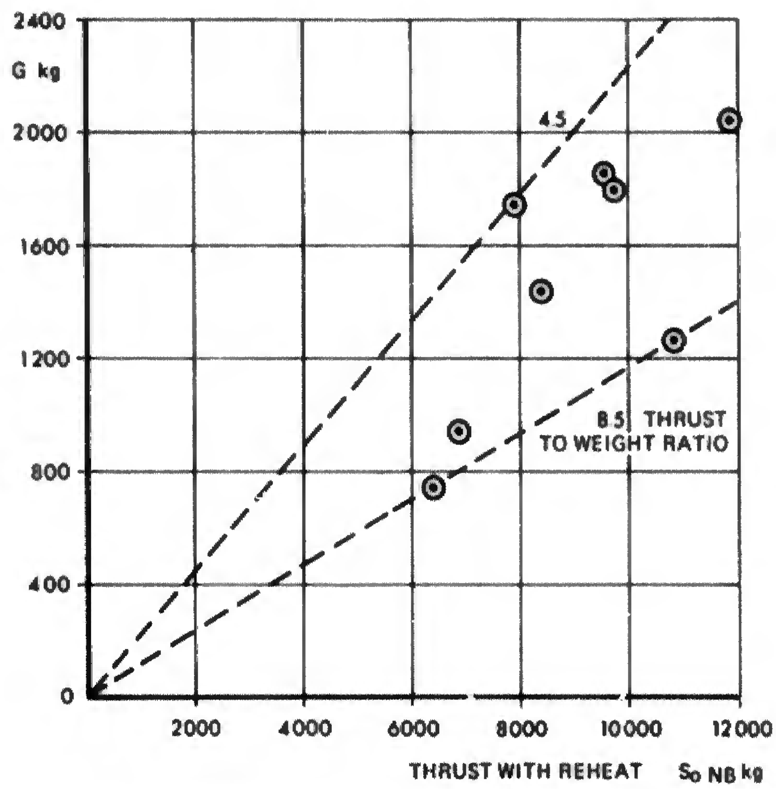


Fig.3 Capability of operational engines in the seventies

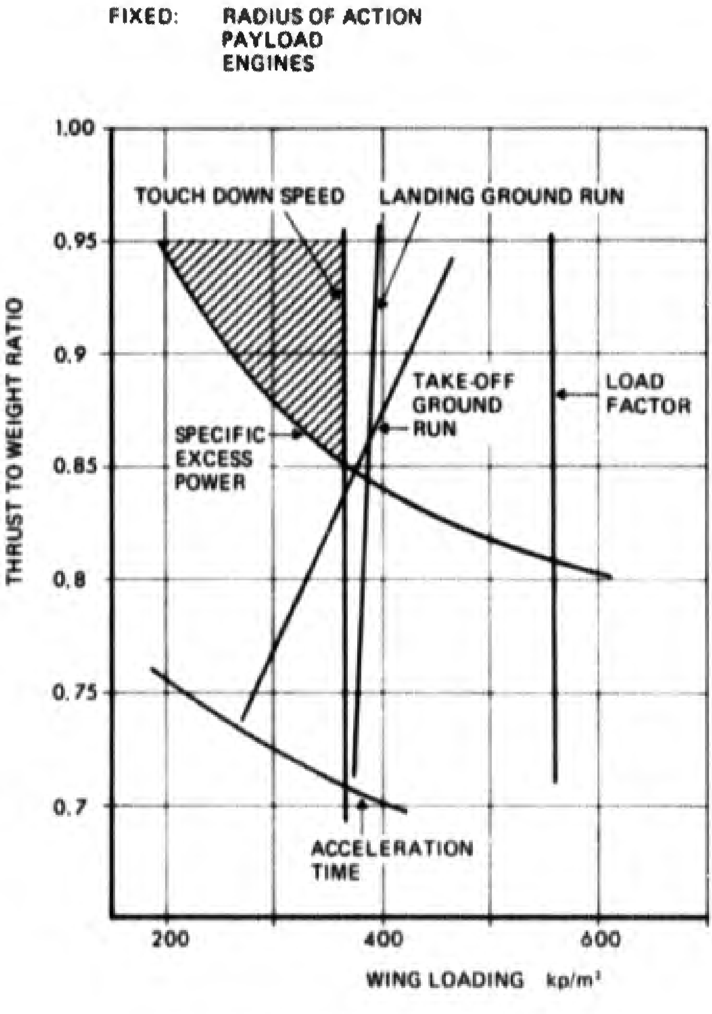


Fig.4 Overall design trades

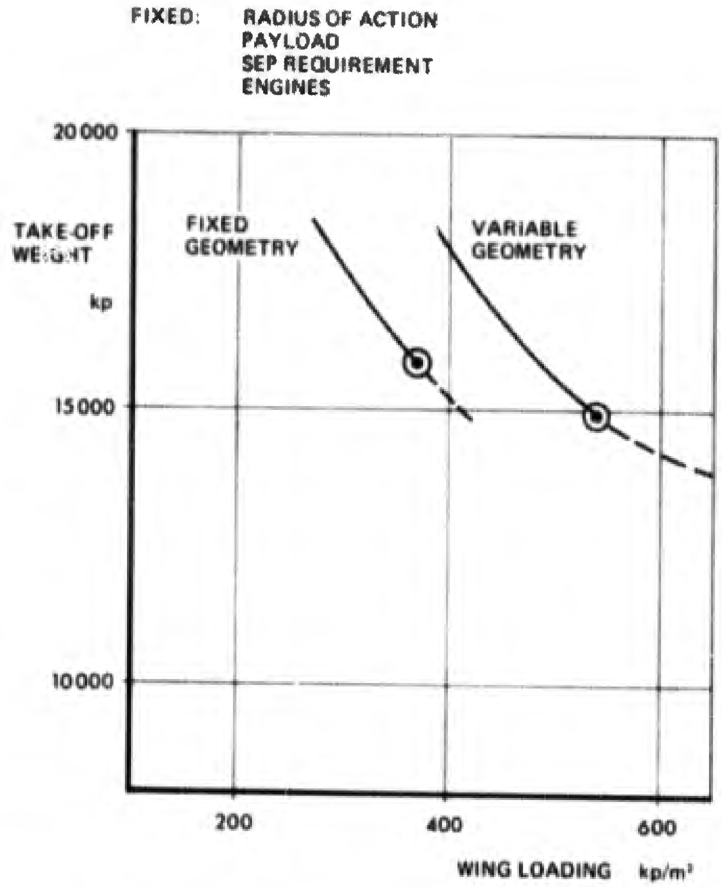


Fig.5 Take-off weight affected by wing loading

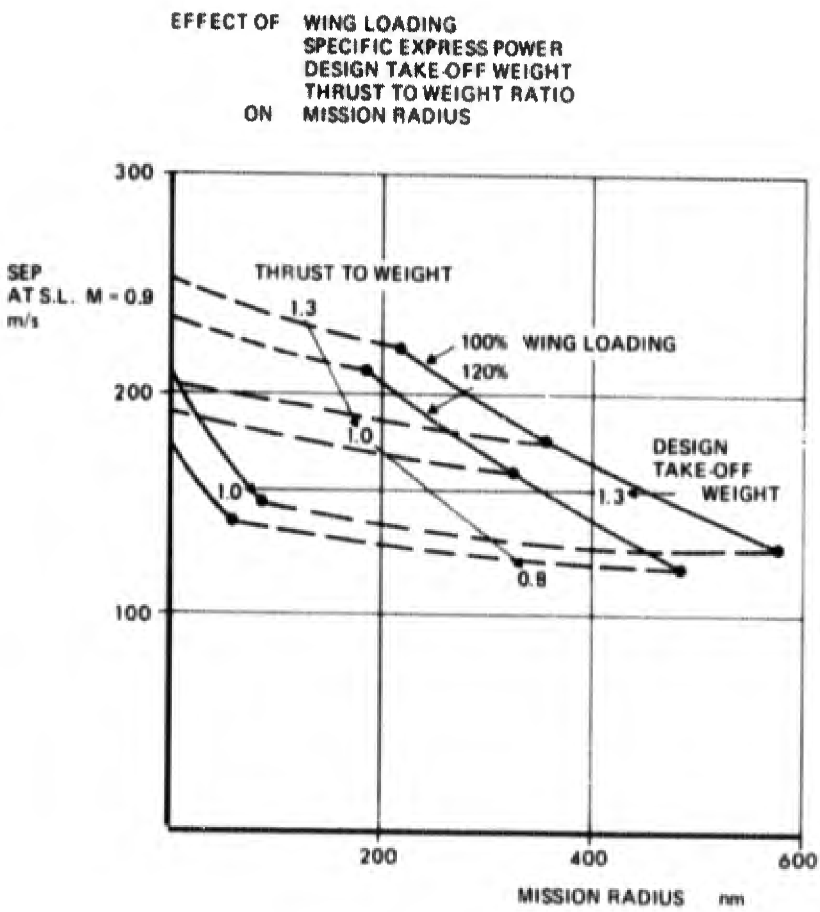


Fig.6 Combat aircraft in 15000 kp class

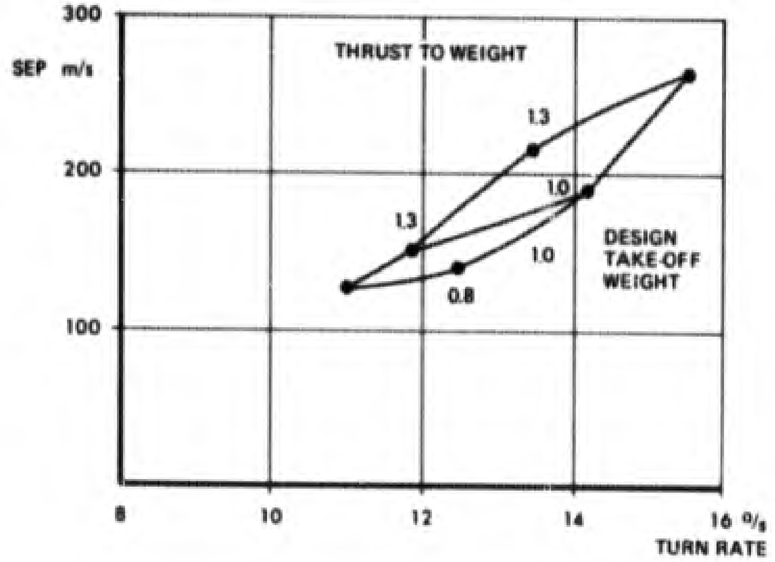


Fig.7 Specific excess power and turn rate at M = 0.9

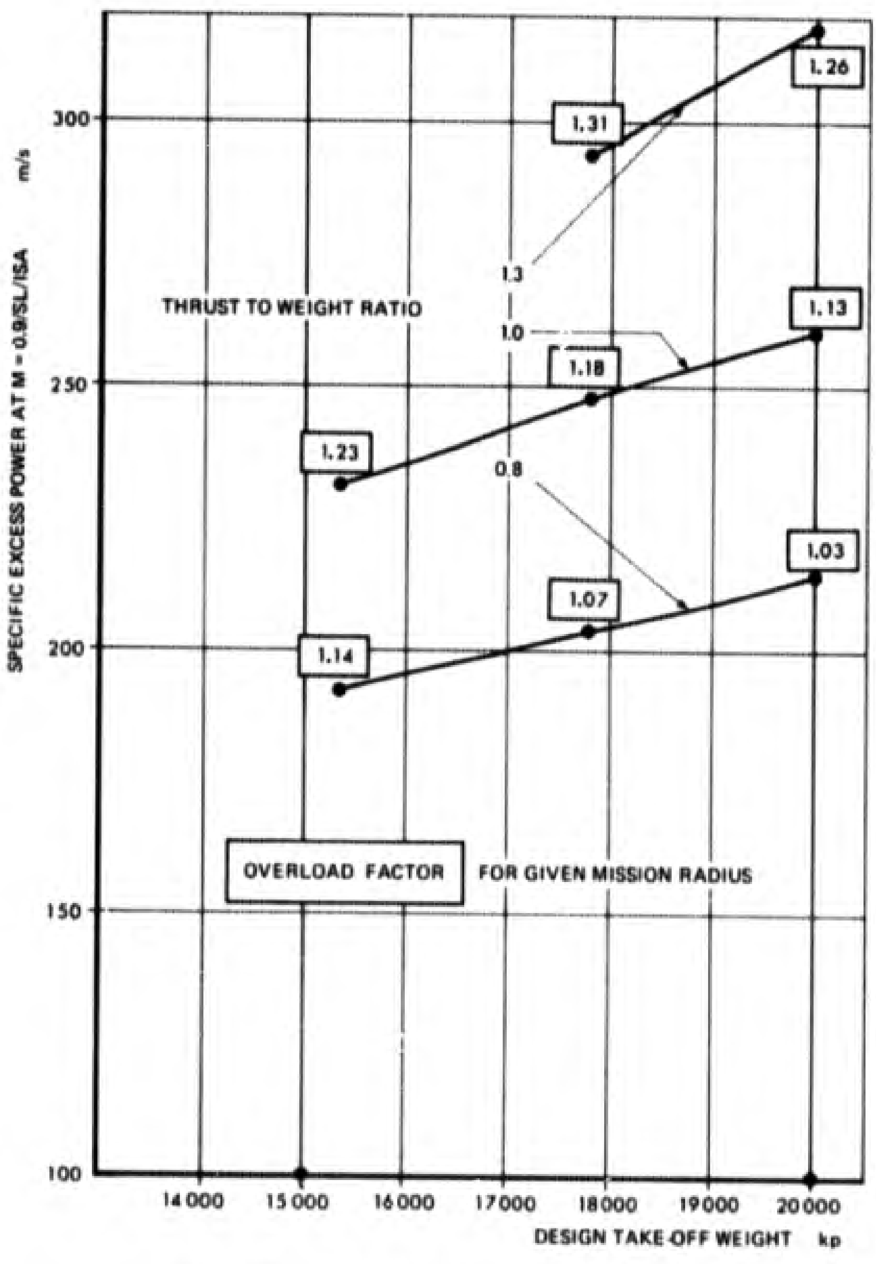


Fig.8 Effect of SEP requirement on take-off weight

METHOD	TECHNIQUES	DEFINITION	COMMENTS
KINKS	6-COMPONENT-BALANCE MODEL RESULTS	$C_L = f(\alpha)$ $C_m = f(\alpha, C_L)$ $C_A = f(C_L)$ $C_l = f(\alpha)$	DETERMINATION OF BUFFET ONSET POSSIBLE NO QUALIFICATION OF BUFFET PENETRATION
TRAILING EDGE PRESSURE DIVERGENCE	STATIC PRESSURES CLOSE TO WING TRAILING EDGE	$C_{pTE} = f(\alpha, C_L)$ $\Delta C_{pTE} = 0.05$	DETERMINATION OF BUFFET ONSET NO QUALIFICATION OF BUFFET PENETRATION
WING TIP ACCELERATION	ACCELEROMETER MOUNTED AT WING TIP	$g\,(RMS) = f(\alpha)$	DETERMINATION OF BUFFET ONSET/PENETRATION
UNSTEADY WING ROOT BENDING MOMENTS (MABEY)	WING ROOT STRAIN GAUGES	$C_{WRB} = f(\alpha)$ (RMS) UNSTEADY WING ROOT BENDING MOMENTS	DETERMINATION OF BUFFET ONSET, NO QUALIFICATION OF BUFFET PENETRATION POSSIBLE
		$C_B^* = f(\alpha)$ (RMS) CORRECTED BUFFETING COEFFICIENT	DETERMINATION OF BUFFET PENETRATION USEFUL METHOD FOR INVESTIGATIONS ON WIND TUNNEL MODELS DEFINITION OF A/C PENETRATION LEVELS PROBLEMATIC

Fig.9 Methods to determine buffet characteristics

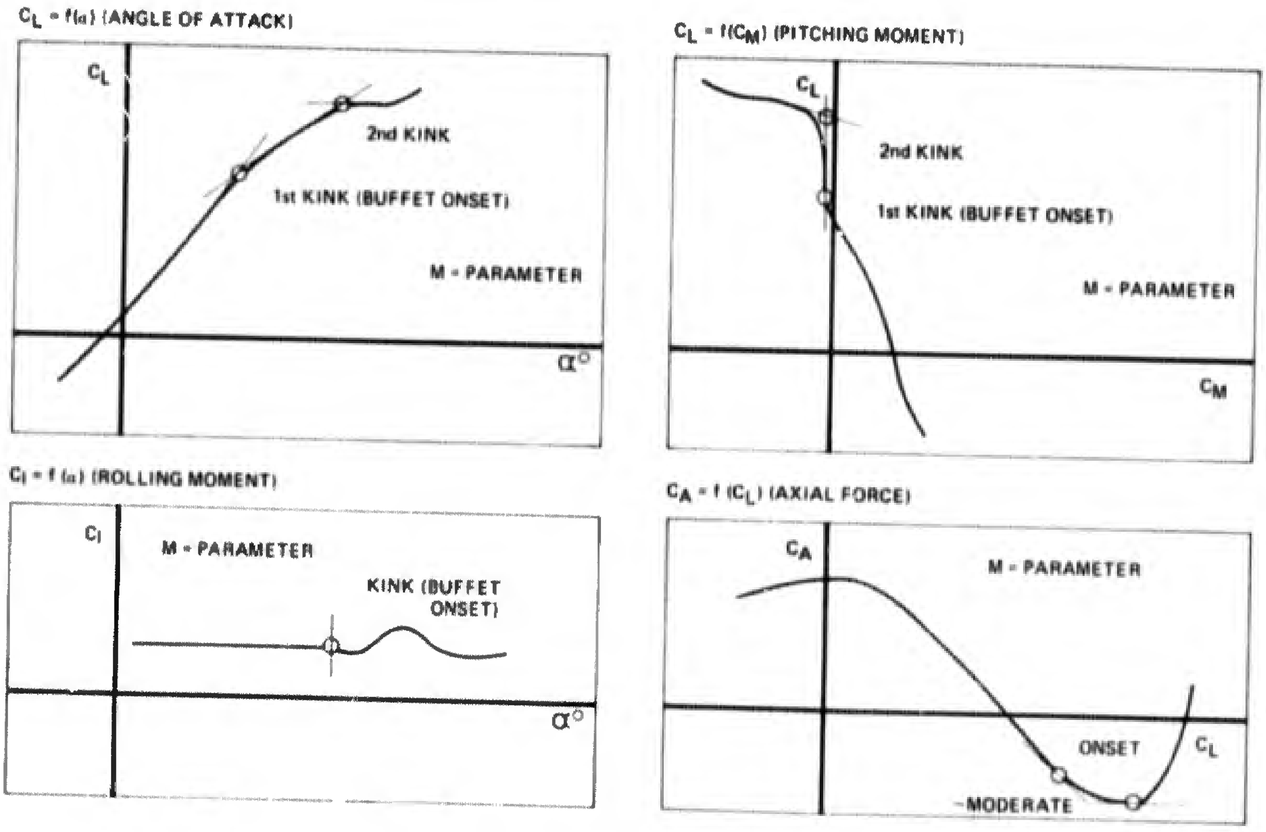


Fig.10 Definition of kinks

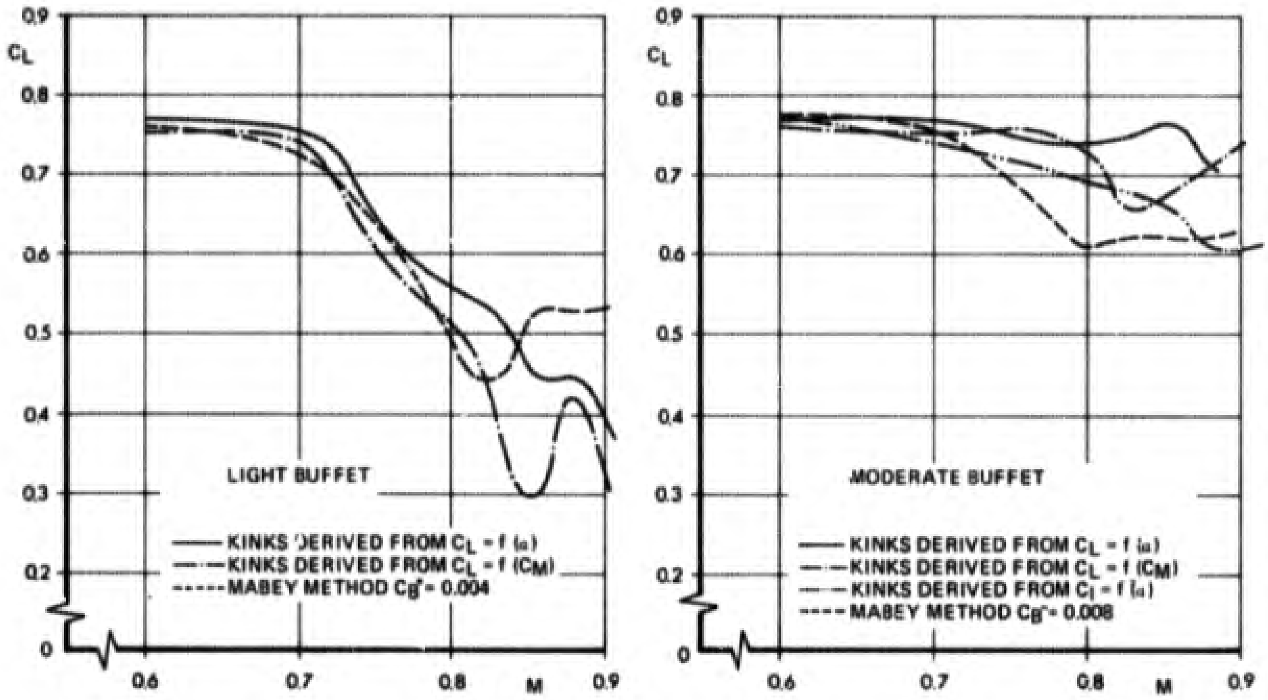


Fig.11 Correlation of unsteady wing root bending signals with data derived from kinks for 35° wing sweep

