SOME EXPERIENCES WITH NUMERICAL OPTIMISATION IN AIRCRAFT SPECIFICATION AND PRELIMINARY DESIGN STUDIES

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Abstract

The development of preliminary design methods to find the best configuration is discussed historically. A method for synthesising a combat aircraft using a large number of design variables, and the associated analysis procedures, is summarised. This is combined with a numerical optimisation method to obtain the minimum value of a nonlinear objective function subject to many nonlinear constraints defined in the design synthesis. Applications of this multivariate optimisation method are described to illustrate the effects of varying performance requirements and incorporating technological advances in the design. The dependency of the optimum configuration on the particular requirements specified and the need for a thorough investigation of the characteristics of the mathematical model that lead to the optimum configuration are emphasised. It is concluded that effort should be made to increase the acceptability of the multivariate optimisation techniques in the pre-feasibility stage of design as it offers a potentially valuable guide in selecting configurations for more detailed consideration.

I. Introduction

Aircraft design has always involved some degree of optimisation. In the very early days of powered flight the process of optimisation was reduced to making a few simple performance calculations (*eg* to determine the wing span to give an adequate climb gradient) and finding the lightest structure that could provide sufficient strength, once the designer had found a suitable power unit. The performance of the resulting machine was deemed satisfactory if lift exceeded weight and thrust exceeded drag. This situation did not last for long as individual designers gained experience and competition grew between them. Specifications for range and other aspects of performance were proposed by potential civil and military users of aircraft. Initially these tended to consist of requests for general performance improvements relative to existing aircraft operated by rivals, rather than considered specifications for particular roles. There was thus considerable scope for private-venture designs aimed at providing performance improvements in particular areas. Due to the relative simplicity of design, development and manufacture, the specification and evaluation of costs was not a significant feature of the design process.

However, by the late 1930s aircraft design teams endeavoured in their preliminary layouts to achieve a good compromise in the values chosen for the principal design parameters (wing area, engine size, etc) to meet a particular specification. Typically this was done by means of parametric studies, involving the interpretation of a series of plots to locate the appropriate configuration (*eg* see methods of Ref 1). This approach developed in complexity during the 1940s, with more requirements being added to the specifications and more aspects of design being considered during the initial layout phase. The arrival of the computer provided the power to do more complex analyses. Initial computer applications were confined to aspects of structural analysis and wing design. There was some resistance to the use of computers in initial project design because of the complex decision-making processes involved. However they enabled more detailed analyses to be made and hence allowed a greater range of carpet plots with additional overlays to be prepared to show the effects of configuration variables on performance.

The application of numerical optimisation techniques was first attempted in structural design[2], presumably because there was a sound analytic foundation for the design procedure, and the configuration variables chosen (*eg* skin thickness) had relatively limited effects on the overall design. There followed a number of attempts at applying numerical optimisation techniques to the initial layout of aircraft. These had little success because of the gross simplifications made in the aircraft design synthesis in order to obtain a solution within a reasonable time using the limited computing power available in the early 1960s. To try to circumvent this problem a Latin Square technique[3] was used. In this method a series of pivotal designs were first synthesised and analysed using the traditional initial project design methods. The results of these separate studies were used to define, by a least-squares fitting procedure, a polynomial surface of some objective function over the range of values of the principal design variables covered by the datum designs. Numerical optimisation algorithms were then applied to find the best value of the objective function over the polynomial surface. This method continues to be used[4,5] as an inexpensive means of finding the region of design space appropriate to a given set of operational requirements. However the approximations inherent in the least-squares fitting procedure do tend to reduce the value of the effort put into defining the pivotal designs. Furthermore these initial designs may not all have been done to the same degree of refinement so that an element of noise remains in the results, which might conceal important design trends.

Although the computing power available to designers grew steadily in the 1960s, and this was applied by several groups[6,7] to model aircraft mathematically to the level of detail necessary for initial project design, optimisation was applied directly in only a few cases[8]. The main reasons for this position were the continuing difficulties of interpreting and visualising the resulting configurations, and the lack of confidence in the mathematical algorithms used for obtaining optima. During this period serious effort began[9] on using computer graphics to assist the designer, partly as an adjunct to the installation of numerically controlled production methods but also to lessen the difficulties of communication between man and computer. Related to this work was the gradual assembly of design data bases containing such items as details of common aircraft components and libraries of analysis programs.

During the 1970s computing power increased while the cost per calculation fell, so that computing became a common currency for designers. The expanding application of computer-aided design and manufacture (CAD/CAM) removed much of the prejudice against automated design methods. Far more robust optimisation algorithms became available (notably for the optimisation of a nonlinear function, subject to nonlinear constraints[10]), and more complex design synthesis programs for initial project studies were formulated[11,12]. However the successful application of optimisation to configuration design was made first not by designers in the aircraft industry but by government agencies more concerned with specifying operational requirements, assessing project studies and exploring the likely effects of research advances[13-16].

The work I am going to describe was begun in 1976, following the development at RAE of multivariate optimisation applied to a transport aircraft design synthesis[14]. The terminology and method of application of the numerical optimisation technique[17] have recently been described in detail by Edwards[18]. I shall therefore concentrate on explaining in general terms the design synthesis method for combat aircraft (section II), as it has several major differences from the transport aircraft synthesis, and then go on to consider some recent applications of the method (section III).

II. Multivariate Optimisation for Combat Aircraft

General Organisation

In the description that follows the terminology introduced by Edwards[18] will be adopted. For convenience this is summarised in Fig 1. An important point to note here is that the roles assigned to various variables (as IV, DV or EV) in a mathematical model of an aircraft design are not unique. Many other formulations are equally valid and may have particular advantages for certain problems. The mathematical model is not arranged in a closed form with respect to the independent variables. A series of constraint functions are defined that have to be satisfied for any solution aircraft (*ie* set of values of the independent variables).. Which solution aircraft is the most appropriate is determined by the process of minimisation of an objective function that is also calculated for the mathematical model. The relationship between the design synthesis and analysis, the numerical optimisation, and the user is shown in simplified block-diagram form in Fig 2. The process and data paths shown in continuous lines are implemented on a computer, while the dashed lines indicate the user interaction. The significance of the variables (as IV, DV and EV) is also shown on this figure. It is possible to hold an independent variable constant during an optimisation, so that it effectively becomes an external variable. The numerical algorithm used is described fully by Purcell[17].

The complete program occupies 53K words of memory on an ICL 1906S computer, and the optimisation loop is traversed with a new set of IV values approximately 8 times/second on this processor. For a study with 18 independent variables and a slightly smaller number of active constraints (see section III), 4000 such loops are typically necessary. This corresponds to little more than 8 minutes of processor time, though substantially greater elapsed times may be needed in a multi-user environment.

A block diagram of the principal features of the design synthesis and analysis is shown in Fig 3. As can be seen this has been organised to allow a simple path through the synthesis and analysis. This simplification has been achieved by avoiding large iteration loops through the appropriate choice of independent variables and constraints (a typical set of independent variables and constraints are shown in Table 1, section III,

INDEPENDENT VARIABLE	(IV)	A variable quantity in the mathematical model of an aircraft, that is chosen to be at the disposal of the optimisation algorithm.
DEPENDENT VARIABLE	(DV)	A variable in the mathematical model whose value is dependent on the set of values of the independent variables and the external variables.
EXTERNAL VARIABLE	(EV)	A quantity that is variable externally to the optimisation process, *ie* part of the design data.
OBJECTIVE FUNCTION	(OF)	A variable in the mathematical model that is chosen as the parameter to minimise in the optimisation process.
EQUALITY CONSTRAINT	(EC)	An equality between two variables in the mathematical model that must be satisfied within a specified tolerance in the solution aircraft obtained by the optimisation process. This type of constraint will thus exert an influence throughout the optimisation and is therefore always ACTIVE.
INEQUALITY CONSTRAINT	(IC)	An inequality between two variables in the mathematical model that must be satisfied within a specified tolerance in the solution aircraft. This type of constraint remains INACTIVE unless a set of values of the IV would cause it to be broken. Thus the membership of the set of constraints that are ACTIVE in the optimisation will change as the IC values vary with different sets of values of the IV.
The optimisation is not terminated until all the constraints are satisfied within the specified tolerance		

FIGURE 1 - TERMINOLOGY

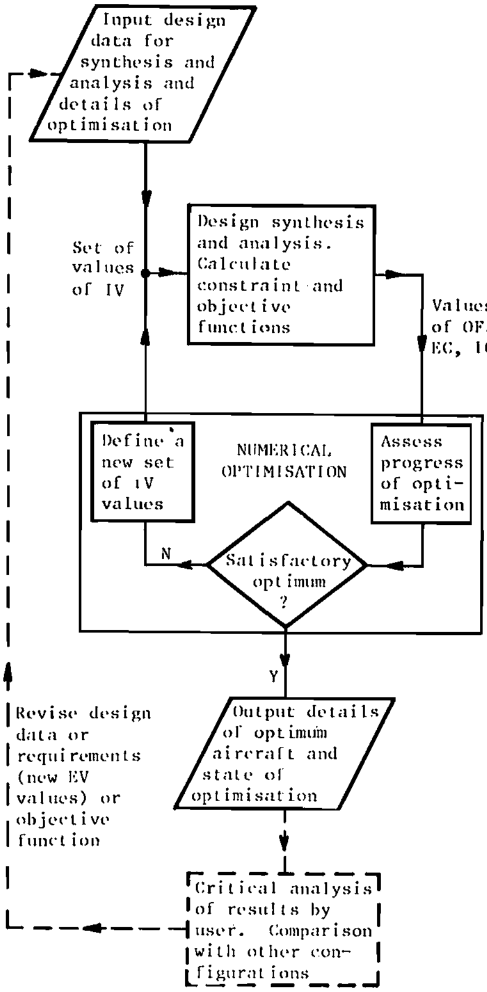


FIGURE 2 - MULTIVARIATE OPTIMISATION

below). For example, the matching of the fuel required for the specified sortie and the fuel that can be carried by the aircraft is achieved by an equality constraint monitored by the optimisation. It is not worth removing the smaller iteration loops in this manner as the benefits are outweighed by the additional computing time needed to handle the extra variables and constraint functions so introduced.