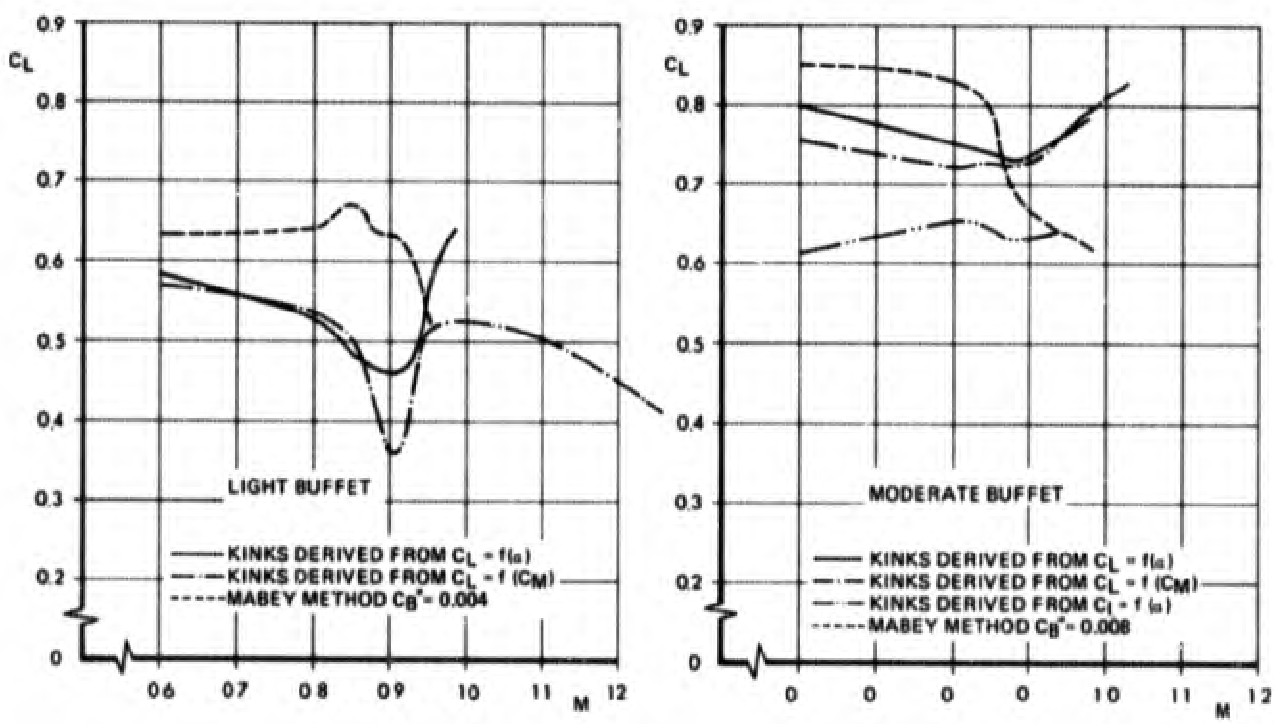


Fig.12 Correlation of unsteady wing root bending signals with data derived from kinks for 50° wing sweep

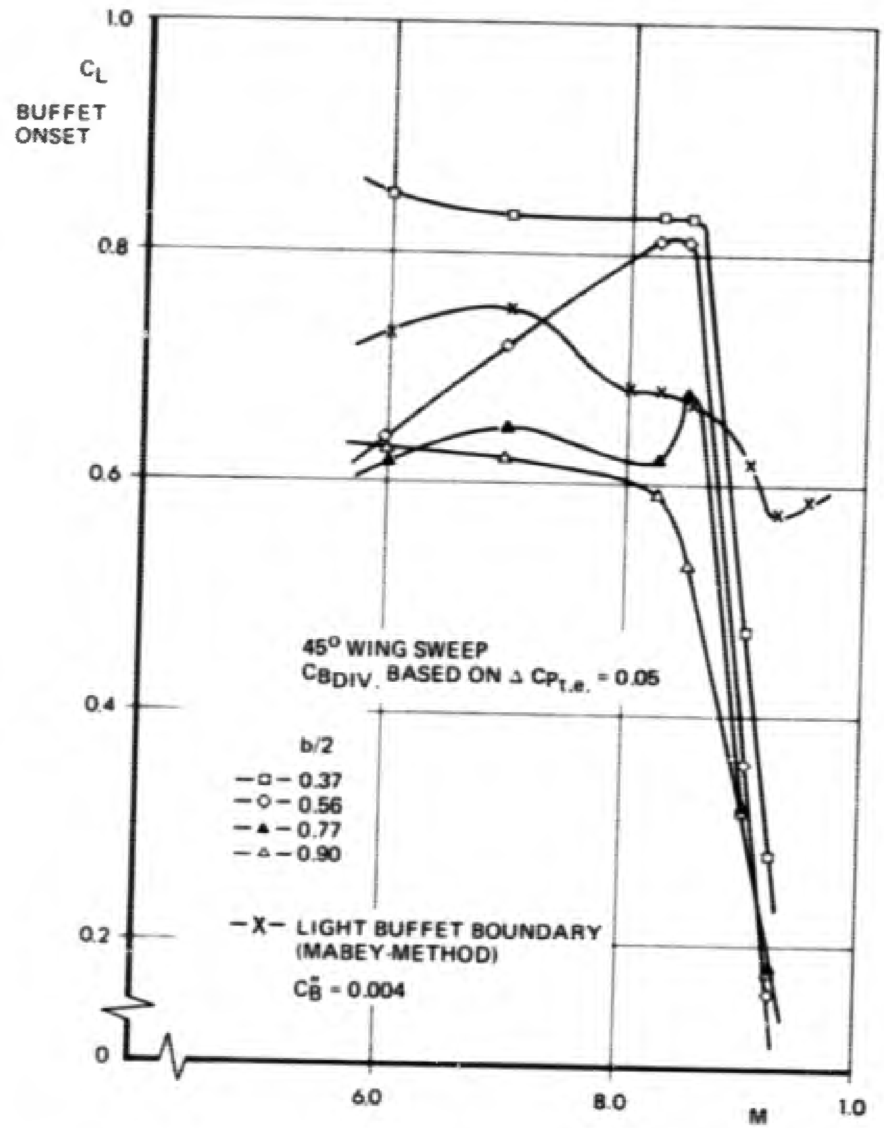


Fig.13 Correlation of trailing edge pressure divergence with unsteady wing root bending signal

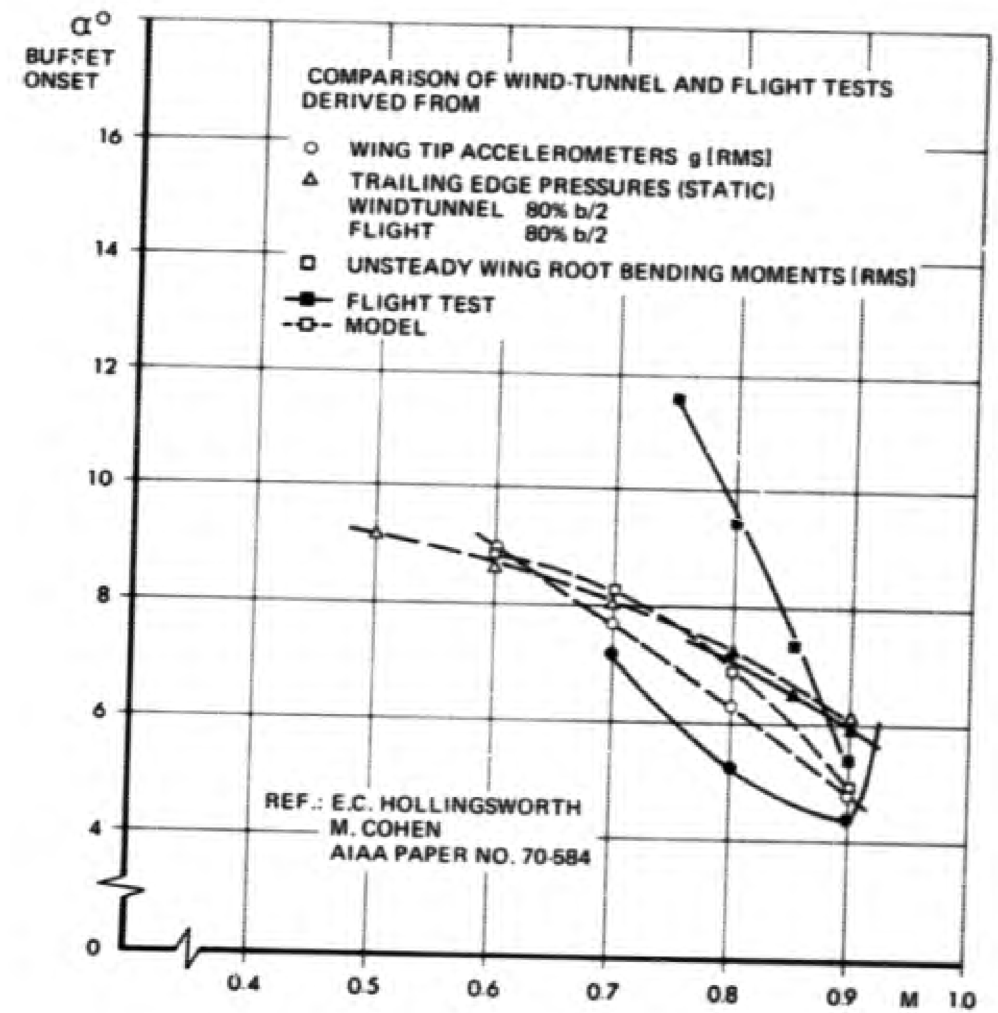


Fig.14 Buffet onset

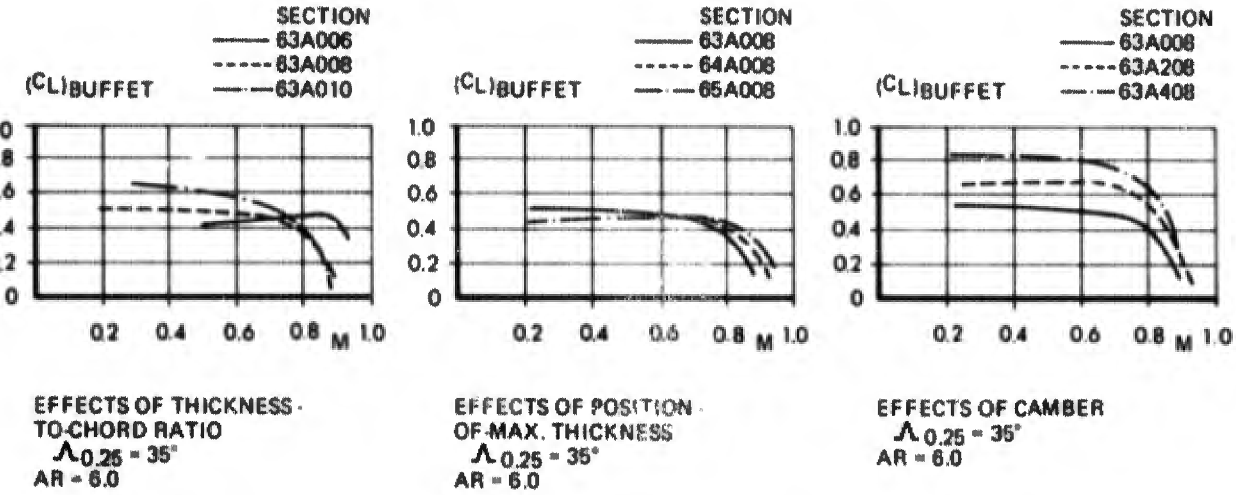


Fig.15 Section characteristics
Basey on unsteady wing root bending signal
NASA TN D-5805

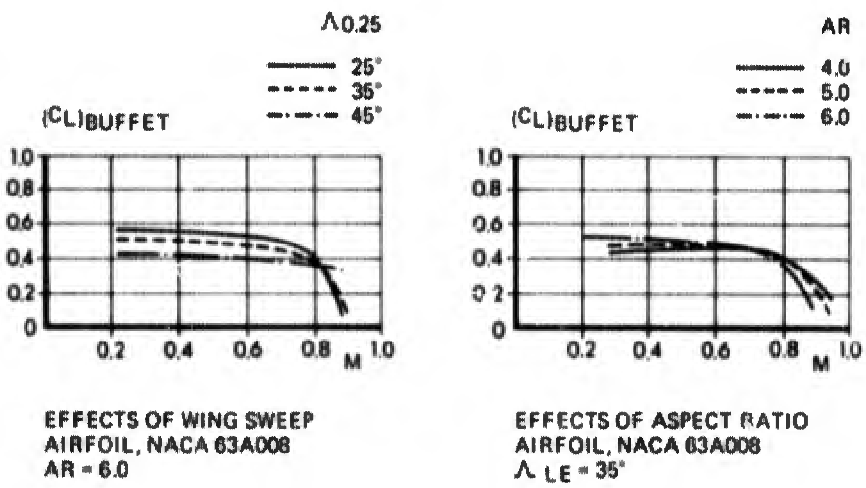


Fig.16 Planform effects
Basey on unsteady wing root bending signal
NASA TN D-5805

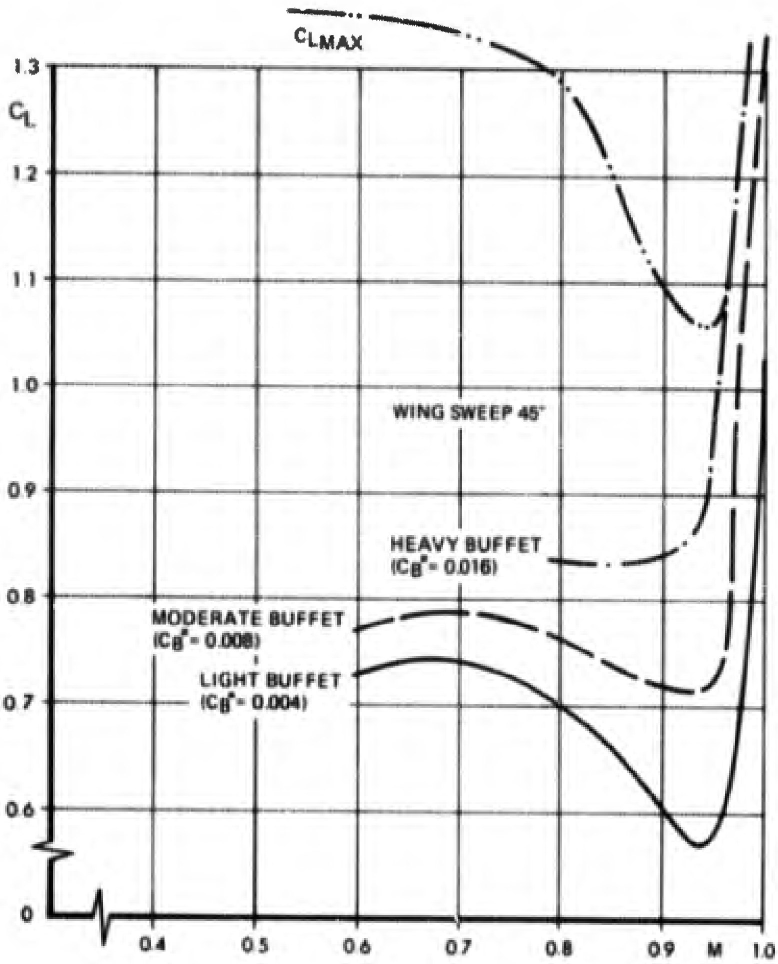


Fig.17 Effect of Mach-number on buffeting penetration and $C_{L_{MAX}}$

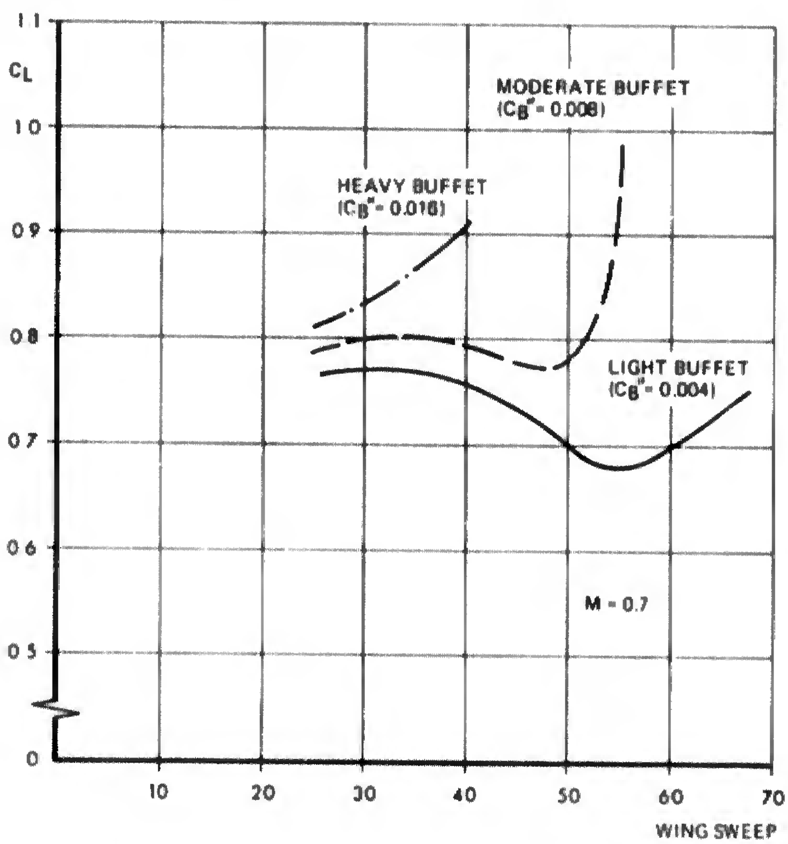


Fig.18 Effect of wing sweep on buffeting penetration

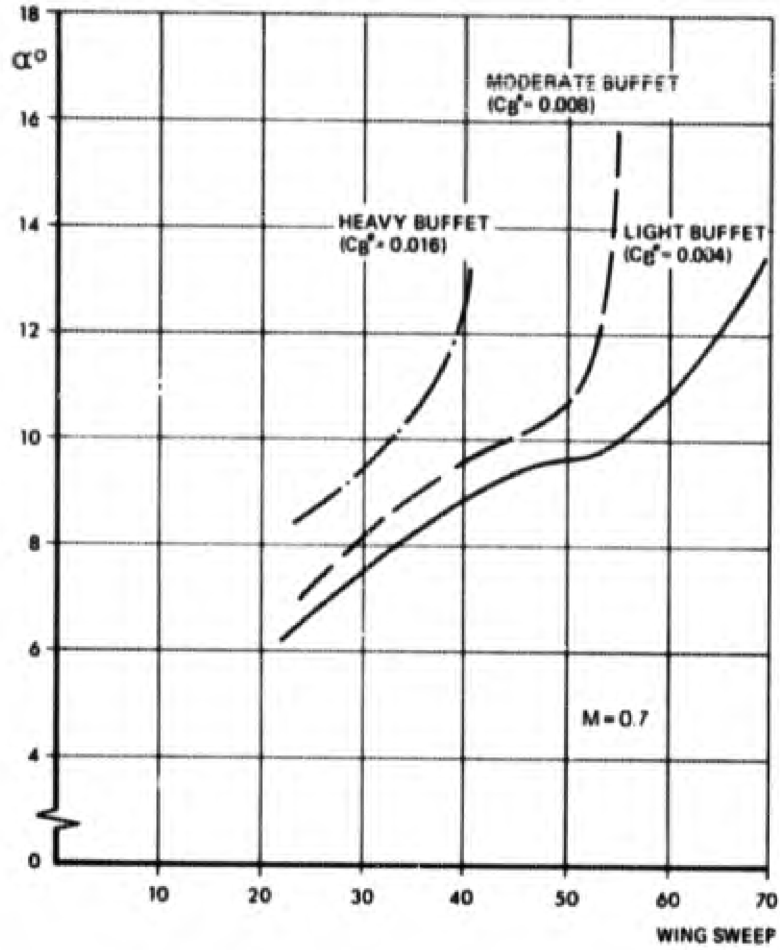
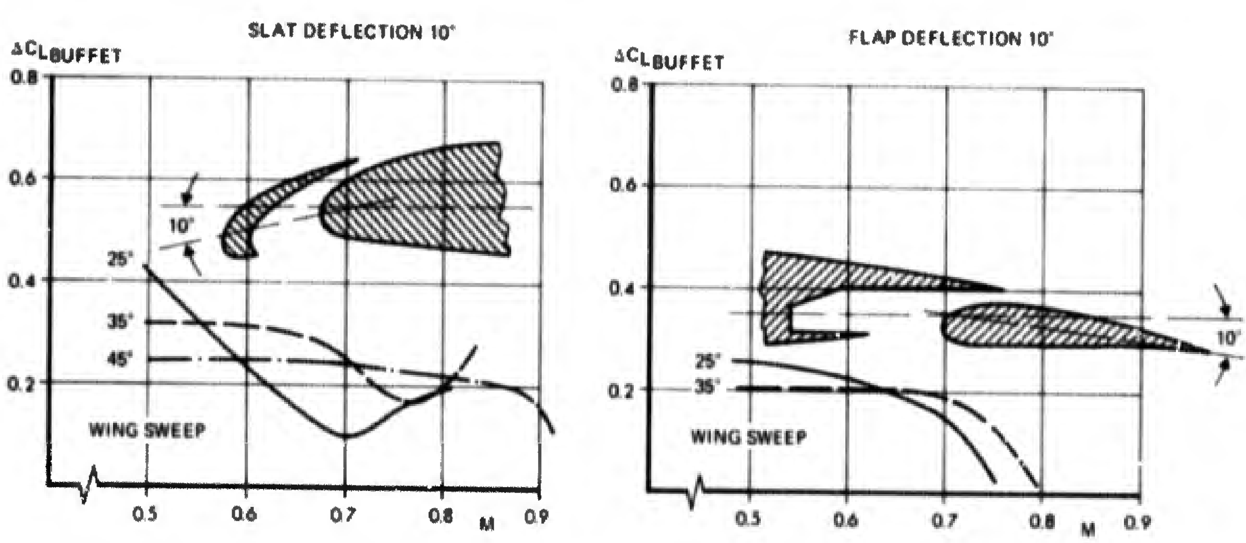