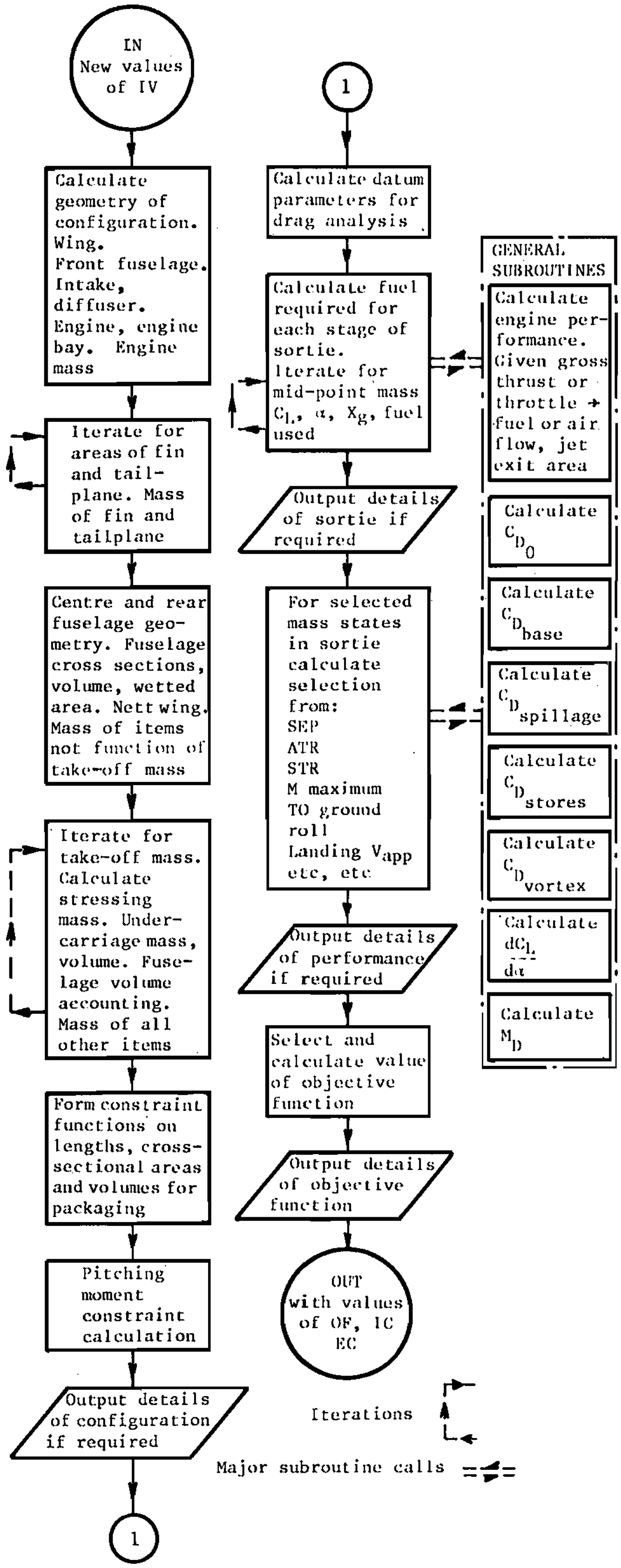


FIGURE 3 - DESIGN SYNTHESIS AND ANALYSIS

Synthesis of the Aircraft Configuration

Wing. A trapezoidal planform of constant thickness to chord ratio is assumed. The front and rear spars forming the wing box are located at fixed fractions of the chord, with the wing box being carried through the fuselage, at the wing root, normal to the fuselage centre line. Eight independent variables define the planform, box and high-lift system (gross wing area, gross wing aspect ratio, gross wing taper ratio, quarter-chord sweep, thickness to chord ratio, front and rear spar positions and trailing-edge flap span). The width of the box centre section is defined by the fuselage shape. It is assumed that fuel may be accommodated in the wing box and the proportion of the volume available that is used for this purpose is an independent variable. The fore and aft position of the wing relative to the fuselage is also an independent variable. Apart from bounds on the independent variables, the form of the planform is restricted by a boundary, expressed in terms of the wing aspect ratio and quarter-chord sweep, that corresponds to the limit of acceptable pitching-moment behaviour approaching the stall. Other features of the wing configuration (spoilers, ailerons, pylon mounting points, etc) are specified in the design data.

Empennage. The fin and tailplane are defined as the nett surfaces external to the fuselage. Nett aspect ratio, nett taper ratio, sweep and thickness to chord ratio are treated as design data. The trailing edges of both the fin and the tailplane are assumed to meet the fuselage at the engine jet exit plane. Two iteration loops are used to determine the fin and tailplane areas that provide the values for the fin and tailplane volume ratios specified in the design data.

Engine. The geometry of a datum engine is provided in the design data and scaled by means of a single independent variable defined as the ratio of the engine gross thrust to that of the datum engine. Four principal engine cross sections and the lengths of the gas generator, reheat fuelling section, reheat burning section, and nozzle are scaled separately with respect to this variable. The engine bay is defined in terms of the minimum clearances round the engine at the four principal stations along its length.

Cockpit. The cockpit geometry is defined using some fixed dimensions, extracted from the relevant US military standards, combined with values for the seat-back angle and the downward-view angle from the pilot's eye point. All these parameters are part of the design data and may thus be treated as external variables.

Fuselage envelope. The starting point for synthesising the fuselage envelope is a longitudinal distribution of cross-sectional area, including the stream tubes for the intake and the intake boundary-layer diverter (Fig 4). This comprises forward and aft fairings, defined by cubics, joined by a cylindrical centre section. Options are available to treat this as either a longitudinal distribution of *gross* cross-sectional area (and thus effectively apply the sonic area rule to the body), or as a longitudinal distribution of *fuselage* cross-sectional area. A radar of specified diameter is assumed to be fitted in the nose and the area at this fuselage station, together with the corresponding slope of fuselage cross-sectional area with axial distance (also specified in the design data), are used as end conditions for the cubic variation of the forward fairing for cross-sectional area. Six independent variables (IV) define the area distribution as shown in Fig 4. An inequality constraint is placed on the maximum boattail angle over the rear fuselage, and an equality constraint is used to make the gross nozzle exit area match that obtained from the scaled engine.

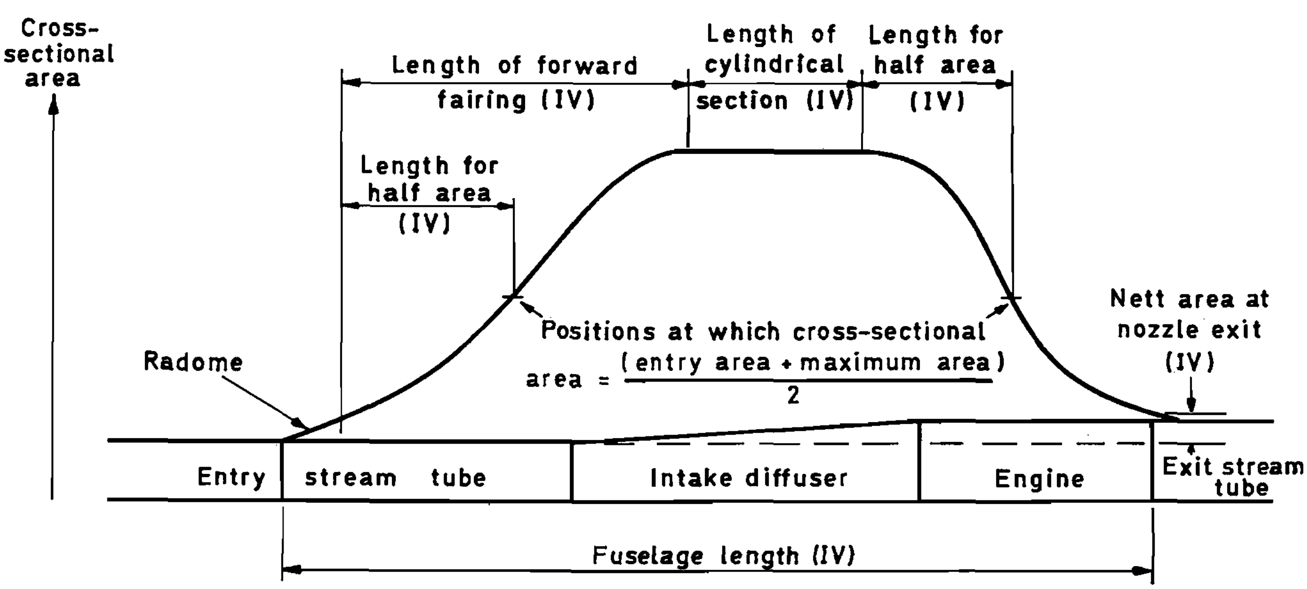


FIGURE 4 - LONGITUDINAL DISTRIBUTION OF CROSS-SECTIONAL AREA FOR THE FUSELAGE

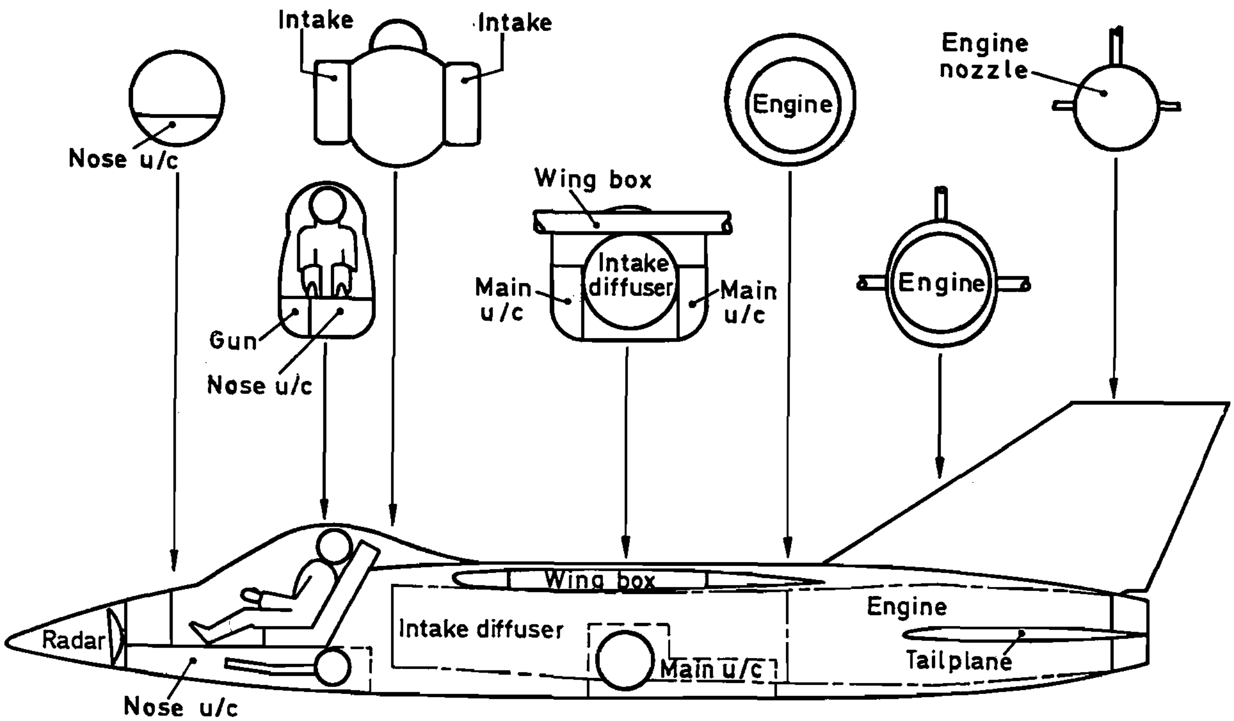


FIGURE 5 - FUSELAGE PACKAGING SHOWING STATIONS FOR CONSTRAINTS ON CROSS-SECTIONAL AREA

Fuselage packaging. Three aspects are considered - lengths, areas and volumes - and each leads to a set of constraints. The lengths of all the major components are calculated from the design data and by applying any scaling for component size that may be necessary. The length of the intake diffuser is determined by subtracting the engine length and the distance of the intake aft of the nose (derived from the design data) from the fuselage overall length. A minimum length for the diffuser is defined in terms of the intake area and used to form an inequality constraint with the actual diffuser length. The longest of four alternative combinations of lengths of items that might define the minimum fuselage length is used to form an inequality constraint (*eg* one combination considered is the distance of intake aft of nose + minimum separation distance between intake lip and wing leading edge + wing chord at body side + minimum separation between wing trailing edge and tailplane leading edge at body side + tailplane chord at body side).

The second class of constraints on the fuselage packaging is concerned with whether the necessary items of equipment can be fitted into the available cross-sectional areas. Seven critical fuselage stations are defined (see Fig 5) and the minimum cross-sectional area necessary to accommodate the particular items for the layout shown in Fig 5 is calculated. An inequality constraint is formed for each station from the difference between the actual cross-sectional area (calculated from the longitudinal distribution of cross-sectional area) and the minimum cross-sectional area.