Fig.19 Effect of wing sweep on buffeting penetration



Effect of leading edge slats on moderate buffet

Effect of slotted trailing edge flap on moderate buffet

Figure 20

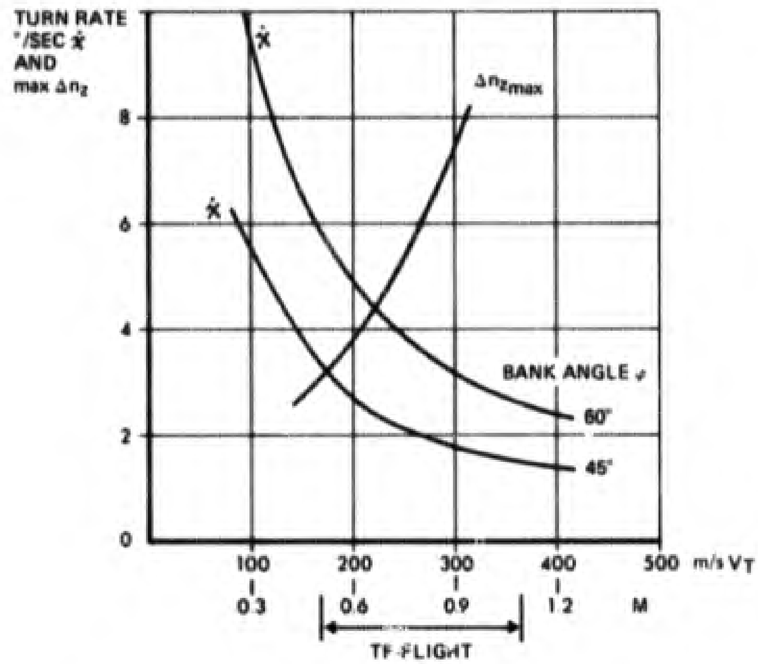


Fig.21 Maneuverability for TF-flight turn rate during target acquisition and maximum pull-up g after hard-over system failure

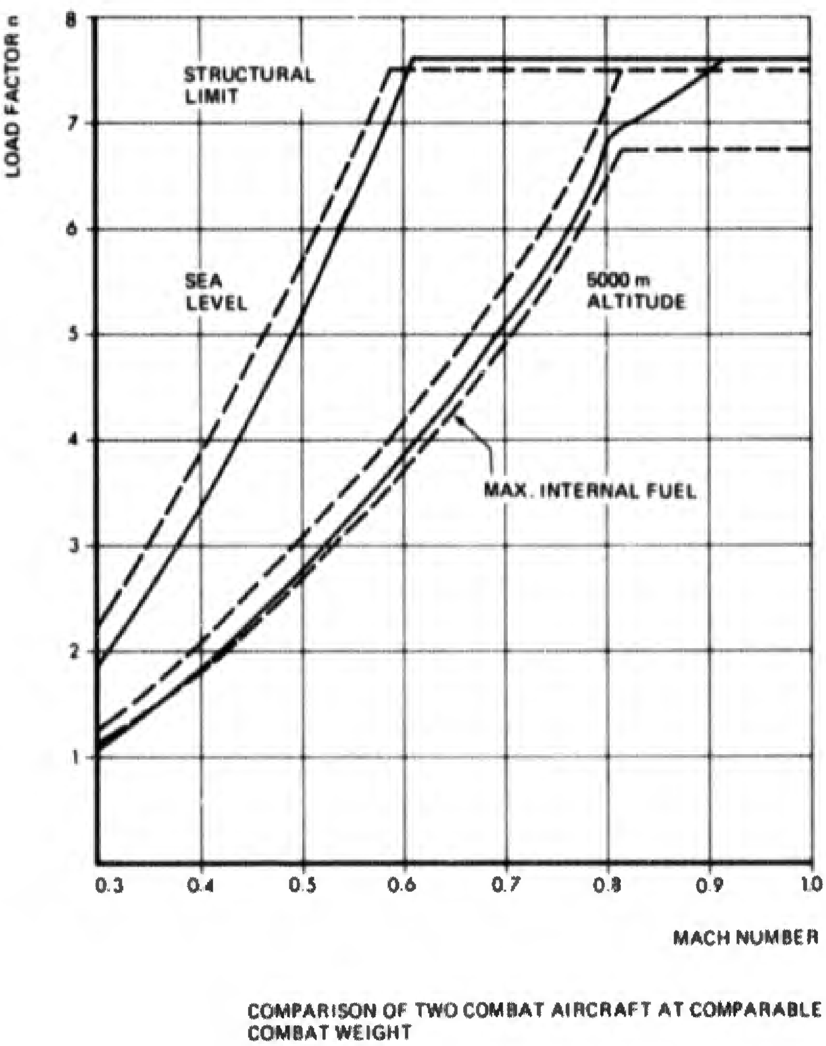


Fig.22 Maximum dynamic load factor

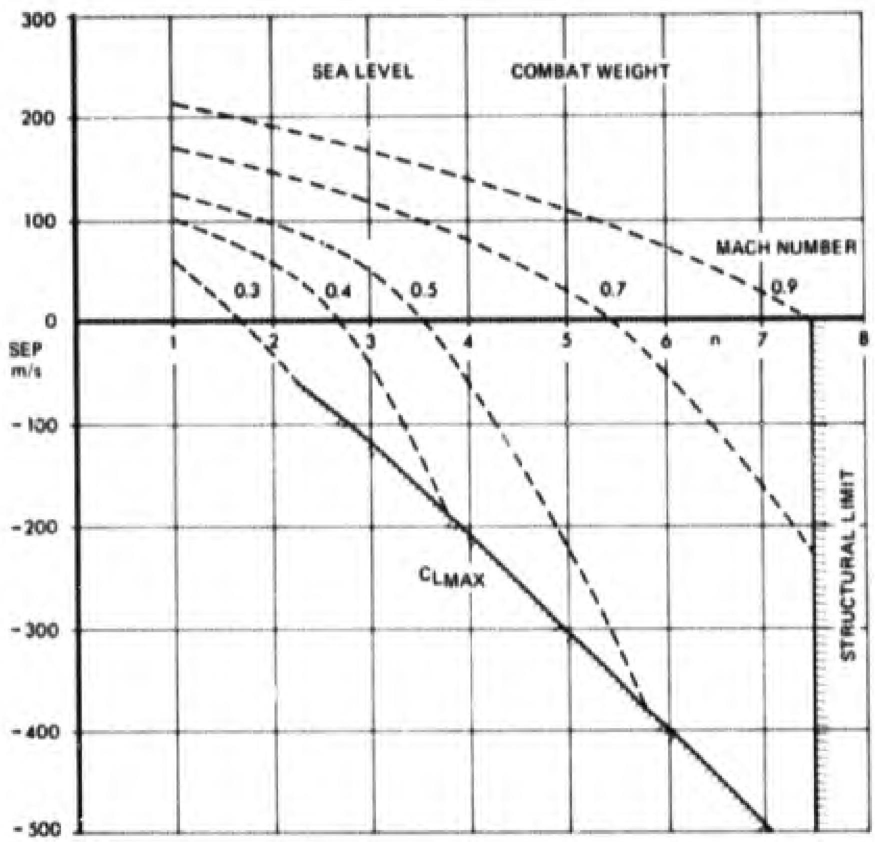


Fig.23 Specific excess power of combat aircraft

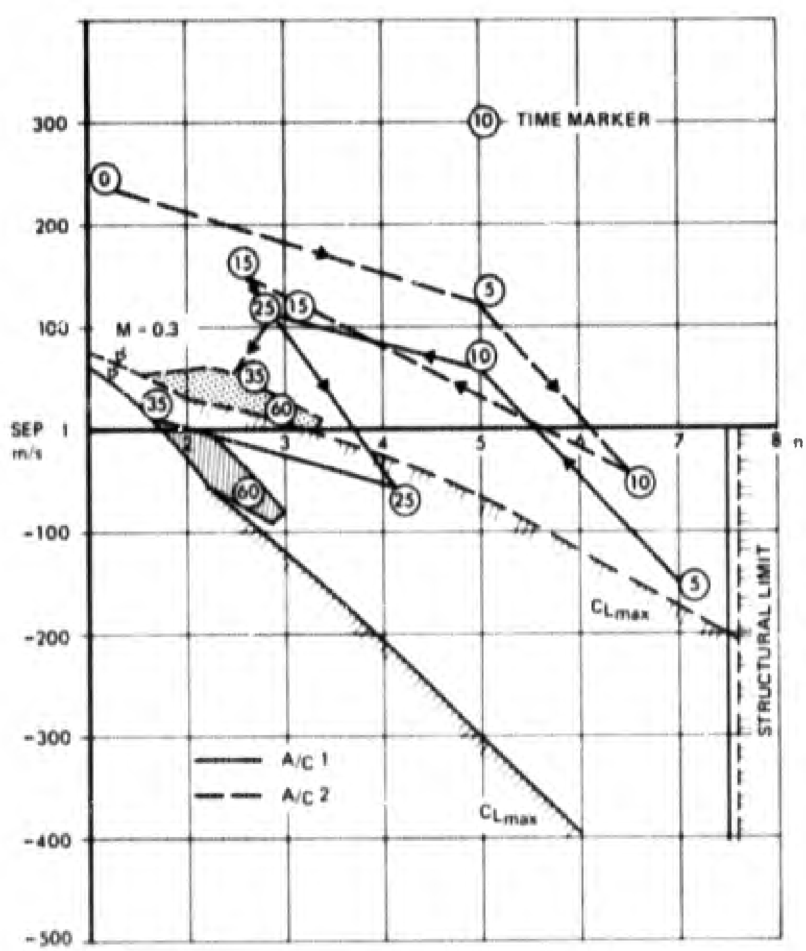


Fig.24 Computer simulation of air combat at low altitude

DESIGN PARAMETER	MANEUVERABILITY FOR AIR COMBAT	RANGE/PAYLOAD PERFORMANCE ATTACK OF GROUND TARGETS
INSTALLED THRUST	HIGH	LOW TO MEDIUM
ENGINE CYCLE (RATIO OF REHEAT THRUST TO DRY THRUST)	HIGH DRY THRUST	HIGH REHEAT THRUST
WING LOADING	LOW	HIGH
MANEUVRE DEVICES AIR BRAKES	SOPHISTICATED	REQUIRED
LOAD FACTOR	HIGH	MEDIUM
AIRCRAFT SIZE	SMALL TO MEDIUM	MEDIUM TO LARGE
VISIBILITY FROM COCKPIT	VERY IMPORTANT	NO PENALTY JUSTIFIED
AREAS OF SPECIAL SOPHISTICATION	CONTROL RESPONSE CREW TOLERATION OF g LOADING	LOW LEVEL PENETRATION

Fig.25 Maneuverability in combat aircraft

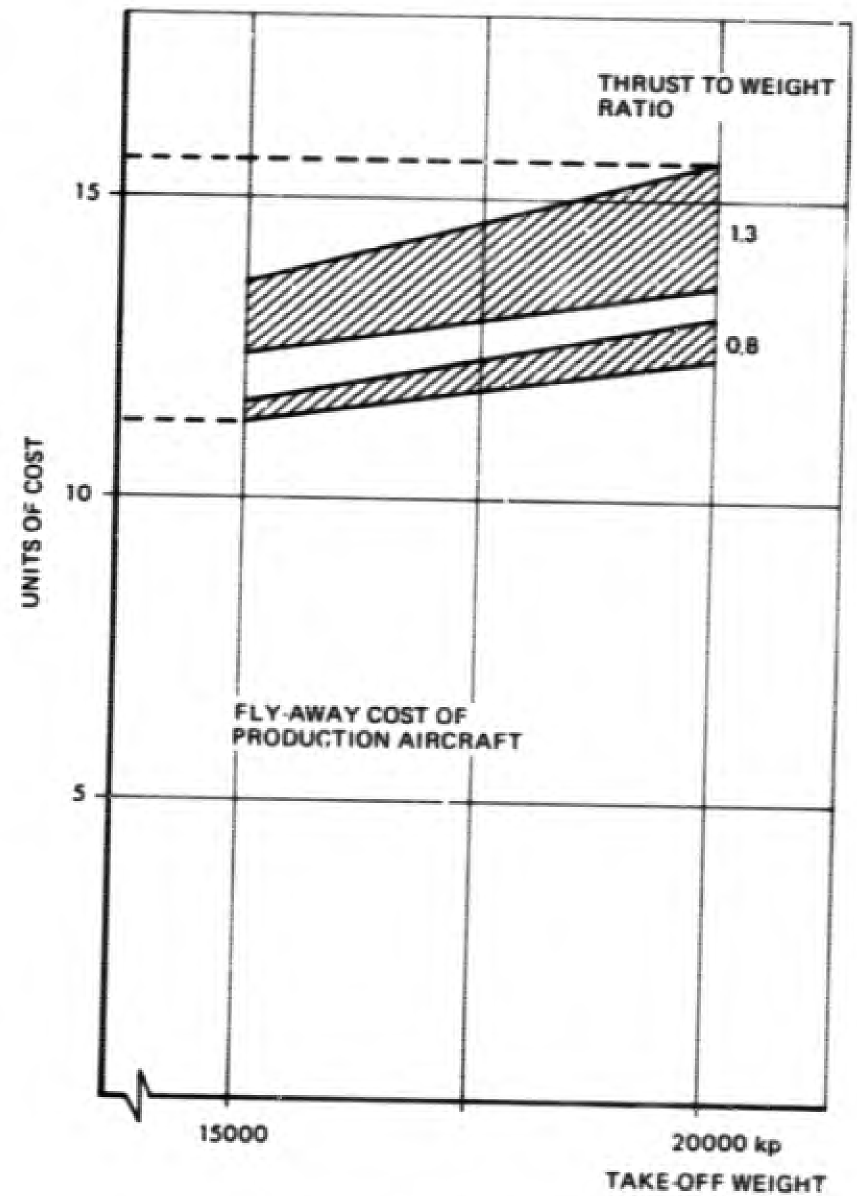


Fig.26 Combat aircraft cost

MODERN ENGINEERING METHODS IN AIRCRAFT PRELIMINARY DESIGN

by

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SUMMARY

This lecture addresses the impact of computer technology on modern engineering methods used in preliminary design. Computerized design synthesis programs are usually constructed in a modular fashion. As preliminary design follows program evolution from tentative requirements through release of a type specification, data resources consistent with the program stage are input to the computer modules. Application of Vought's Aircraft Synthesis and Analysis Program, ASAP, to the early Program Definition Phase of preliminary design is discussed in detail. When requirements are confirmed, Concept Formulation begins. Synthesis and analysis activities accept new sources of data such as developmental test results and especially design manned simulator programs. During Contract Definition preliminary design objectives are directed to optimization for high value specification criteria. A typical air superiority fighter program is selected to illustrate the lecture theme. Movie film strips intersperse the lecture to graphically portray these new tools of the trade. Nondimensional technical material is frequently used and certain artistic license is taken for reasons of security. Methods discussed and engineering tools described are factual.

LIST OF SYMBOLS

W/S	Wing loading, pounds per square foot
T/W	Thrust-to-weight ratio
L/D	Lift-drag ratio
RCS	Radar cross-section
REF	The basic value chosen for a parameter or constraint
REF $+\Delta$	The positive variation in a parameter from REF
REF $-\Delta$	The negative variation in a parameter from REF
$C_L \times S$	The product of lift coefficient times wing area
AR	Wing aspect ratio
P_S	Specific excess power, feet per second
ΔP_S	The difference in specific excess power between two adversary fighters
$\dot{\theta}_H$	Horizontal equilibrium (sustained) turn rate, degrees per second
ASAP	Aircraft Synthesis and Analysis Program
ACS	Air Combat Simulator
LAMBS	Large Amplitude Moving Base Simulator

1. INTRODUCTION

Computerized aircraft design synthesis methods have been under serious development in the United States for a decade. Large capacity digital computer systems and their peripheral equipments provide the opportunity to bring all significant technical and managerial disciplines together in an integrated program. The versatility of such a program is required in today's competitive market. Examples of aerospace companies in America who rely heavily on computerized design synthesis are shown in Figure 1. It is not the purpose of this lecture to describe the development of Vought Systems Division's program. Those who may be interested in the details of Vought's Aircraft Synthesis and Analysis Program (ASAP) are referred to Reference (1).

Most design synthesis computer programs are architecturally constructed in a modular fashion. That is, the integrated whole is made up of modular subsets that are controlled by the technical disciplines responsible for the state-of-the-art in each particular area. In general, these disciplines will apply their input data at a level of detail consistent with the phase of the preliminary design process. Referring to Figure 2, the phases of preliminary design are described in ascending order of detail as: (1) Program Definition; (2) Concept Formulation; and, (3) Contract Definition.

Given a correctly developed synthesis program, technical disciplines can input their submodules with the best data available at any phase of preliminary design. It is a rewarding experience to observe the preliminary design process unfolding; first from empirical estimates that search for designs that approximately satisfy early requirements, to the trade-off phase that seeks to refine the design and help firm up requirements, and finally through the optimization phase where a competitive edge is being sought. My purpose in this lecture is to share with you this process as it took place recently at my company.

Primary modules of Vought's ASAP are shown in Figure 3. Many papers have been written on the theory of constructing and relating Analysis and Optimization modules. How the process is started, Initiation, is a matter of technical expediency; it can be integral with the program or conducted external to the program. Program Control is merely the executive routine that interfaces the other modules. The man/machine interface module is the key to the degree of success achieved in a preliminary design program. This is the primary avenue whereby technical management influences the results with decisions affecting the degree of innovation sought, the accuracy and depth of input data and the means for generating high confidence in the minds of the customer. As we review the preliminary design process that was conducted on a typical air superiority fighter an overview perspective can best be retained if you see the examples from the point of view of the project engineer exerting his influence through the man/machine interface function.