The third class of constraints is concerned with volume accounting. The minimum volumes for the front fuselage forward of the intake and the rear fuselage aft of the front face of the engine compressor are determined by summations of the volumes of items that must be installed in these sections. Apart from those items already referred to, these include a gun and ammunition, radar, other avionics, electrical systems, hydraulic systems, air systems and control systems. Details for the first three are included in the design data, while the latter four and the undercarriage size are related to the aircraft size. The actual volume of the front, centre and rear fuselage sections is calculated by integrating the longitudinal distribution of cross-sectional area between the appropriate limits. Two inequality constraints are then defined as the difference between the actual and minimum volumes for the front and rear fuselage sections. Any spare volume remaining in the front, centre and rear fuselage sections, after accounting for the volumes of all the items mentioned above, is assumed to be potentially available for fuel storage, and three corresponding volume utilisation factors for fuel are introduced and treated as independent variables. The upper bounds to these variables are chosen to allow for the relative ease of making use of the volumes for storing fuel in these three locations.

Analysis of the Aircraft Configuration

Mass estimation. The masses of the components of the configuration are determined from empirical correlations of the masses of particular components on existing combat aircraft. These correlations are formulated in terms of the major design parameters. For large items such as the wing and fuselage the correlations involve a detailed breakdown of the component masses (*eg* wing structural box, fixed leading-edge structure, fixed trailing-edge structure, flaps, spoilers, etc), and were derived by design engineers in the UK aerospace industry. For items which do not need to be determined so precisely in the synthesis (empennage, electrical and other systems) simpler correlations are sufficient. The mass at take-off is calculated by summing the component masses and assuming that all the available fuel tankage is full (for the particular values of the volume utilisation factors (IVs)). This mass is used to redefine the stressing mass used in the wing and fuselage structural mass calculations, and the process of estimation of component masses is repeated until convergence is obtained. To determine the fuselage structural mass the volume and surface area of the fuselage are required. The latter is calculated from the longitudinal distribution of the fuselage cross-sectional area by using a longitudinal distribution of a shape parameter defined as the ratio of the circumference of a particular section to the circumference of a circle of the same area.

The component masses obtained are then used to determine the contribution of the component to the static pitching moment about a moments centre, specified in the design data relative to the aerodynamic centre. The resultant moment is made an equality constraint that must be reduced to zero by the appropriate choice of the wing fore and aft position in the numerical optimisation.

Aerodynamic Analysis, General. In order to reduce computation to a minimum the analysis is confined to the minimum necessary for the calculation of the sortie and point performance parameters described below. Thus while a comprehensive drag analysis is essential, a simple estimate of the aircraft attitude to achieve a given lift is sufficient as the attitude only influences the performance parameters via the components of gross thrust normal and parallel to the flight path. Similarly the maximum attainable lift is currently estimated for a class of aircraft, using a specified level of technology for the high-lift devices. It is tabulated versus Mach number as part of the design data. The drag of the stores and associated pylons are also tabulated versus Mach number in the design data.

Lift-curve Slope. For subsonic Mach numbers a method given in the USAF Datcom[19] is used to estimate the lift-curve slope. A section lift-curve slope with a correlation for viscous effects is first determined and corrections for finite wing, sweep and Mach number effects applied to obtain the wing lift curve slope at a given Mach number. It is assumed that the body contributes sufficient lift to match the lift that would be produced by an increase in area from the nett to the gross wing planform. At supersonic speeds where aircraft attitude is generally low, mainly because of loading limitations, it is sufficient to use linear theory to estimate lift-curve slope. The subsonic and supersonic variations of lift-curve slope with Mach number are faired together by means of a cubic in the transonic regime.

Drag. The total drag is obtained as the sum of the basic airframe zero-lift drag, zero-lift wave drag, lift-dependent drag, base drag related to the jet exit area, spillage drag, intake momentum drag and store drag. In order to minimise the amount of computation the calculation of these drag components is split into two stages. In the first stage those parameters that are purely configuration dependent (*eg* the wetted areas and form factors in the basic zero-lift drag calculation) are calculated and in the second stage those parameters that are dependent on Mach number, altitude or engine throttle setting are calculated. Thus only the second stage of the calculation needs to be repeated for the set of flight conditions corresponding to the specified sortie and point-performance parameters. The variation of the airframe zero-lift drag with Mach number is built up in the following manner. The basic zero-lift drag above a Mach number of 0.8 is calculated by applying a factor, which is a function of Mach number, to the basic zero-lift drag at 0.8M. An allowance is made in this basic drag for excrescences, control gaps and interference effects. To this basic drag is added wave drag. Initially this is defined by the increment and slope of the wave drag coefficient with Mach number at three points; the drag rise Mach number, $M = 1$ and $M = 1.3$. Supersonic base drag is included in this calculation as it is an inherent part of the afterbody drag. The wave drag at other Mach numbers is then determined by interpolation. The drag components that are functions of engine performance are calculated for the values of air mass flow and jet exit area corresponding to the particular flight conditions. The intake momentum drag is calculated to allow a more accurate formulation of the horizontal and vertical equations of motion to be made in the sortie and point performance calculations.

Lift-dependent drag is represented as a two-part linear function of C_L^2. The value of C_L at which the change in slope of the lift-dependent drag occurs is assumed to be a fixed function of Mach number for a given family of wings, as it is mainly dependent on the camber variation capabilities incorporated in the wing design. The lift-dependent drag factor, k_1, below this critical C_L is calculated for Mach numbers below 0.8 and supersonically at $M = 1.4$ and 2. A linear variation of the supersonic k_1 through the latter two points is assumed, and a cubic fairing used to join the subsonic and supersonic values of k_1. The variation with Mach number of the slope of the lift-dependent drag with C_L^2, k_2, above the critical C_L is defined relative to k_1 in the design data.