2. PROGRAM DEFINITION AND THE INITIATION PROCESS

Mr. Weissman has reviewed how design and operational experience are used to formulate Tentative Operational Requirements. For the air superiority fighter under discussion principal requirements are summarized in Figure 4. Simplicity and low cost were driving goals; however, it was firmly believed that a high degree of innovation and outstanding agility could be achieved within these goals. Small size, a single design mission and highly selective use of new and emerging technologies were the keys to success. Considering each of these tentative requirements, it seemed that the most challenging one was the desire to obtain a high degree of maneuvering agility at supersonic speeds. If this could be done, there would exist the potential of extending the air combat arena into the supersonic regime.

Many techniques can be used to develop the initial layout. In this case, a search of statistical data was used to establish parameters that are the initial inputs to the configuration module of ASAP. Figure 5 shows several examples of statistically correlated data we used to initiate the preliminary design process.

At this point let me introduce ASAP. This can be best achieved by showing the first of several film strips that will be used in the lecture. Major elements of hardware consist of a CDC 6600 digital computer system, a Model 765 Calcomp Plotter and a CDC 274 Digigraphics Console. The complete ASAP program comprises around one million words in FORTRAN language. The segmentation feature of the CDC 6600 is useful to activate only that portion of ASAP that is of immediate interest.

When the statistical data has been properly programmed, the configuration module accounts for the major elements of the airplane; cockpit, engine, air induction and exhaust systems, wing, tails, etc; arrange them; and uses shoulder point theory to develop the first mold lines around the elements. Physical dimensions and metric data are computed along with fuel available/fuel required solutions. This forms the centroid design for the first parametric studies of the primary variables.

A nine-point carpet of fuel balanced, rubber airplanes is constructed around the centroid design. In constructing the carpet, engine size is fixed. Rational variations in wing area and variations in radius of action around the tentative requirement are selected as independent variables. Take-off gross weight forms the dependent variable. Wing loading and thrust loading clearly were parameters that would effect the level of combat performance being sought. Again, rational variations of W/S and T/W are selected for computation and plotting on the carpet as constraint lines. A typical plot is illustrated on Figure 6a. Independent variables and constraint lines are nondimensional about the centroid design value.

Sustained turn rates at moderate altitude for both supersonic and subsonic speeds and time to accelerate from subsonic to supersonic speed were selected as dominant measures of maneuvering agility. The performance module of ASAP computes these as dependent variables on the nine-point carpet. Results are shown on Figures 6b, 6c and 6d. Absolute values are shown. As subsequent phases of the preliminary design process are discussed, improvements in the design will be illustrated by comparing values of these performance goals.

When the engineer is satisfied that all input and output data is correct, permanent records of this first parametric study are machine plotted.

3. PROGRAM DEFINITION AND THE ANALYSIS PROCESS

These initial investigations showed that tentative mission requirements and performance goals were reasonable. Now a series of trade studies are conducted to refine the physical characteristics of the sized airplane. Typical trade-off areas are shown in Figure 7.

An example of how ASAP is used in the analysis phase is shown on Figures 8 and 9. The independent variable in this trade study is wing aspect ratio. Nine-point carpets and constraints are shown in Figure 8 as computed for aspect ratios of 3.0, 4.0 and 5.0. Although increasing aspect ratio slightly raised the fighters take-off gross weight and increased acceleration time, both subsonic and supersonic sustained turn rates improved with the higher values of aspect ratio. In later phases of this design program, the aspect ratio trade study proved to be of great value. Combat fuel allowance in the mission profile became "task oriented"; that is, it was determined by a specific number of sustained turns at high subsonic and supersonic flight conditions and a specific acceleration profile. This point will be emphasized later in the lecture. Figure 9 summarizes the aspect ratio trade study.

When all significant trade studies identified for analysis were done the first phase of preliminary design is completed. The aircraft has been sized, basic dimensional and metric data has been established by the trade studies and confidence in meeting customer requirements and goals has been obtained.

4. CONCEPT FORMULATION AND THE INITIATION PROCESS

The Program Definition Phase just described usually reveals areas where requirements should be adjusted. Once these adjustments are made the tentative connotation is removed. Requirements become specific and the preliminary design process is focused. Figure 10 shows how the requirements were specified and indicates where further design refinements should be directed.

Sufficient definition of the configuration now exists to begin developmental wind tunnel testing and exploratory manned simulator studies. Experimental evidence hardens the data base and usually reveals a number of detail problems that must be resolved.

Low and high speed wind tunnel data provide experimental evidence for levels of zero lift drag, wave drag, drag due to lift, buffet onset and maximum lift values and lift-drag ratios over the complete flight envelope. Since a premium now is attached to high values of sustained turn capability at both subsonic and supersonic speeds, one design goal became clear; to seek combinations of wing planform geometry, wing loading, camber and twist and high lift system operation that would yield the highest attainable lift-drag ratio at elevated lift coefficients.

The development of the target lift-drag ratio and lift coefficient relationship at the supersonic speed condition is illustrated on Figure 11. Horizontal flight turn rate is proportional to normal load factor, N_Z, which is equal to L/D X Thrust/Weight. With engine thrust fixed, equilibrium turn rate is maximized when flight weight is lowest and when the turn is accomplished at the maximum L/D value attainable at the thrust equal drag condition. This situation exists when the product of lift coefficient times wing area (C_L x S) is maximized at the T = D point. In order to place reasonable bounds on wing area as it relates to weight, acceleration, and other performance goals, a design target lift coefficient near 0.5, slightly beyond the $(L/D)_{max}$ point, was selected from high speed wind tunnel results. Differing wing characteristics provided improvements in the L/D relationship. A twisted wing of aspect ratio 3.5 maximized C_L x S for the supersonic turn condition. The end result was a 7 percent reduction in wing area from the reference value used in the Program Definition Phase of preliminary design.

Similar goals were established for the remaining high-value performance criteria. Once established, these goals comprised a set of design objectives with which to begin the concept formulation phase of preliminary design.

5. CONCEPT FORMULATION AND THE ANALYSIS PROCESS

Using various ASAP submodules, analysis consists of many iterations of the design solution. The most important aspect of the process is intelligent application of new data sources through the man/machine interface of the computerized design synthesis system. In reality this is no more than updating instructions to the basic computer program. A strong feel for design objectives and experience in applying proper combinations of the many data resources will accelerate the configuration convergence process.

Typical of iterations used at this stage of preliminary design is a maneuverability study conducted on Vought's manned, Air Combat Simulator (ACS). This facility is functionally similar to NASA's Differential Maneuvering Simulator. Trade studies during the Program Definition Phase using Colonel John Boyd's Energy Maneuverability Theory showed that the aircraft as initially sized was marginally superior to an adversary fighter. This adversary was synthesized by Vought to provide a realistic point of comparison for design optimization progress and to introduce pilots with air combat experience into the preliminary design program. Use of the Air Combat Simulator as a design tool is illustrated as follows.

Beginning with Figure 12, differential P_S contours between the design and adversary aircraft show a margin of superiority for the design in the subsonic air combat arena, but inferiority at transonic and supersonic speeds. This view of the comparison was quantified in terms of win-loss-draw, or exchange-ratios, using the simulator.

The air combat simulator, shown in this portion of the film clip, consists of two fully operable fighter cockpits enclosed in truncated spherical screens. Overhead projectors provide each pilot with a computer generated image of his opponent's aircraft, a horizon and ground plane, and a gunsight with a lead computing pipper. Each pilot wears a G suit that is inflated proportional to his maneuver load factor and the entire visual display, including the cockpit instrument lighting, dims to represent pilot physiological effects of G and time. Buffet is provided by a hydraulic stick shaker that increases in amplitude as the buffet boundary is penetrated. Operated in real time through associated digital and analog computer equipment the simulator generates and records output parameters for subsequent analysis. Accurately reproducing the real world situation, the system engages two qualified pilots in a highly realistic air-to-air duel. Each pilot is constrained only by his capabilities and those of his simulated aircraft.

Using accepted statistical test procedures, data are obtained for correlating the analytical energy maneuverability results with real time air combat maneuvering in the simulator. Simulator results for the design-adversary engagements are presented in Figure 13. These are histograms of the exchange ratios resulting from 100 one-on-one engagements. The individual bars of the histograms are segregated into discrete Mach regions where wins, losses, and draws were registered by each aircraft. As predicted by Colonel Boyd's approach, the design shows a favorable exchange ratio subsonically but progresses to an inferiority when air battles are pressed to near sonic speeds.

Wind tunnel data conducted in parallel uncovered a unique arrangement for the leading edge flap system that promised a sizable improvement in lift-drag ratio at maneuvering lift coefficients. On Figure 14, drag reductions of up to 118 counts at high subsonic Mach numbers are shown. This benefit is equivalent to increasing excess thrust during hard turning in the combat arena by an average of about 2,500 pounds. Recognizing that thrust equals drag at equilibrium turn conditions, this advantage can be converted into a higher turning load factor at the same flight condition.

This design change to the high lift system improved the energy maneuverability picture substantially as Figure 15 shows. When the change was tested in the simulator, a similar improvement was registered in the exchange ratio obtained by pilots flying the design improvement, Figure 16. Many such studies were conducted using the air combat simulator facility. As a general statement, pilot participation was useful in this stage of preliminary design to quantify effects of design alternatives in terms of win, lose or draw.

Design changes such as this produce a domino effect on resizing the original point design. Remembering that combat fuel allowance was specified on task-oriented (turning) ground rules, improvements in subsonic and supersonic sustained turn rates reduced the time to execute the prescribed maneuvers thus reducing fuel consumed during the maneuvers. When the growth factor is considered, the effect on reducing the take-off gross weight of the design is sizable.

Another example of pilot participation at this stage is illustrated using Vought's Mission Simulator. The film strip shows in this case a single-channel cockpit that is designed to study the weapon delivery system in detail. This particular study investigated the trade-offs involved in providing a "night window" for delivery of air-to-air and air-to-ground weapons during periods of darkness. There were questions to be answered concerning sensor location, the degree of off-boresight pointing needed for target acquisition and - in particular - whether vertigo might result from the necessary small field-of-view that can be generated through a head-up display. Specific design requirements were provided to the engineers by pilot data obtained from the Mission Simulator.

The total effect of developmental wind tunnel testing and of simulator aids was a design concept with considerable confidence based on the depth and breadth of experimental evidence. At the end of the concept formulation phase of preliminary design the aircraft's weight has been reduced to 0.89 of the program definition value. A comparison of improvements in dimensional and performance numbers with those at the end of Program Definition is given on Figure 17.

6. CONTRACT DEFINITION AND THE DESIGN OPTIMIZATION PROCESS

Preliminary design now has progressed to the point of a hardware competition. Contractors all have been working with the customer to understand the high value mission requirements that are implicit in the Type Specification. With the Request for Proposal imminent, each contractor now concentrates on design innovations that will yield a competitive edge over the others.

Developmental wind tunnel models are replaced with final configuration models. Low and high speed aerodynamic models feature a wide variety control and high lift system design variations. Inlet and duct models frequently employ high response pressure transducers that are flat to 10,000 H_z to confirm low distortion patterns. Nozzle-afterbody tests are run with hot gas exhaust and using skin balances to isolate drag from thrust. Typical details of these class models are shown on Figures 18, 19, and 20.

The value of pilot participation increases as the configuration converges during design optimization. Consider the specific requirement relating to stall departure and spin avoidance. The goal is to obtain satisfactory high angle of attack characteristics in the basic aerodynamic configuration. A minimum acceptable method is to control stall and spin properties with stability augmentation.

Forebody shaping and vortex control with the use of forward strakes were shown by wind tunnel tests to operate favorably on static lateral and directional stability coefficients. Figure 21 illustrates the effects. The degree of achievement of the goal was researched and later demonstrated using Vought's Large Amplitude Moving Base Simulator, LAMBS, and its General Purpose Visual Display System. This unique facility, illustrated in Figures 22, 23 and described in the following film strip, is ideally suited for real-time, pilot operating research into problems relating to flying qualities, precision tracking and weapons delivery. Large amplitude motions are provided in five degrees of freedom. Instantaneous rotary acceleration values are: ±68 Rad/Sec^2, ±29 Rad/Sec^2, ±59 Rad/Sec^2 in pitch, yaw, and roll. Instantaneous translational heave and side-force acceleration values are ±3.4 g and ±1.5 g respectively.

Before forebody shaping and forward strake application, characteristic motions beyond an angle of attack of 25 degrees were oscillatory and progressively increasing in amplitude. As the stall is approached, a violent yaw slice occurs. Usually the stall departure could be controlled by immediately reducing angle of attack. When the pilot persists in pulling into the stall, the departure develops into a violent, oscillating motion in roll and yaw. Frequently the end result is a nose down spin.

When the stabilizing effects of forebody shaping and forward strakes were simulated, pilots can pull into a fully developed stall and actually control the airplane about all axes on command. There is little tendency towards inadvertent wing drop or yaw slice and reasonable combinations of control usage will not produce a spin. Confidence in the high angle of attack flying qualities was enhanced through the use of LAMBS to research adequacy of design solutions and subsequently to demonstrate characteristics to customer representatives.

LAMBS also was used to study pilot opinion of balancing the aircraft in pitch to a negative static margin. This approach to Controlled Configured Vehicles, CCV, was calculated to reduce trim drag at all flight conditions, particularly at supersonic speeds. This promised a further competitive edge by reducing the quantity of fuel required for the combat tasks in the design mission profile. Reduced fuel and subsequent resizing again lowered take-off gross weight.

In the Navy application approach speed stability and airframe response to control surface and throttle commands are high value considerations in carrier suitability. This next film strip describes Vought's Night Carrier Approach and Landing Simulator. It features a cockpit mounted on a small amplitude moving base. Visual display of the lights of the carrier at night including the Fresnel lens landing aid is shown on Figure 24. Proper blending of longitudinal and lateral response are investigated and specific design problems relating engine approach power compensation with the Automatic Carrier Landing System are solved.

Finally, detailed attention is given to achieving the highest possible value of lift-drag ratio at turning lift coefficients. Wave drag levels effect turning L/D at supersonic speeds. With design hard points such as engine mounts, main gear location, fuel volume and cockpit scaling dimensions firmly established, the area rule subroutine of ASAP is used to refine mold lines and smooth area distributions at key supersonic speeds. The film strip illustrates how interactive computer graphics permits the designer and the aerodynamicist to work together to reduce wave drag to the minimum level consistent with configuration hard points.

The summation of activities in the design optimization phase of preliminary design has introduced a number of design innovations. Improved tracking, flying qualities and carrier operating characteristics have been obtained along with further reduction in physical size and weight with the attendant improvement in key performance criteria. Figure 25 compares dimensional, metric, and performance values between Program Definition, Concept Formulation and Contract Definition phases of preliminary design.

7. CONCLUDING REMARKS

Modern engineering methods are revolutionizing the preliminary design process. The high state of computer technology brings to the aircraft designer new and novel tools of the trade. New facilities, such as manned simulators and other laboratory design aids, are often initiated in response to a specific problem that exists at the time. With a little foresight, these facilities can be programmed to support future design programs with great effectiveness. They are of particular value during preliminary design in providing:

a. Disciplined interfacing of project direction with technical activity

b. Versatility in providing and/or accepting new sources of data

c. Early introduction of active pilot participation

d. Management visibility into significant, technical areas effecting competitive position

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3. Boyd, Col. John — Expanded Energy Maneuverability Theory; Unpublished.

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- THE BOEING COMPANY (CDPS)
- GRUMMAN AEROSPACE CORPORATION (IDEAS)
- LOCKHEED CALIFORNIA COMPANY (ASSET)
- McDONNEL DOUGLAS CORPORATION (CADE)
- ROCKWELL INTERNATIONAL, B-1 DIVISION (CAP)
- LTV AEROSPACE CORPORATION, VOUGHT SYSTEMS DIVISION (ASAP)
- NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (IPAD)

Figure 1 Examples of Computerized Design Synthesis Programs

PROGRAM DEFINITION	CONCEPT FORMULATION	CONTRACT DEFINITION
• Assess Technology	• Verify Technology	• Validate Design
• Evaluate Requirements	• Confirm Requirements	• Identify Risks and Solutions
• Evaluate Performance	• Establish Levels of Performance	• Define Program and Costs
• Develop Initial Design Concept	• Optimize Design Concept	

Figure 2 Phases of the Preliminary Design Process

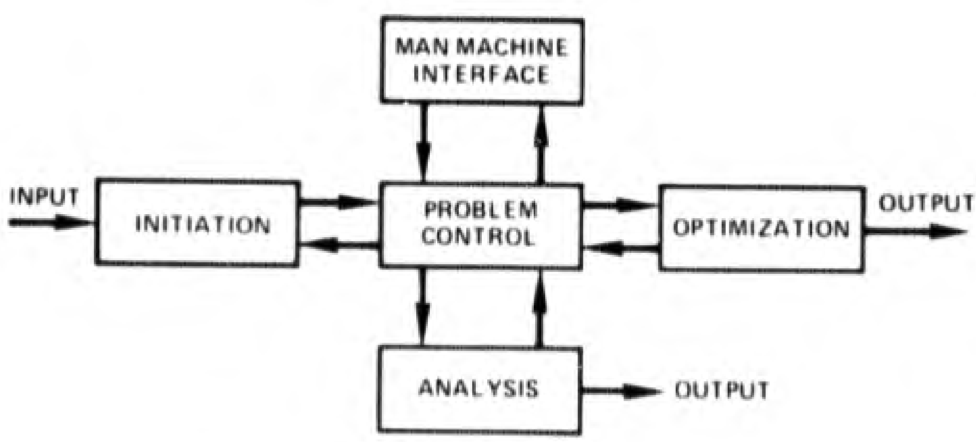

Figure 3 Primary Modules of Vought's Aircraft Synthesis and Analysis Program, ASAP

REQUIREMENT	DESIGN IMPACT
• Light Weight	• Small Size
• Low Cost	• Simple and Low Risk
• Supersonic Maneuvering Agility	• W/S, T/W, Supersonic L/D
• Low Observables	• Small Size, Low RCS
• High Sortie Rate	• Simple and Maintainable
• Superior High Angle-of-Attack Flying Qualities	• High Angle-of-Attack Stability

Figure 4 Tentative Requirements for the Air Superiority Fighter

GEOMETRIC PROPERTIES

- Takeoff Gross Weight and Total Airframe Volume
- Elemental Length, Width, Height for Fighters
- L–W–H Adjustments for Packaging State-of-Art
- Fuselage Volume
- Lifting Surfaces Volumes
- Wetted Areas
- Cross-Section Areas

AERODYNAMIC PROPERTIES

- Low Speed Friction and Form Drag – D/q
- Supersonic Wave Drag – D/q
- Drag Due to Lift – DATCOM
- Lift Curve – Polhamus – Benepe

Figure 5 Typical Examples of Statistically Correlated Design Parameters

See Figure 6 on Following Page

- WING PLANFORM AND SECTION CHARACTERISTICS
- AIR INDUCTION AND EXHAUST SYSTEMS CHARACTERISTICS
- HIGH LIFT SYSTEM ALTERNATIVES
- APPLICATION OF NEW TECHNOLOGY