Engine Performance. Brochure performance data for a typical modern turbofan are used as the basis for defining the variation of engine gross thrust, fuel and air mass flow, and jet exit area with altitude and Mach number. These data are non-dimensionalised with respect to sea-level, static conditions. The variation of the engine parameters with throttle setting is divided into sections corresponding to the different operating regimes of the engine. Separate correlations are used for the reference conditions defining the boundaries of each section and the variation through these sections, with provision for smooth matching at the junctions of the sections. For the operating regime between combat dry thrust and minimum reheat thrust, for which no brochure data is available, arbitrary performance correlations with gross thrust are constructed to ensure a smooth transition from dry to reheat thrust conditions.

As many of the aircraft performance

calculations are requested at particular combinations of height and Mach number these conditions are specified in the design data, and hence the preliminary calculations to find the reference conditions for the engine performance at the given flight conditions may be done and the results stored before embarking on the optimisation loop shown in Fig 2. Thus when the thrust required is known (*eg* when the drag of the synthesised aircraft has been estimated) the engine reference conditions for the particular height and Mach number may be retrieved and the dimensional values of the performance parameters calculated. For some types of aircraft performance calculation, such as maximum speed, the complete calculation must be repeated for each approximation to the maximum Mach number.

Sortie Performance. The main purpose of estimating the sortie performance of the synthesised aircraft is to determine how much fuel is required for a combination of stages flown at specified altitudes and Mach numbers. The aircraft flight path need only be considered in sufficient detail to enable the sortie fuel to be determined. The fuel required for take-off is calculated by assuming the use of a given engine throttle setting for a fixed time interval. The fuel required for climbs, accelerations and descents is assumed to be allowed for by suitable modification of the stage range. Reserve fuel is specified as a fixed percentage of the total that the aircraft can carry.

Each stage of the sortie is specified in the design data by an altitude, Mach number, normal load factor and duration. To allow some optimisation of the sortie the altitude and Mach number for two of the stages may be treated as independent variables. The duration of the stage may be expressed as a time or range or a number of sustained turns in the horizontal plane. The payload state of the aircraft is specified by the numbers of the stages at the beginning of which bombs or external fuel tanks are dropped, missiles fired or all the ammunition used. The aircraft lift, attitude and drag are then estimated at the average mass of the aircraft for each stage of the sortie. Resolution of the vertical and horizontal forces on the aircraft enables the gross thrust required and hence also the fuel mass flow to be calculated. This leads to revised estimates for the average mass of the aircraft during a stage, and the engine-related drag contributions. Iterations of this form continue until satisfactory convergence is obtained. Apart from giving the mass of fuel used during a stage, the aircraft mass at the end of a stage is retained for subsequent use in the point-performance calculations. The values for lift, attitude, drag and thrust are also saved to provide an improved starting point for later iterations to find the fuel consumed in a particular stage at the next entry to the design synthesis and analysis section of the multivariate optimisation, when a new set of independent variable values have been defined in the numerical optimisation routines (Fig 2). Comparison of the estimates for fuel consumption obtained by the simple method described above, with the value obtained by splitting the stages into a large number of elements and integrating for the total fuel consumption has shown a difference of less than 0.2% over a wide range of altitudes, Mach numbers and throttle settings.

To ensure that the optimum synthesised aircraft can carry sufficient fuel to fly the specified sortie an equality constraint is formed from the difference between the fuel load, calculated during the volume accounting described previously, and that consumed during the sortie (including reserves). Provision is also made for an inequality constraint on the fuel consumed before an air-combat stage is reached in the sortie, to make this greater than the fuel stored in the exposed portion of the wing box, should this be thought desirable.

Point Performance. This term is used to cover miscellaneous calculations of aircraft performance, apart from range performance. The field performance is always calculated, using the aircraft mass information obtained from the sortie performance calculation. Simple correlations for the take-off ground-roll distance and flight distance from unstick to 15 metres altitude are formulated in terms of thrust to weight ratio, wing loading and the maximum C_L available. The approach speed for landing is derived from the stalling speed of the aircraft when in the landing configuration.

All the other performance calculations are optional, and the following parameters may be determined: sustained turn rate, instantaneous (attained) turn rate, specific excess power, maximum Mach number, time to accelerate over an increment in Mach number, ride quality and certain combinations of the turning performance parameters that may be used as a measure of manoeuvrability (*eg* the application referred to in section III). For each desired parameter a Mach number, an altitude, an engine throttle setting, and a mass state (from the sortie performance) are specified in the design data. Where necessary the horizontal and vertical equations of motion are solved iteratively to find the aircraft attitude, and hence the component forces. An extended iteration is necessary in the determination of the maximum Mach number, wherein the horizontal equation of motion is solved for the highest velocity at which there is no nett horizontal force in 1 g level flight. As in the sortie performance calculations the resultant values of attitude and forces are stored for use as starting conditions at the next entry to the design synthesis and analysis for which the particular calculation is required. A given parameter (*eg* SEP) may be calculated at several combinations of altitude, Mach number and mass state.

The results of all these calculations may be used in three ways, as directed by the user in the design data. Some calculations may be made purely for information purposes, and hence have no influence on the design optimisation. These calculations are therefore only performed for the initial and final synthesised aircraft. Other calculations are made in order to specify the point performance required for an aircraft and are therefore used to generate constraint functions from the difference of the actual and desired values of the performance parameters. These will normally be inequality constraints and hence are calculated only when the constraint is a member of the active set of constraints, and for the initial and final synthesised aircraft. Finally some performance calculations may be required for every synthesised aircraft in order to determine the value of the objective function used in the optimisation.

Objective Function. The last stage of the analysis of each synthesised aircraft is the calculation of an objective function to be minimised in the numerical optimisation process. Four alternative objective functions are available and the choice is made via the design data:

(1) aircraft total mass at take-off. This parameter has traditionally been used in parametric studies to determine an optimum configuration, and is broadly related to aircraft cost;

(2) aircraft empty mass. By minimising aircraft mass without fuel and other items dependent on a particular sortie a better approximation to purchasing cost is possible;

(3) unit production cost. A simple estimate of unit production cost is made by considering the cost of labour and materials per unit mass of the major components of the aircraft (structure, engine, avionics, etc);

(4) cost/performance compound function. A linear combination of unit production cost and a manoeuvrability performance parameter, including the sustained and attained turn rates, is formed using two weighting constants specified in the design data. The first, positive, constant provides an element of minimisation of unit production cost, while the second, negative, constant provides an element of maximisation of the performance parameter. By altering the relative magnitude of the weighting constants a family of aircraft can be synthesised to show the interrelation between cost and manoeuvre performance for a particular set of fixed requirements.

Unfortunately there is no agreed datum costing method for combat aircraft corresponding to the direct operating cost of transport aircraft, but some form of life-cycle costing is often attempted for new designs. The above objective functions can be used to form the basis of a more complete cost analysis.

III. Applications

To illustrate the use of multivariate optimisation for combat aircraft, some aspects of a prefeasibility study for a hypothetical strike aircraft are described. In this work the sortie shown diagrammatically in Fig 6 was used for the aircraft which had to carry a weapon load of 2000 kg. Associated with this sortie are two independent variables (the cruise Mach numbers for legs 1 and 5), an equality constraint on the fuel load, and three inequality constraints on performance to ensure the aircraft can attain the required penetration Mach number and sustained turn rate when carrying the weapons, without recourse to reheat, and to ensure a ride quality that is sufficiently good to allow accurate delivery of the weapons. The set of 18 independent variables and 24 basic constraints is listed in Table 1. Some of the variables referred to in section II have been kept constant for this application. In view of the vulnerability of a strike aircraft to ground fire the volume utilisation for fuel in the rear fuselage (*ie* adjacent to the engine) was fixed at zero, to remove a potential fire hazard. The chordwise positions of the front and rear spars of the wing and the spanwise extent of the trailing-edge flap were fixed, which, because the high-lift devices are positioned at fixed chord fractions in front of the front spar and behind the rear spar, results in the leading- and trailing-edge high-lift devices having a constant proportion of the wing area. The altitude for the cruise legs in the sortie was fixed at 1500 metres.

Because of the nature of the prime role of this aircraft, subsonic strike, the option allowing the application of the sonic area rule in the synthesis of the fuselage was not used. However the mass and volume specified for avionics in the fuselage was sufficient to allow for other, subsidiary, roles.

Effect of Requirements

With unit production cost (UPC) as the objective function the multivariate optimisation method was used to determine the optimum set of values for the independent variables subject to the basic set of constraints. Of the 18 independent variables, two reached the bounds defined in the design data and remained on these bounds throughout the applications described below; a lower bound of 0.2 set for the taper ratio (IV5), and an upper bound of 0.5 set for the volume utilisation for fuel in the front fuselage (IV15). The latter bound was set at this low value to allow for the difficulty of using volume for fuel in the cockpit area. A further variable, the volume utilisation for fuel in the exposed wing portion of the wing box (IV7) also reached its upper bound (1.0) for this first optimisation. In addition to the three equality constraints, five of the inequality constraints were active in the optimum configuration. The fuselage shape was influence by two constraints on cross-sectional area (IC4 and IC8), and the length determined by the minimum allowable length of the intake diffuser (IC16). The wing planform and engine scale were determined by two of the performance requirements, the maximum Mach number and sustained turn rate at 100 metres altitude on dry engine thrust (IC19 and IC20 respectively). Values of some of the major variables for the optimum aircraft are shown in the first column of Table 3 (aircraft reference A). The aircraft has a small wing to reduce zero-lift drag, but a high aspect

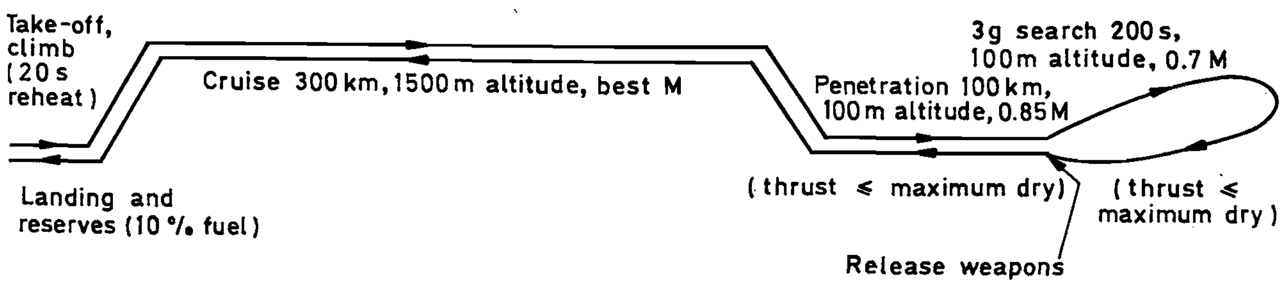


FIGURE 6 - SORTIE FOR STRIKE AIRCRAFT STUDY

INDEPENDENT VARIABLES	
IV1	Engine scale
IV2	Wing area
IV3	Sweep of wing at 0.25 chord
IV4	Thickness : chord ratio of wing
IV5	Taper ratio of wing
IV6	Aspect ratio of wing
IV7	Proportion of wing box volume in exposed wing used for fuel
IV8	Fuselage length
IV9	Length of forward fairing of fuselage
IV10	Length for half area on forward fairing
IV11	Length of cylindrical section of fuselage
IV12	Length for half area on aft fairing
IV13	Distance of wing mean 0.25 chord point aft of nose
IV14	Fuselage cross-sectional area at engine nozzle exit
IV15	Proportion of front fuselage volume used for fuel
IV16	Proportion of centre fuselage volume used for fuel
IV17	Mach number for outbound cruise in sortie
IV18	Mach number for inbound cruise in sortie

EQUALITY CONSTRAINTS	
EC1	Difference between actual and derived nozzle exit areas
EC2	Nett moment about specified CG position
EC3	Difference between fuel available and required