Figure 7 Typical Trade Studies during Program Definition

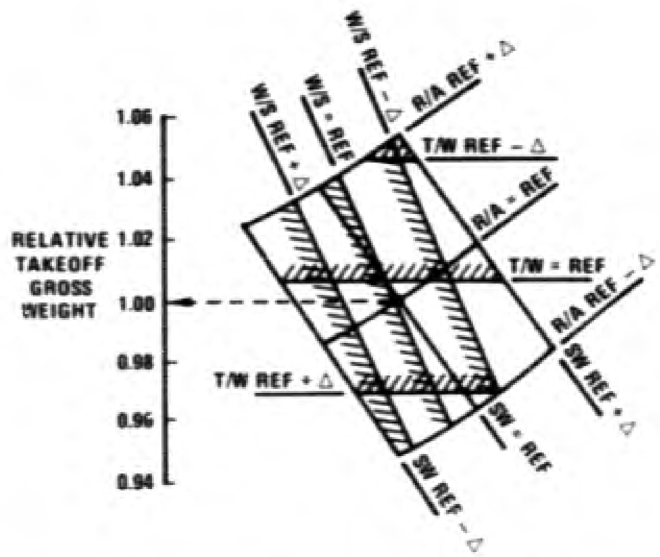

Figure 6a Takeoff Gross Weight Sizing Carpet

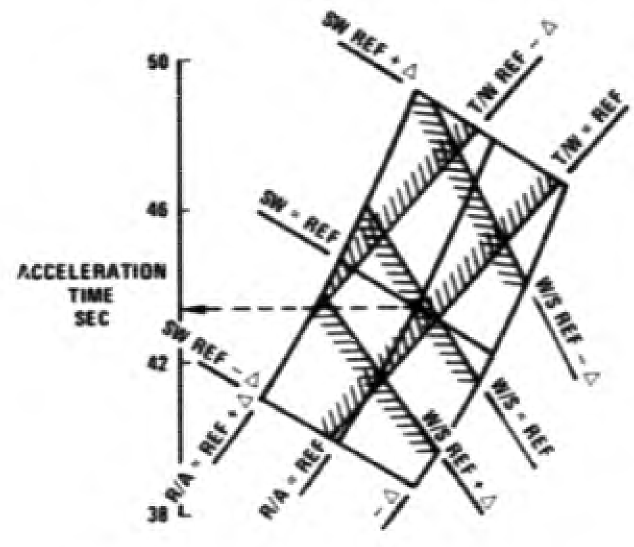

Figure 6b Acceleration Time Carpet

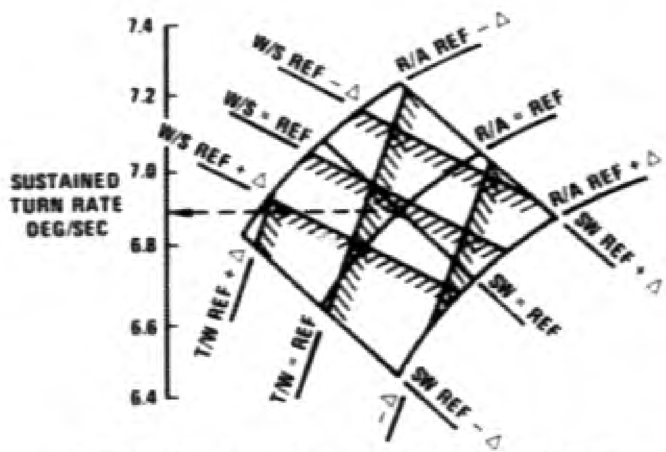

Figure 6c Supersonic Sustained Turn Carpet

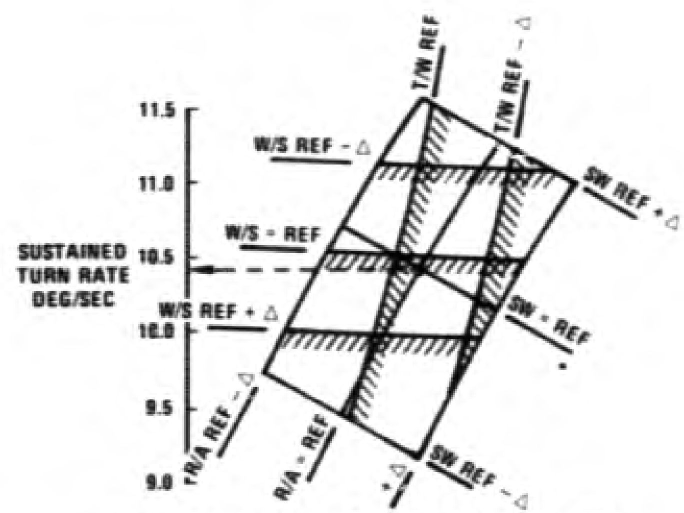

Figure 6d Subsonic Sustained Turn Carpet

Figure 6 Initial Sizing and Performance Carpet Plots

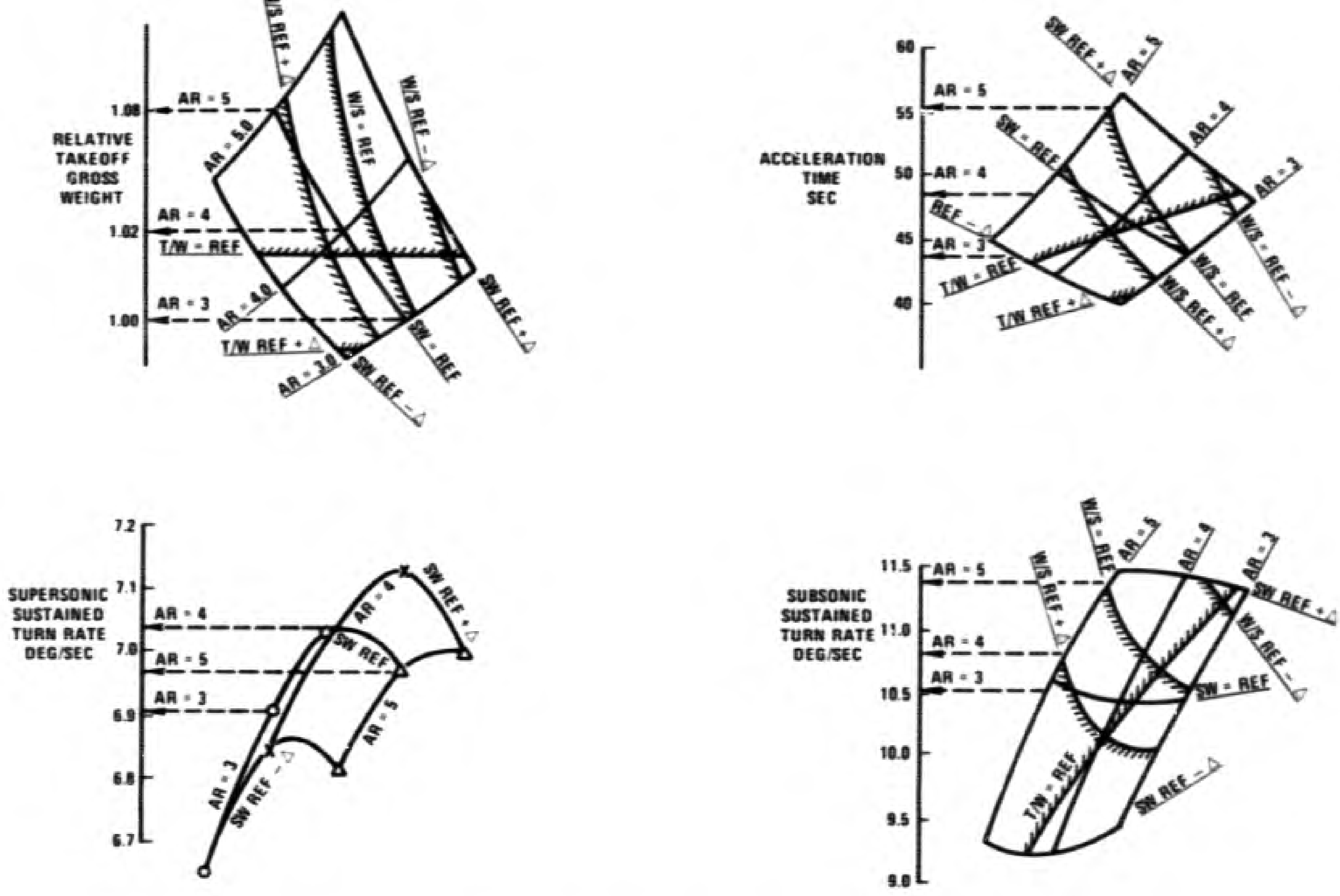

Figure 8 Wing Aspect Ratio Tradeoff Carpets

ASPECT RATIO	3.0	4.0	5.0
RELATIVE TOGW	1.00	1.02	1.08
SUPERSONIC TURN RATE, DEG/SEC	6.91	7.03	6.97
SUBSONIC TURN RATE, DEG/SEC	10.51	10.57	11.35
ACCELERATION TIME, SEC	43.7	48.5	55.5

Figure 9 Wing Aspect Ratio Trade Study Results

- SPECIFIED MAXIMUM WEIGHT
- SPECIFIED MAXIMUM PRODUCTION COST
- PERFORMANCE EVALUATION CRITERIA SPECIFIED
  - SUPERSONIC SUSTAINED TURN RATE
  - SUBSONIC SUSTAINED TURN RATE
  - ACCELERATION TIME; SUBSONIC TO SUPERSONIC SPEED
  - FALLOUT ALTERNATE MISSION CAPABILITY
- GROUND RULES FOR COMPUTING
  - AIRFRAME OBSERVABLES
  - SORTIE RATES
  - COMBAT FUEL QUANTITY
- NO STALL DEPARTURE, SPIN PREVENTION

Figure 10 Specific Requirements for the Air Superiority Fighter

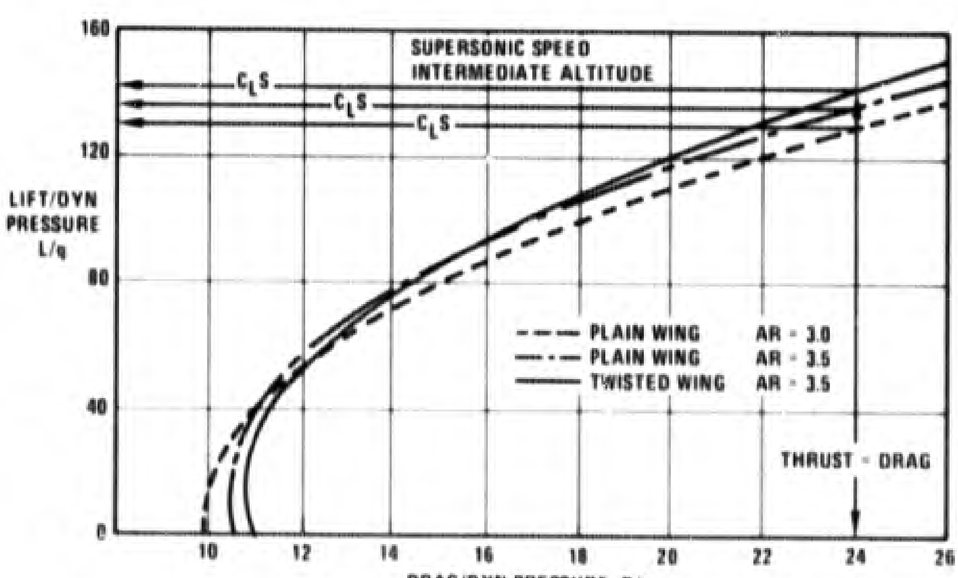

Figure 11 Wing Optimization Procedure

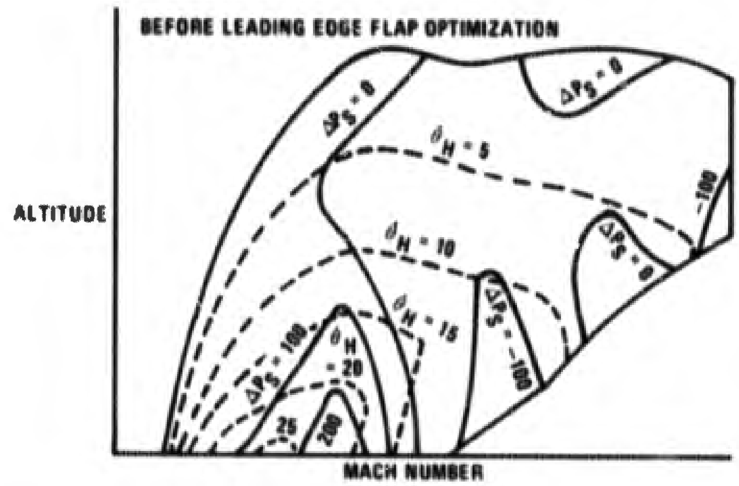

Figure 12 Specific Excess Power Difference Contours

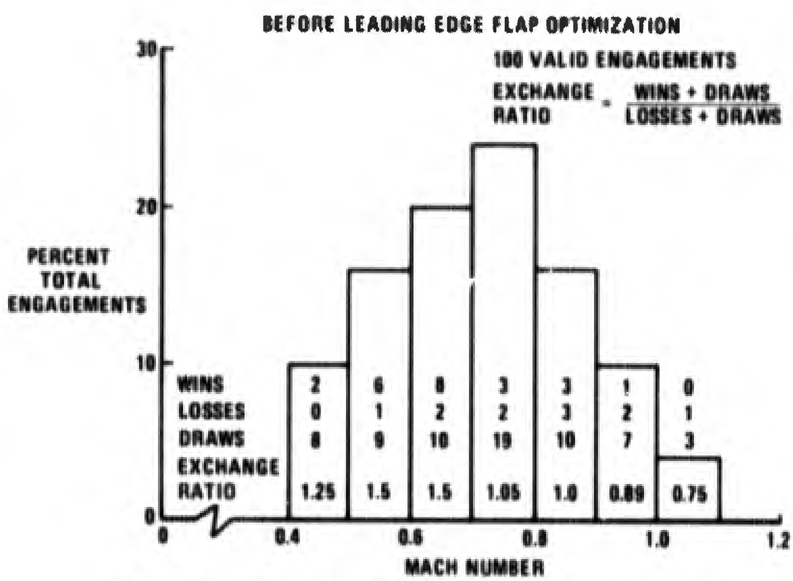

Figure 13 Histogram of Air Combat Simulator Results

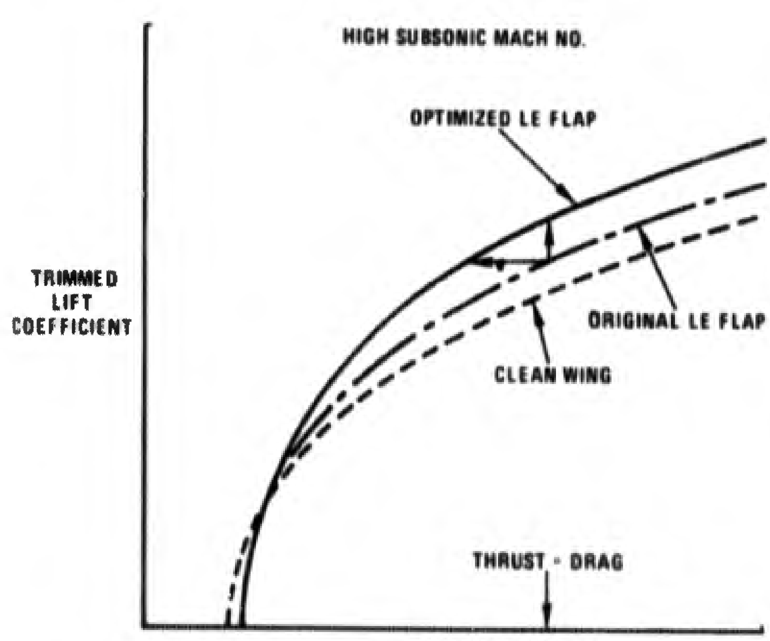

Figure 14 Effects of Leading Edge Flap Optimization on Trimmed Drag Characteristics

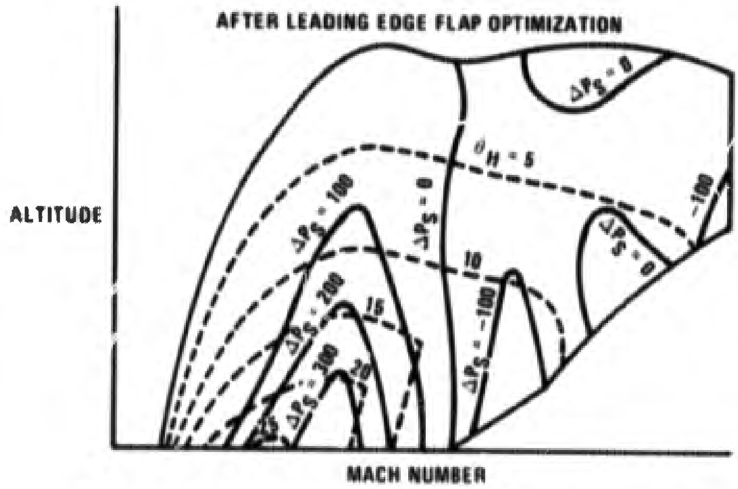

Figure 15 Effects of Leading Edge Flap Optimization on P_s Difference Contours

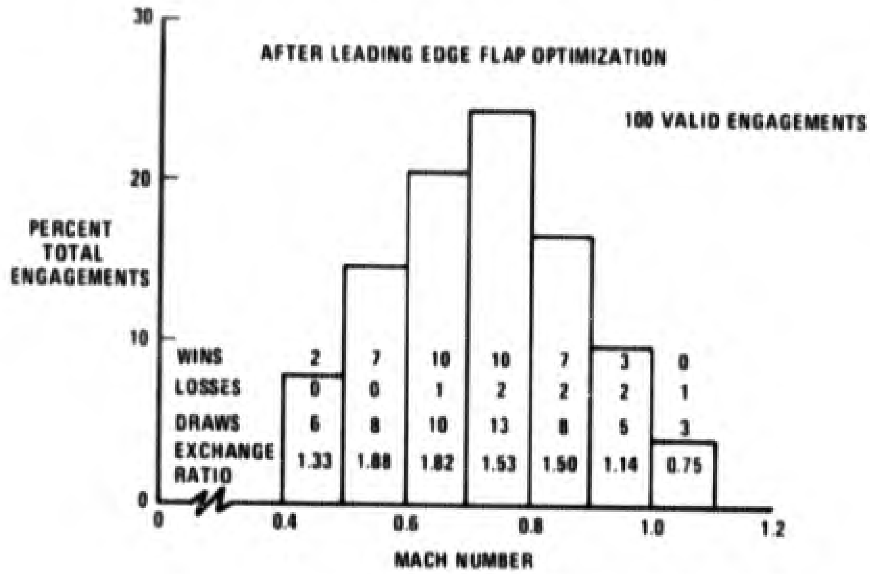

Figure 16 Effects of Leading Edge Flap Optimization on Air Combat Simulator Results

	PROGRAM DEFINITION	CONCEPT FORMULATION
TAKEOFF WEIGHT	1.00	0.89
WING AREA	1.00	0.93
FUEL VOLUME	1.00	0.85
$\dot{H}_H$, SUBSONIC	10.5	10.8
$\dot{H}_H$, SUPERSONIC	6.9	8.1
ACCELERATION TIME	43.7	40.2

Figure 17 Improvements in Air Superiority Fighter Properties at the End of Concept Formulation

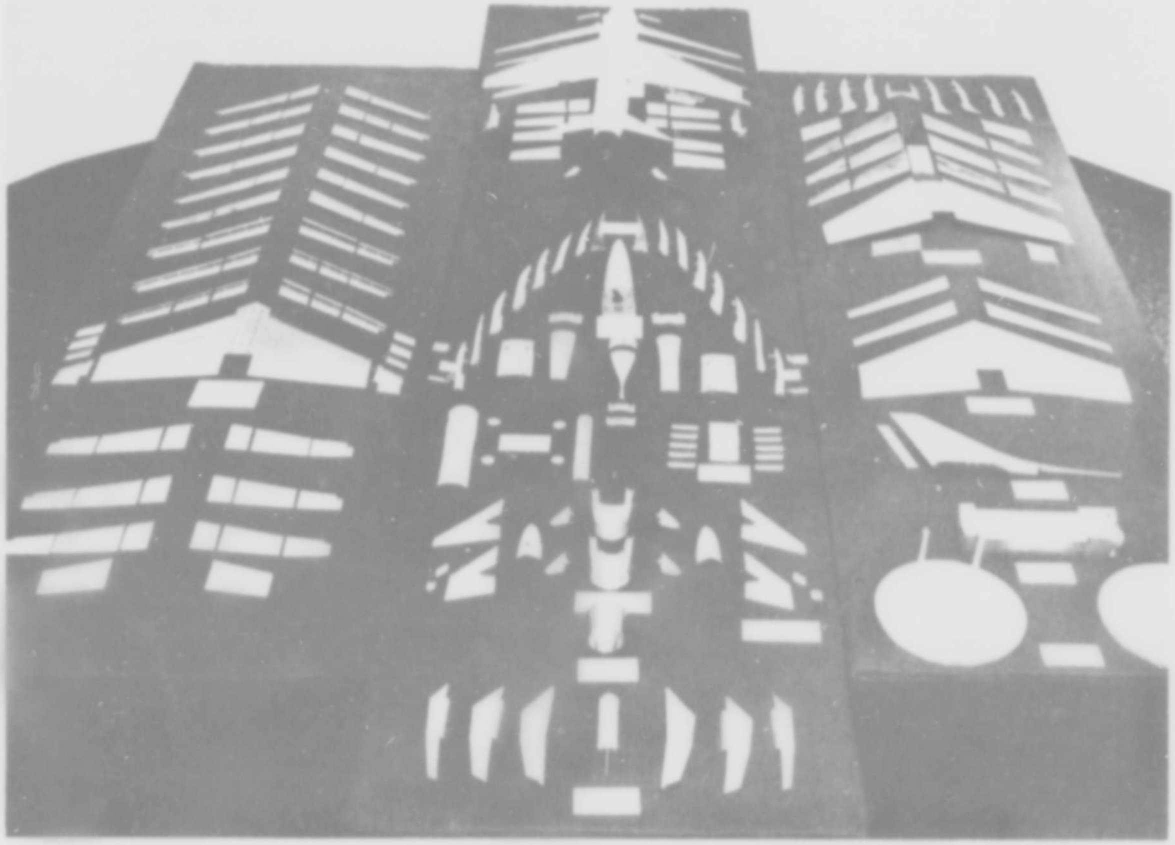

Figure 18 Final Configuration High Speed Aerodynamic Model

Figure 19 Final Configuration Inlet-Duct Model

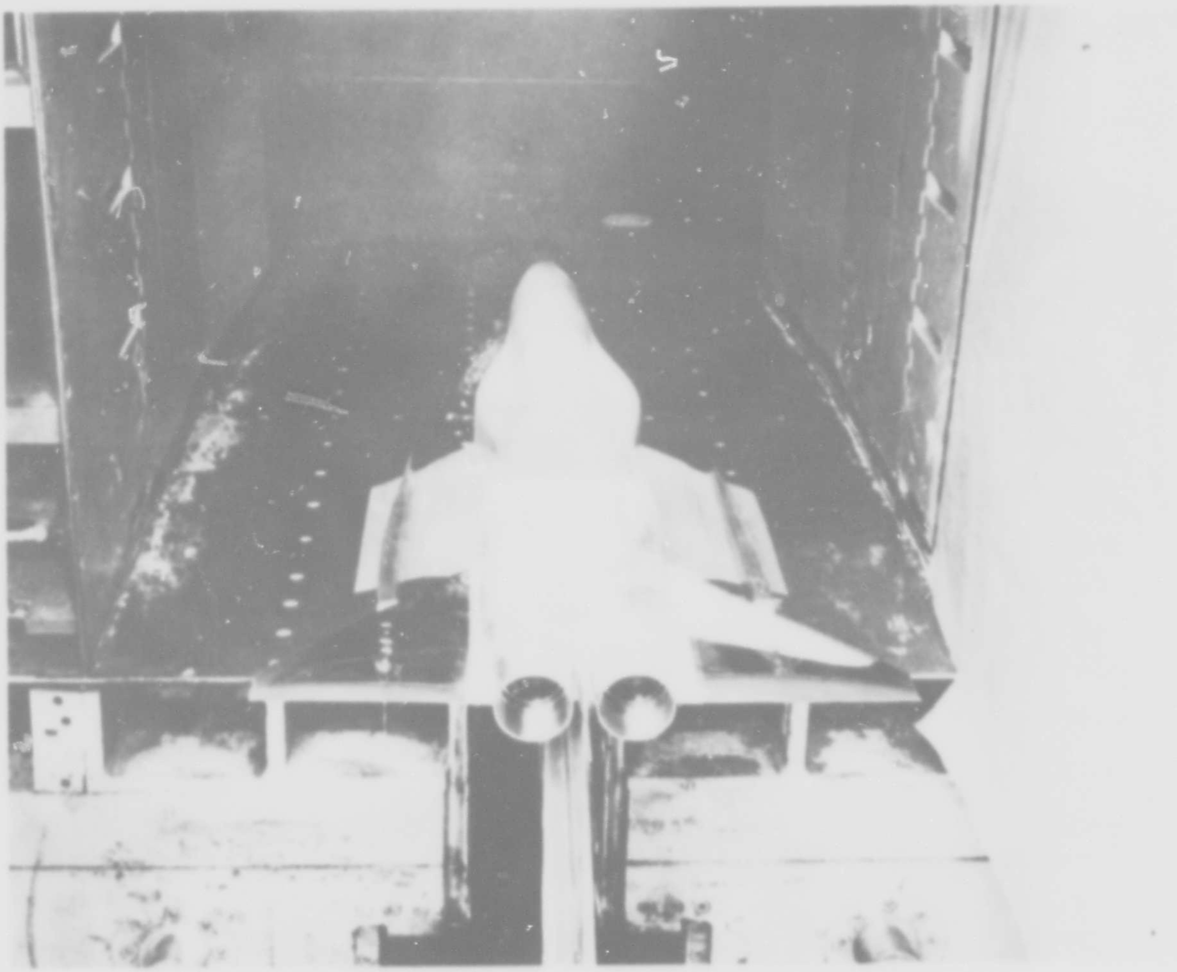

Figure 20 Final Configuration High Speed Nozzle-Afterbody Model

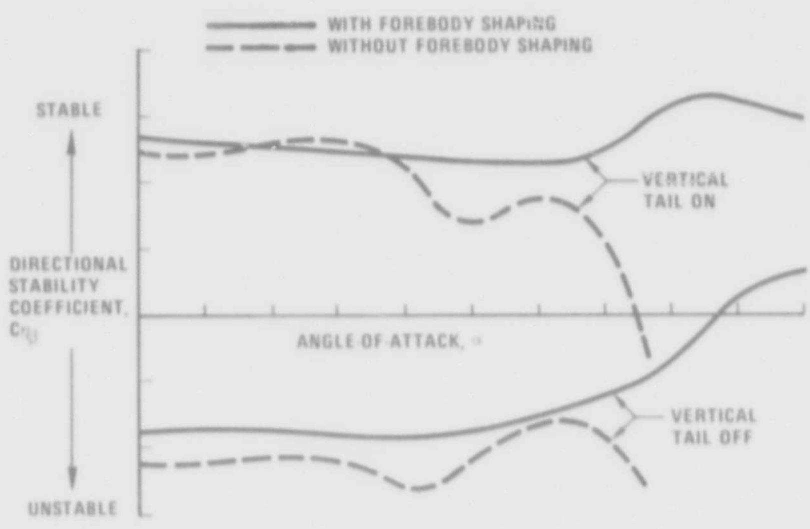

Figure 21 Effects of Fuselage Forebody Shaping on Static Directional Stability

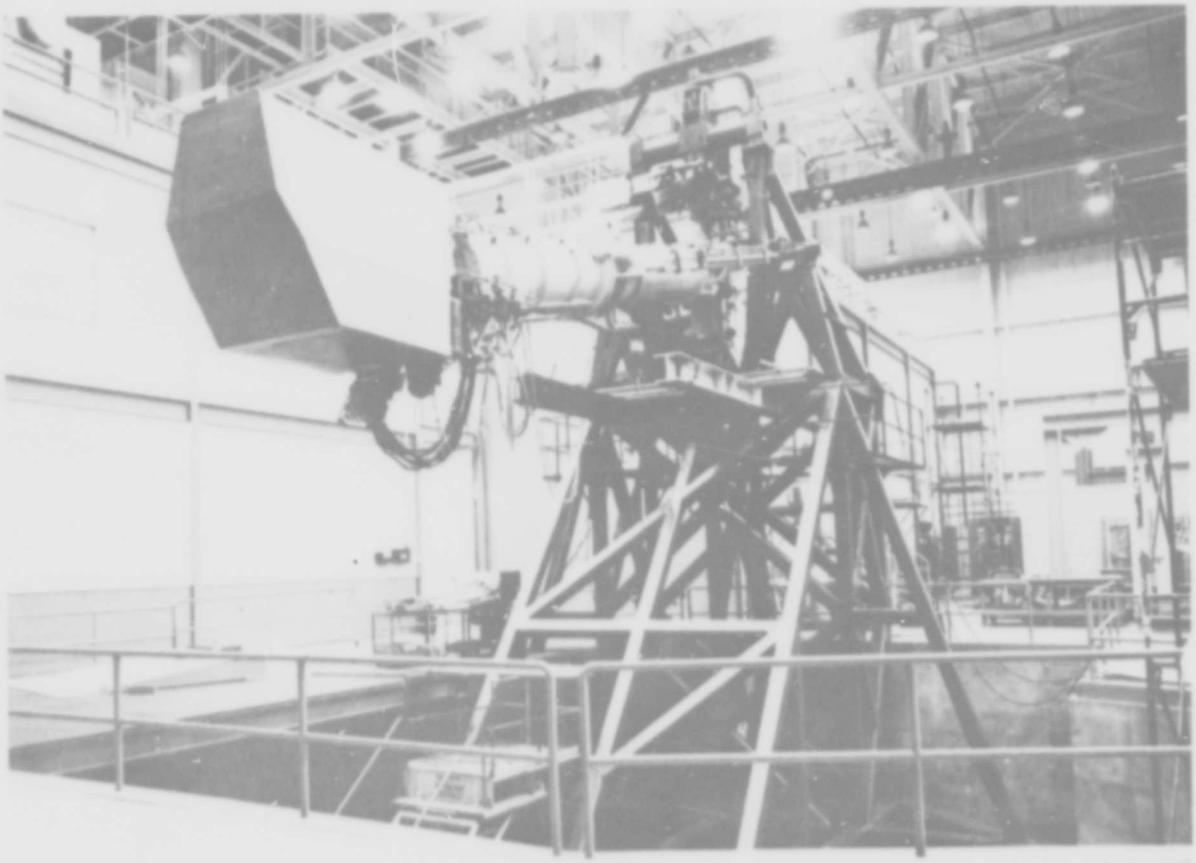

Figure 22 Large Amplitude Moving Base Simulator

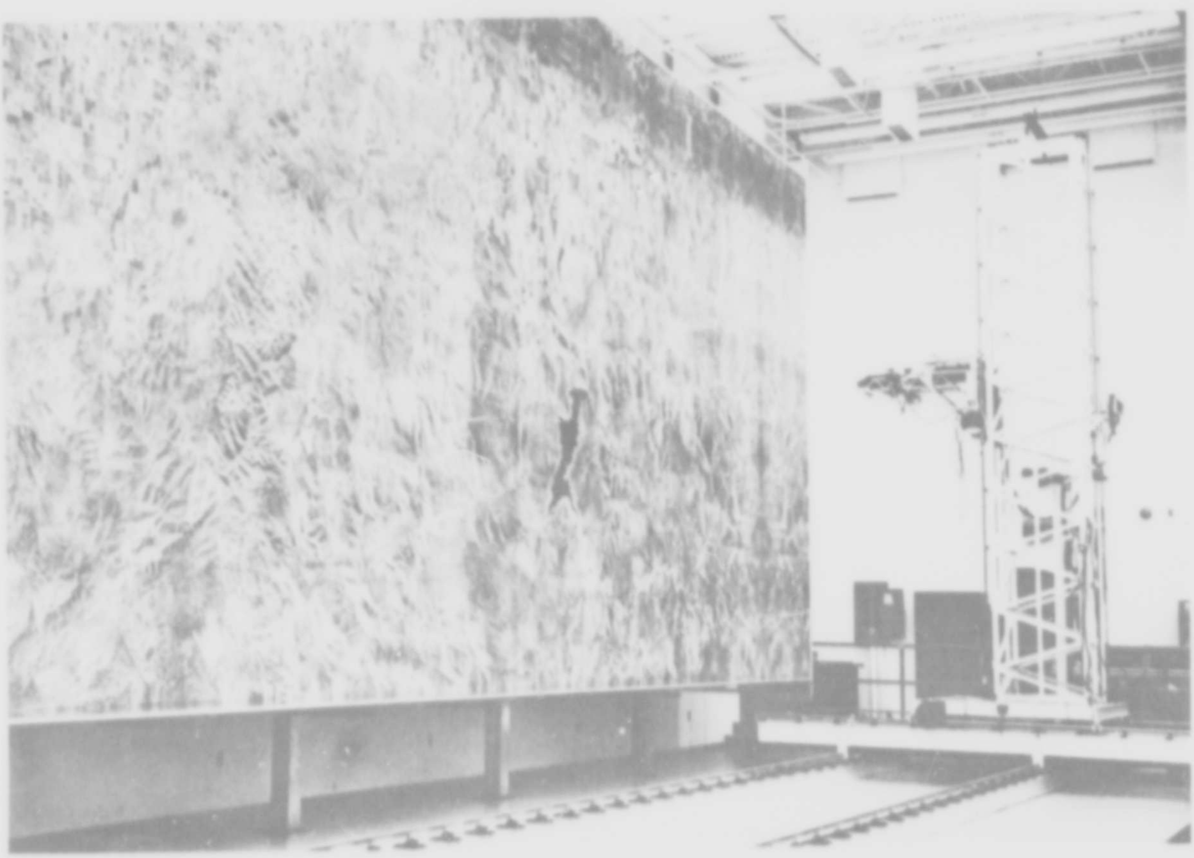

Figure 23 General Purpose Visual Display System

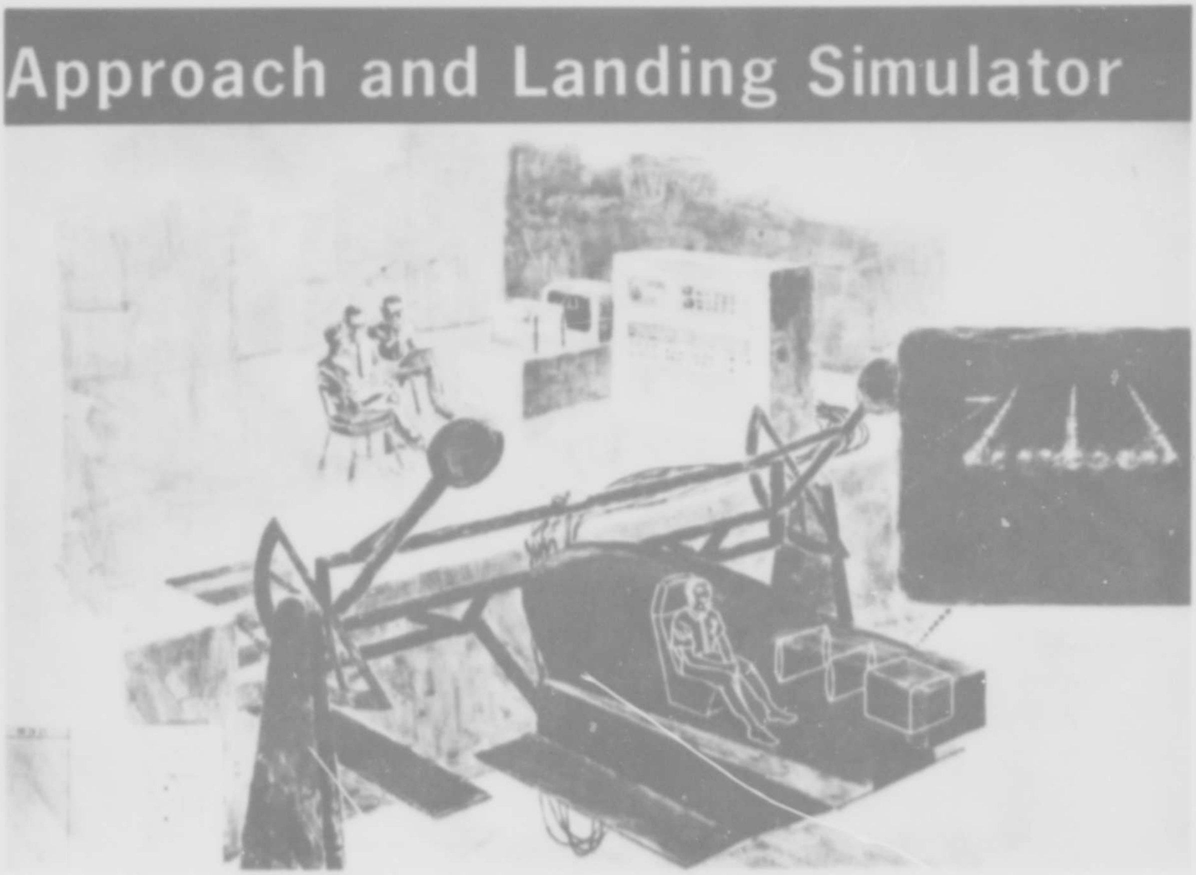

Figure 24 Carrier Approach and Landing Simulator

PROGRAM PHASE	PROGRAM DEFINITION	CONCEPT FORMULATION	CONTRACT DEFINITION
TAKEOFF WEIGHT	1.00	0.89	0.84
WING AREA	1.00	0.93	0.93
FUEL VOLUME	1.00	0.85	0.77
$\dot{\theta}_H$, SUBSONIC	10.5	10.8	12.0
$\dot{\theta}_H$, SUPERSONIC	6.9	8.1	9.3
ACCELERATION TIME	43.7	40.2	37.0

Figure 25 Comparison of Air Superiority Fighter Characteristics at Three Phases of Preliminary Design

THE TEAM LEADER'S ROLE IN "DESIGN TO COST" PRELIMINARY DESIGN

by

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INTRODUCTION

Over the past twenty-five years the requirements process has been formalized to such an extent that when a set of requirements has been established, they are considered inflexible. The evolution of this process to today's state has been driven by a vast growth of technology which provides the requirement planners with a bewildering array of options to choose from. These choices are optimized with the help of operations research studies which generally try to get the most cost-effective solution by emphasizing maximum effectiveness, not minimum cost. Several recent programs in the United States are splendid examples of this approach. We in industry have done our part in the process by approaching the requirement planners with enticing and optimistic performance estimates achievable with new technology. We have also helped the planners to invent scenarios to justify the need for the performance promised by the new technology. Once justified the requirements become inviolate. Since the requirements usually rest on an over-optimistic foundation, all bidders for the program come in with similar promises lest they be considered non-responsive. Thus, the winner of the auction is stuck with an impossible task. The end result of all this has been a weapon system cost growth over the last twenty-five years that is seriously straining defense budgets, particularly when the taxpayers do not consider the military threats used to justify some new systems as credible.

A parallel development to the formalization of the requirements has also been going on. This is the growth of new disciplines and sub-disciplines such as reliability, maintainability, survivability, system safety, system analysis and others. These new disciplines, or "cults" as some people call them, have spawned a host of specifications, procedures and methodologies which have become contract requirements just as rigid as the military requirements. Thus, both government and industry employ large numbers of specialists in these disciplines whose jobs depend on retaining and strengthening these requirements. Naturally, it is difficult to get relief and simplification of these requirements.

THE STUDY

It was in an attempt to slow or reverse this trend that design-to-cost programs such as AX, Lightweight Fighter, and AMST were started several years ago. The Lightweight Fighter program is unique in that there is no stated military requirement at this time; rather the prototypes are intended to explore new technologies, to improve maneuverability and, if proven successful, to provide options for the Force structure.

Shortly after the F-15 competition, General Dynamics and other companies started in-house studies to examine what could be done if the design mission was limited to only air superiority. Later General Dynamics and Northrop were both given study contracts by the Deputy for Development Planning, Aeronautical Systems Division of the Air Force Systems Command.

The study was for the purpose of validating the integration of advanced energy maneuverability theory with trade-off analyses. Emphasis was placed on low unit cost and high transonic maneuverability. Weight was used as a primary measure of unit cost and only minimum mission-essential equipment was employed. Multimission capability was ignored until after initial sizing of the various aircraft that were synthesized. Trade-offs were made between single versus twin airplanes, a very small single-engine airplane and other design features as follows:

1. Supercritical wing
2. Composite material usage
3. Inlets
4. Wing geometry
5. Structural criteria
6. Mission rules
7. Self-sealing tanks
8. Tail hook

The specific results of this study show that visual air-to-air day fighters at weights less than one-half of current air-superiority fighters can be developed to have superior maneuvering performance and adequate mission range and combat fuel allowance without the use of advanced technologies. It is the mission-essential/combat-relevant/design-discipline approach to the concept that provides the superior maneuverability necessary to win air battles against future threats. The nature of the concept -- small size and simplicity -- will ensure low procurement and operating costs. Each of the many requirements that could be added to the concept (e.g., sophisticated inlets for better high-Mach capability, higher structural load factor, self-sealing fuel tanks, tail hooks, speed brakes, autopilot, nose wheel steering, etc.) does not by itself add a significant penalty to the aircraft to perform the design mission or markedly reduce its maneuverability; however, taken collectively, they destroy the feasibility of providing a truly superior maneuvering fighter and increase the procurement and operating costs. The greatest achievements are attained by excluding each design criterion and specification that does not contribute directly to winning the air-to-air engagement through superior maneuverability in the primary air battle arena.

All of the foregoing study activities were accomplished well in advance of the Air Force decision to issue an RFP for the Lightweight Fighter program and resulted in an inherently low-cost preliminary design which has been comprehensibly treated by Mr. William C. Dietz in Reference (a), presented at the AGARD Flight Mechanics Panel in Florence, Italy, in October 1973.

THE ORGANIZATION

The writer was assigned as Program Director in November 1971 when we were starting to build our proposal team, and the best way to describe the team leader's role is to describe all of the activities during the course of the program. The first order of business was to select the team, particularly the engineering part of the team. The team assembled (shown in Figure 1) was co-located in a hanger away from the mainstream activities. The team that put the proposal together was to be the team to carry out the contract if we won. All management positions on the team were filled by people who had had direct and important experience with at least three or more design and development aircraft programs.

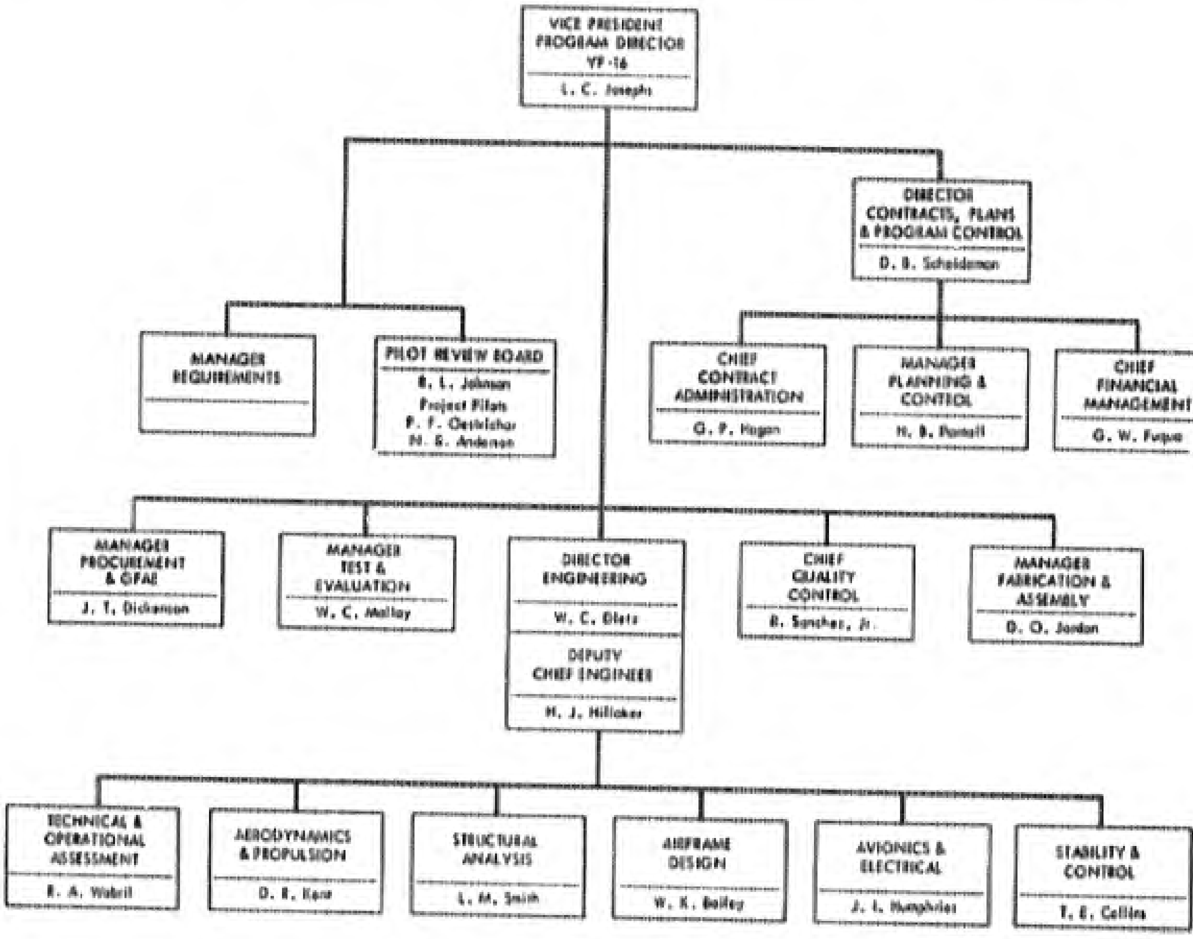


Figure 1 PROTOTYPE ORGANIZATION

THE INTELLIGENCE GATHERING

The next order of business was to find out what the Air Force and Department of Defense really intended to do with the Lightweight Fighter prototype program. Deputy Defense Secretary David Packard had stated that the purpose of his proposed prototype program was (Figure 2):

- EXPLORE NEW TECHNOLOGY
- PROVIDE OPTIONS FOR THE FORCE STRUCTURE
- PROVIDE A MEANS TO KEEP DESIGN TEAMS TOGETHER
- REDUCE RISK IN FULL SCALE DEVELOPMENT BY "FLY BEFORE BUY"

Figure 2 PURPOSE OF PROTOTYPE PROGRAM

The Air Force in adopting the prototype idea concurred, but in addition Lt. Gen. Stewart, Commander of ASD, wanted to try some new procurement and management ideas (Figure 3).

- MAXIMUM FREEDOM FOR CONTRACTOR - BEST EFFORTS BASIS
- PARTICIPATION BY THE AIR FORCE BY OBSERVATION
- MINIMUM FORMAL PAPER WORK
- FIXED DOLLAR LIMITS

Figure 3 ASD MANAGEMENT APPROACH

The RFP, which was issued on 6 January 1972, followed those principles. There were no firm requirements. Everything was considered a design goal. For example, the maneuverability goal was described in terms that did not include performance numbers (Figure 4). This approach made it unnecessary to be overly optomistic with performance estimates in order to avoid being considered non-responsive. In addition, at the bidders' briefing the Air Force asked each bidder to submit a wind-tunnel model for test and evaluation as a part of the source-selection process. The RFP emphasized that advanced technology was important in the prototype designs offered. Low cost was also emphasized although the specific design-to-cost number was not established until the time the contracts were signed. A flyaway unit cost number of approximately $3 million was widely discussed, however.

No management techniques were specified. Instead, each bidder was asked to propose his management approach. The bidder's response was limited to 50 pages for technical and 10 for management exclusive of the cost proposal. Only seven copies of the proposal were requested. It was stated that the proposals would be evaluated as a whole subjectively and not broken down by disciplines and scored as has been the custom in the past. The Air Force also announced how much money they had for the winning two contractors.

THE AIRCRAFT SHOULD PROVIDE:

- MAXIMUM MANEUVERING CAPABILITY AT AVERAGE COMBAT WEIGHT

EMPHASIS SHOULD BE PLACED UPON:

- SUSTAINED TURN CAPABILITY AT M 1.2 AND M 0.9, 30,000 FT.
- LEVEL ACCELERATION BETWEEN 0.9 - 1.6M AT 30,000 FT.
- MAXIMUM FULLY CONTROLLABLE G AT M 0.8 AT 40,000 FT.

Figure 4 PERFORMANCE

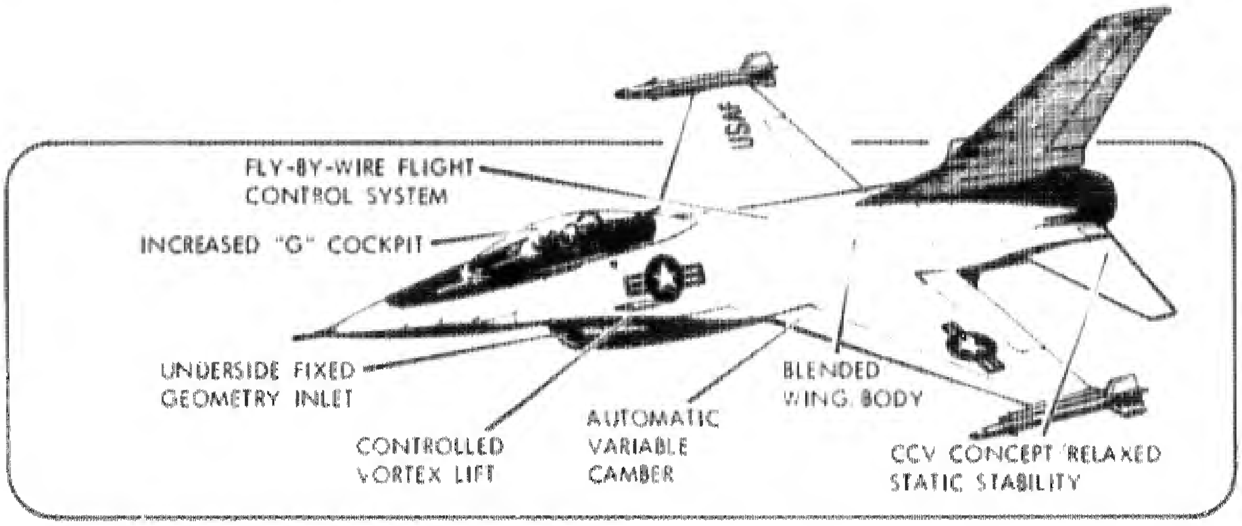
THE RESPONSE

After determining what the customer wanted, it was necessary to review the design that resulted from the preliminary studies and wind-tunnel tests. As noted above, the formal study report had concluded that advanced technologies were not required if one were austere enough in limiting design features which we now take for granted. It was, however, obvious that the airplane had to have advanced technologies in order to justify prototyping at all. In addition, it had to have many of the features which are now considered standard but which add weight. Fortunately, most of the advanced technologies which were chosen served to reduce weight more than was added for so-called standard features and added very little cost.

The choice of technologies was made using the following criteria:

1. Must contribute directly to design goals
2. Must be sufficiently advanced
3. Must not be too high a risk
4. Must fall within constraints of cost, complexity and utility

Figure 5 shows the advanced technologies chosen for our design. They are described in detail in Reference (a).



- 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 5 ADVANCED TECHNOLOGY - DESIGN INNOVATIONS

As may be expected, we made a significant change in our design from the study airplane to the proposal airplane both as a result of the addition of advanced technologies and further wind-tunnel tests. Figure 6 shows the evolution of the design.

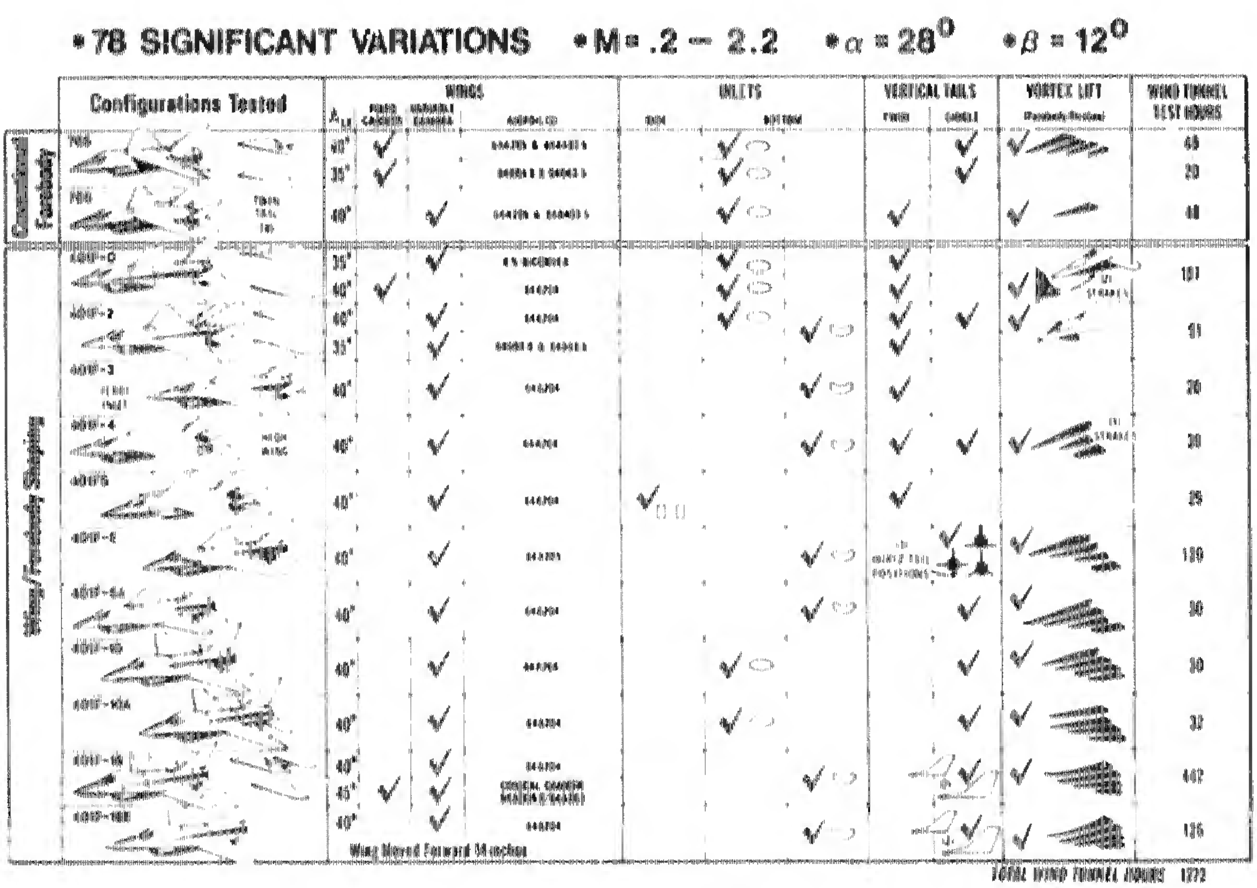


Figure 6 LWF FORCE MODELS TESTED

Concurrent with the design effort, it was necessary to structure the management approach, determine the scope of work to be offered within the fixed dollars set by the Air Force, and establish our policy with respect military specifications. We structured our management approach and plans around a simplified work breakdown structure shown in Figure 7. In addition, we named an element manager for each of the controllable elements of the WBS. As is usual, we had the most trouble with the cost estimates. Our first cost estimate was done parametrically and came out some $6 million more than the Air Force had available. We then estimated task by task using the WBS framework and got down to the desired level. At this point the official RFP came out and we re-estimated again. This time we were back up to $6 million over. We then eliminated specific tasks to get back to the desired level. At no time did we make arbitrary management reductions in cost estimates. Figure 8 shows how this final estimate was achieved. Incidentally, at no time did we make any arbitrary optimistic reductions in drag or weight either. This policy paid off since, as mentioned above, the Air Force ran its own wind-tunnel tests using contractor's models as part of the evaluation.

With respect to military specifications, we decided to adhere to them as much as possible in order to avoid redesign in the event we were successful in going on into production with the F-16.

The next and probably most difficult problem was to describe our design and performance in 50 pages and our management approach in 10 pages. Engineers are frequently very poor writers, particularly when concise summaries of complex technical material are needed. Also, since the proposal was going to be evaluated as whole and not by technical discipline, it was necessary to tell our story in a homogeneous way, omitting any material that was not relevant. We finally engaged a well-known consultant to work with the relatively few people doing the actual writing and editing. He helped us to sharpen up and otherwise improve our writing so that after we had won one of the contracts we were told that our proposal was the best ever submitted to ASD.

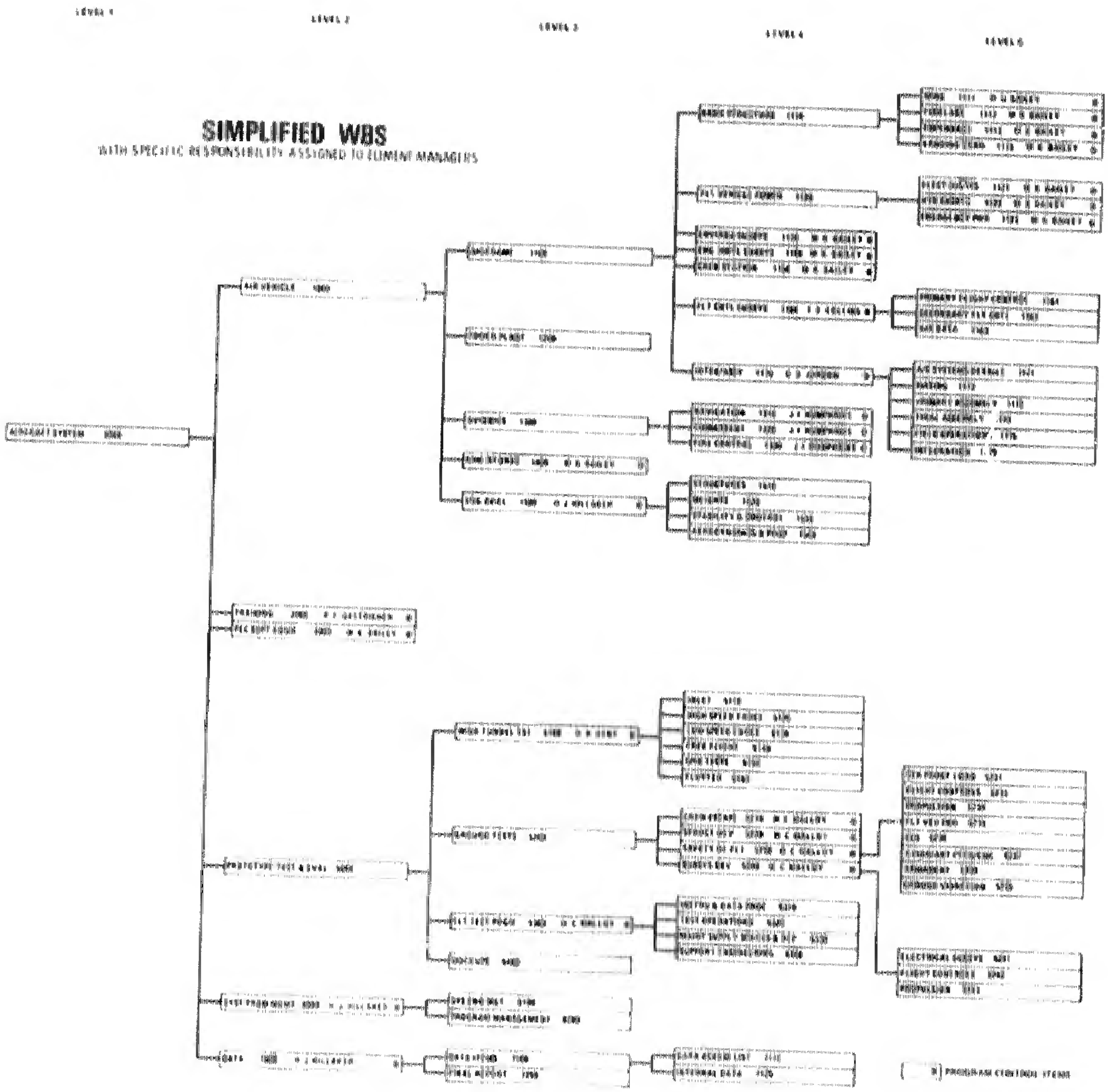


Figure 7 SIMPLIFIED WBS

BY WBS ELEMENT

(Amounts in Thousands Less Fee)	BASELINE ESTIMATE CONFIGURATION 16E	PROPOSAL 18 FEB 1972
AIRCRAFT SYSTEM	40,622	34,095
AIR VEHICLE	23,446	21,710
AIRFRAME	20,912	19,843
BASIC STRUCTURE	9,754	8,715
Wing	1,735	1,627
Fuselage	5,222	4,672
Empennage	1,779	1,571
Landing Gear	1,018	845
FLT. VEHICLE POWER	2,054	2,368
Electrical Subsystem	719	622
Hydraulic Subsystem	718	1,221
Auxiliary Power	247	230
Accessory Drive	370	295
ENVIRONMENTAL SUBSYSTEM	770	660
FUEL SUBSYSTEM	505	339
ENGINE ANCILLARY EQUIPMENT	449	472
CREW STATION	1,227	1,150
FLT. CONTROL SUBSYSTEM	3,487	2,300
INTEGRATION/ASSEMBLY	2,666	3,839
Integration	1,551	2,992
Mating	194	166

Figure 8 TRACEABLE COST ESTIMATES

THE EXECUTION

We were placed under contract on 14 April 1972. The contract requirements to accomplish the 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 F2P-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 was 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. The schedule for accomplishing the program is shown in Figure 9. The budget established for the program is shown in Figure 10.

You will note that the first six months of activities were to be a period of design refinement before starting to release drawings for manufacture. During this period in most programs substantial changes take place in the design because of over-optimism in the proposed design and interference from the customer's technical staff who insist on compliance with many requirements that may or may not be necessary.

We made the decision to hold firmly to the proposal design since we had not proposed optimistic performance, and by the terms of the contract we had complete freedom from usual constraints laid on by the specialists. Accordingly, we proceeded with a series of wind-tunnel tests that were to be for confirmation and fine tuning only. Figure 11 shows how successful we were in holding the configuration firm. This is the first of the two main reasons the program has proceeded on schedule and under budget. The second reason for our success so far has been the way we managed the program. As mentioned earlier, all supervisory personnel were highly experienced. The entire project was located where the airplanes were built, but the engineers were not isolated from close communication with their home departments when problems arose. Decision making was delegated to the lowest level possible. No one waited for ratification of a decision, but decisions were reviewed after the fact. Because of the experience level of the managers, few mistakes were found and corrective action was easy. Communication was facilitated by holding a daily meeting at 8:00 a.m. of all project supervision. This meeting was open to all customer personnel who might happen to be visiting at the time. No problems were allowed to go unresolved before the personnel involved went home at night; either a solution was found or a course of action agreed upon.

Cost information was available to the designers, and frequently design approaches were changed at the request of the purchasing, tooling, or manufacturing people who were also co-located in the program area and had full participation in design decisions. As a result, it is now possible to state that the design-to-cost goal can be met in production without any departure from the prototype design.

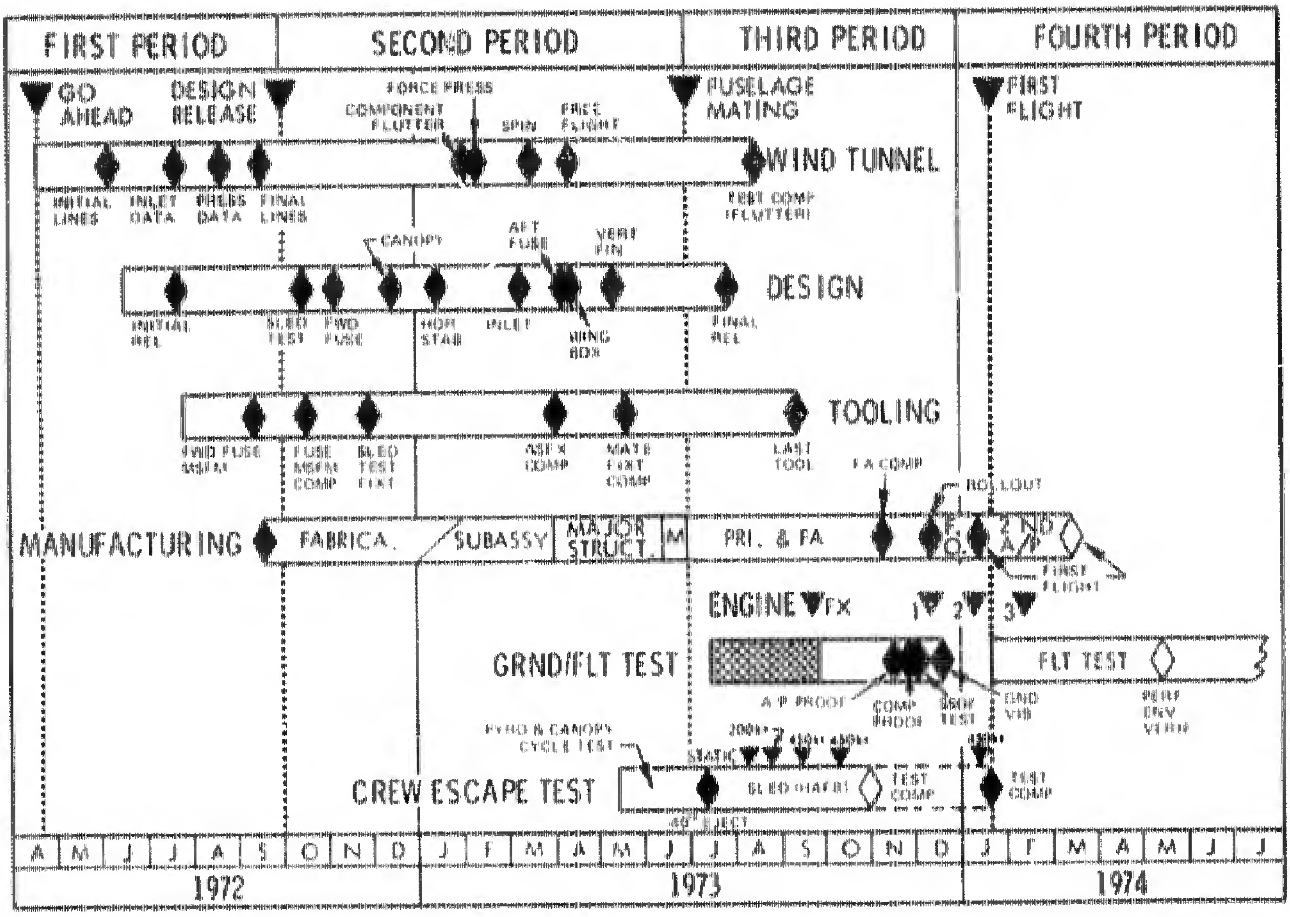


Figure 9 YF-16 PROTOTYPE PROGRAM

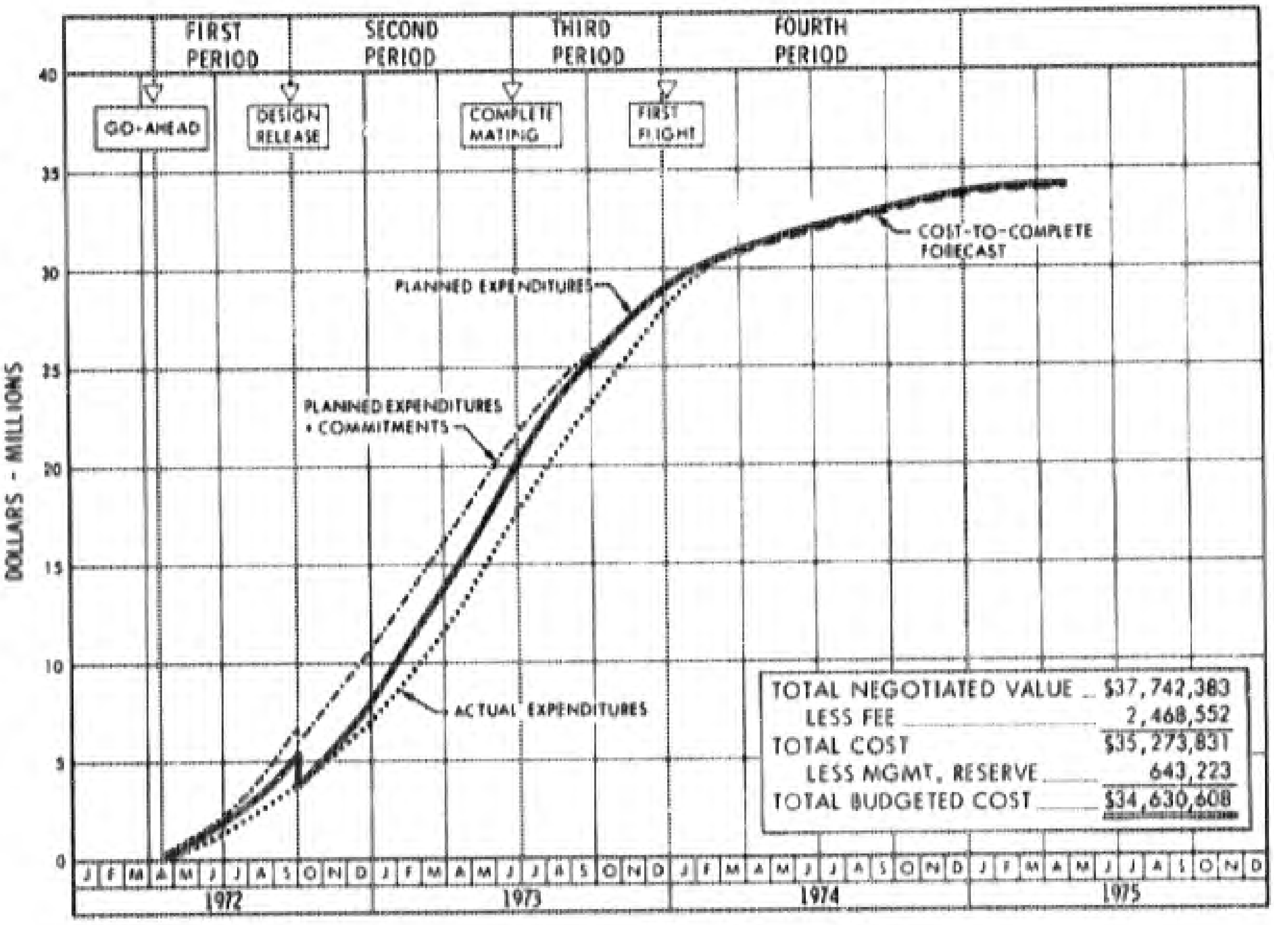


Figure 10 TOTAL BUDGET COST vs EXPENDITURES

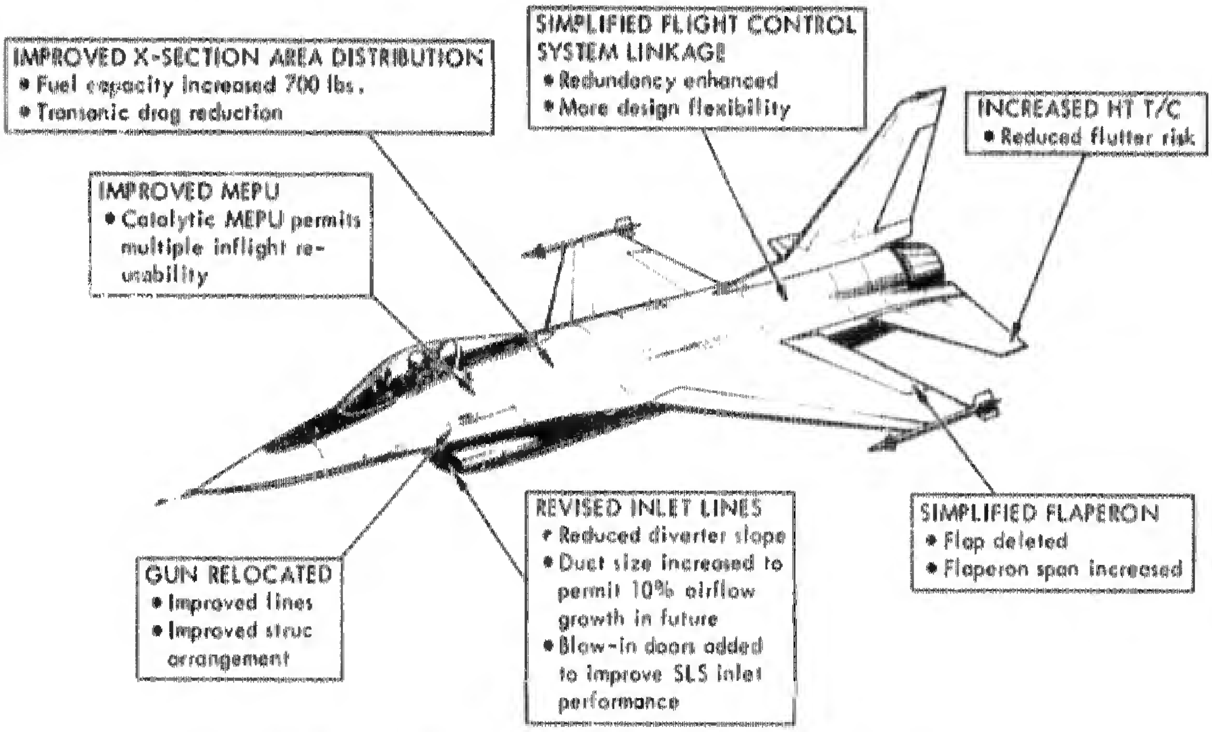


Figure 11 CONFIGURATION CHANGES SINCE PROPOSAL

THE RESULTS

The program is now in the flight-test phase, and it is still on schedule and under budget. We have had 28 flights on Number 1 airplane as of 18 March. Figure 12 shows that we opened up approximately 70 percent of the flight envelope during this period. Figures 13 through 18 show that during this period we established that the airplane meets its design goals in every respect. A 10-minute film shows the airplane in flight.

Design-to-cost has long been a way in the commercial sectors of Western economies, and it is now a reality in the United States in the military sector. The YF-16 program shows that it can be done without sacrificing essential performance.

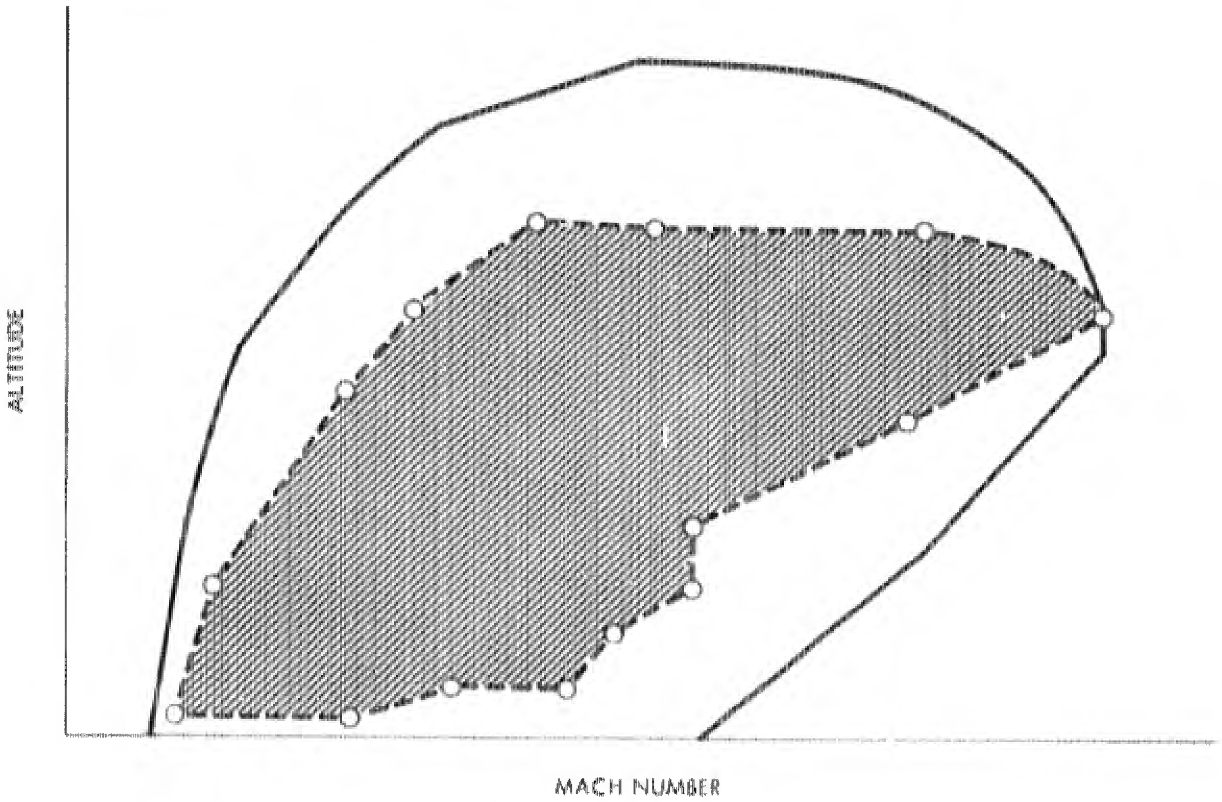


Figure 12 YF-16 FLIGHT ENVELOPE

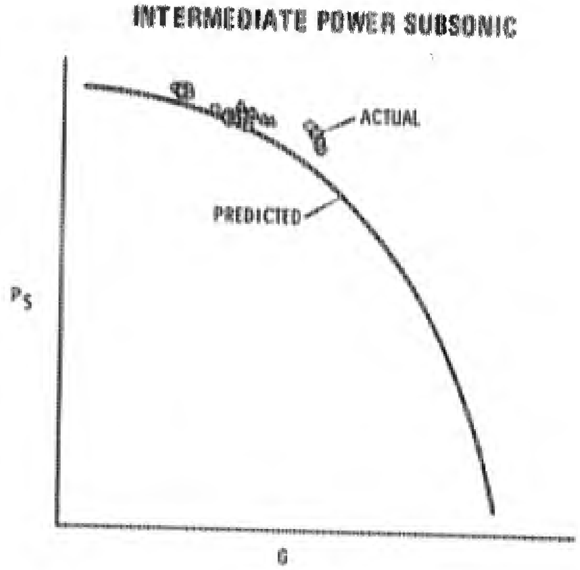


Figure 13

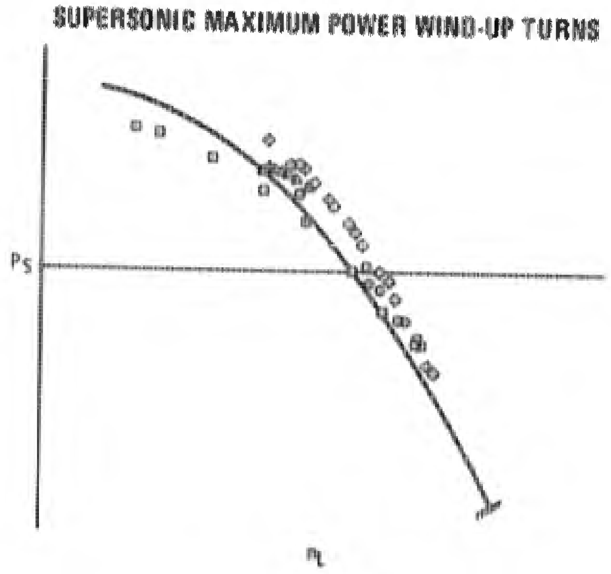


Figure 14

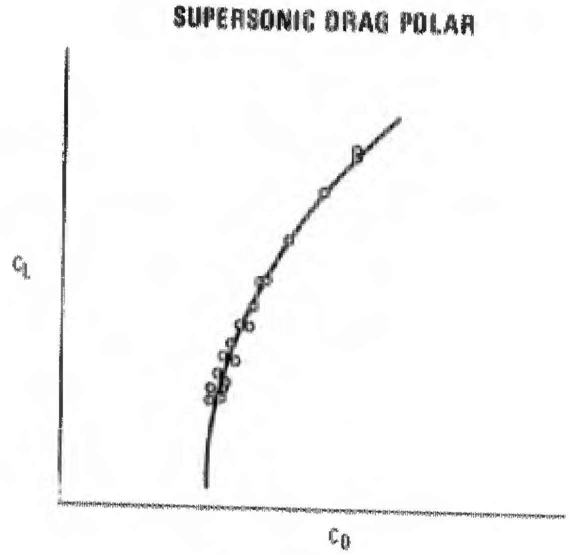


Figure 15

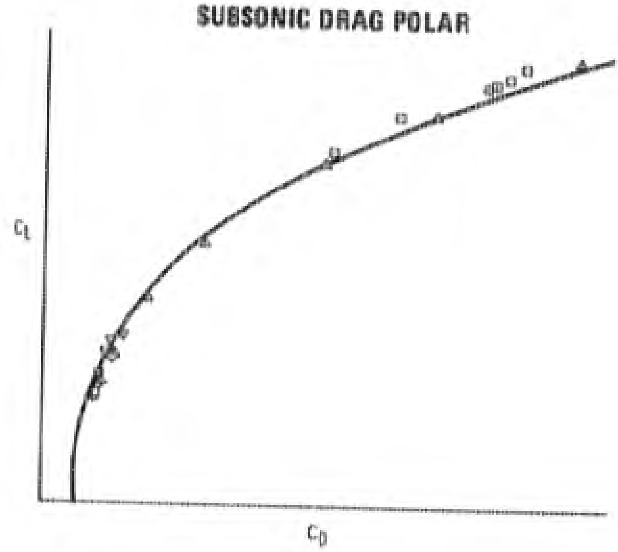


Figure 16

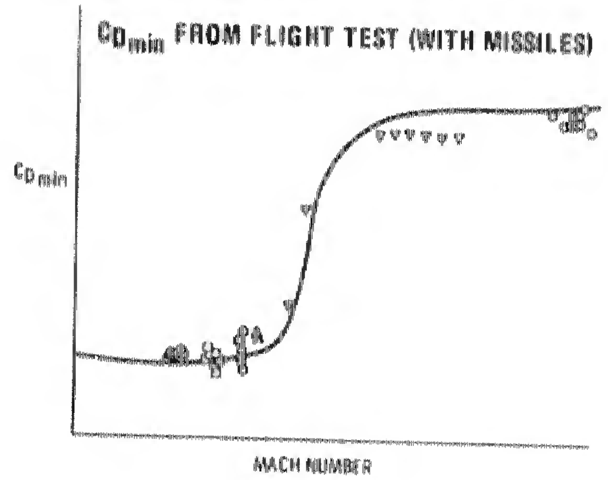


Figure 17

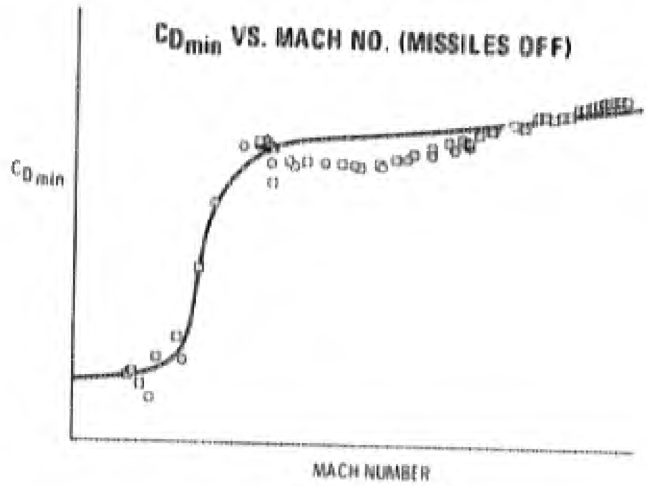


Figure 18

Reference (a) Dietz, William C., "Preliminary Design Aspects of Design-To-Cost for the YF-16 Prototype Fighter," presented at the AGARD Flight Mechanics Panel Symposium, October 1973, Florence, Italy.