INEQUALITY CONSTRAINTS	
IC1	Separation between wing and tailplane
IC2	Pitch-up limitation on aspect ratio and sweep of wing
IC3	Fuselage cross-section at front of cockpit
IC4	Fuselage cross-section at centre of cockpit
IC5	Fuselage cross-section at engine intake
IC6	Fuselage cross-section at main undercarriage bay
IC7	Fuselage cross-section at engine compressor
IC8	Fuselage cross-section at engine gas generator
IC9	Fuselage cross-section at start of nozzle
IC10	Minimum fuselage boattail angle
IC11	Maximum fuselage boattail angle
IC12	Separation between intake and wing leading edge
IC13	Separation between main undercarriage and engine
IC14	Separation between wing box and engine
IC15	Minimum fuselage length
IC16	Minimum intake diffuser length
IC17	Front fuselage volume
IC18	Rear fuselage volume
IC19	$M \geqslant 0.85$ on outbound penetration leg of sortie
IC20	Sustained turn rate $\geqslant 3$ g in search leg of sortie
IC21	Ride quality at 0.85M

TABLE 1 INDEPENDENT VARIABLES AND BASIC CONSTRAINTS FOR STRIKE AIRCRAFT STUDY

ratio to maintain low lift-dependent drag while still providing the desired turning performance in the sortie. As a result the wing mass loading is high, and the top speed at 10 km altitude and the low-level ride quality are good. The optimum value for the outbound cruise Mach number is greater than the inbound cruise Mach number because the higher aircraft mass, due to the stores and higher fuel mass, leads to a larger increase in lift-dependent drag than the increase in zero-lift drag due to the stores.

In a pre-feasibility study the investigation of a type of design would not be restricted to the consideration of the simple performance requirements included in the inequality constraints listed in Table 1. To indicate the range of requirements that might be considered, the series of additional inequality constraints listed in Table 2 were added in succession to obtain a set of synthesised aircraft, each optimised for minimum unit production cost. Some details of the aircraft obtained in this manner (referred to as aircraft B to G below) are given in Table 3.

The first two additional constraints are concerned with airfield performance. An aircraft of the type being considered would be based relatively close to the battlefield and hence would need a good field performance in order to be able to operate after substantial runway damage. Adding a requirement for a landing approach speed of not greater than 75 m/s (IC22) produces aircraft B. Wing area is almost doubled compared with aircraft A in order to reduce the approach speed from the previous value of 102 m/s. Because of this effect on wing area, the new constraint (IC22) replaces the constraint on the sustained turn rate during the search leg of the sortie (IC20) as a member of the active set of constraints, which is otherwise unchanged. The maximum Mach number in the sortie (IC19) effectively sizes the engine. To minimise the mass of the much larger wing the aspect ratio is decreased and wing thickness increased. Because of the increased thickness, the leading-edge sweep of the wing is also increased to prevent an increase in the drag rise at high subsonic speeds. As a result engine size is only marginally larger. The increase in zero-lift drag coefficient leads to lower cruise Mach numbers.

A requirement for a take-off ground roll of not greater than 500 metres (IC23) was added next to produce aircraft C in Table 3. This constraint becomes a member of the active set of constraints, and the landing constraint (IC22) ceases to be active. Because of the dependence of take-off ground roll on thrust to weight ratio the take-off constraint leads to an increase in engine size. This larger engine enables the maximum Mach number required in the sortie (forming IC19) to be

IC22	Speed for landing approach $\leqslant 75$ m/s
IC23	Ground run for take-off $\leqslant 500$ m
IC24	Maximum Mach number at 10 km altitude $\geqslant 2$
IC25	Specific excess power at 0.9M, 10 km $\geqslant 100$ m/s
IC26	Sustained turn rate at 0.9M, 10 km $\geqslant 5.5^{\circ}$/s
IC27	Attained turn rate at 0.9M, 10 km $\geqslant 10^{\circ}$/s

TABLE 2
ADDITIONAL CONSTRAINTS TO SPECIFY REQUIREMENTS

Aircraft reference	A	B	C	D	E	F	G
Requirement	Basic	A + landing (IC22)	B + take-off (IC23)	C + M_{max} (IC24)	D + SEP (IC25)	E + STR (IC26)	F + ATR (IC27)
Engine scale	1.470	1.475	1.517	1.513	1.879	2.190	2.169
Wing area m^2	9.76	18.52	21.86	22.21	21.17	30.08	31.31
Leading-edge sweep ^{o}	18.95	41.08	40.91	42.56	15.12	18.03	18.24
Thickness/chord	0.071	0.099	0.103	0.077	0.053	0.048	0.048
Aspect ratio	5.514	3.642	4.004	3.437	3.370	3.057	2.914
Fuselage length m	12.62	12.63	12.71	12.71	13.41	13.95	14.15
Forward fuselage fraction	0.332	0.329	0.327	0.326	0.346	0.372	0.350
Aft fuselage fraction	0.371	0.234	0.353	0.229	0.198	0.164	0.144
Maximum cross-section area m^2	2.168	2.169	2.181	2.180	2.253	2.457	2.442
Wing position/fuselage length	0.561	0.568	0.572	0.572	0.580	0.596	0.588
Outbound cruise M	0.553	0.519	0.494	0.511	0.533	0.520	0.527
Inbound cruise M	0.538	0.506	0.482	0.498	0.519	0.504	0.512
Take-off mass kg	11939	12153	12409	12490	13592	15132	15227
Fuel kg	2712	2724	2743	2788	3164	3557	3571
Landing approach m/s	102.0	75.0	70.1	69.7	74.8	66.8	65.8
Take-off ground run m	1070	582	500	500	500	374	368
Maximum M 10 km altitude	2.25	1.97	1.89	2.00	2.24	2.23	2.24
SEP at 0.9M, 10 km m/s	73.2	87.3	88.6	89.4	100.0	106.4	103.6
STR at 0.9M, 10 km ^{o}/s	2.98	5.03	5.71	5.49	4.74	5.50	5.50
ATR at 0.9M, 10 km ^{o}/s	3.59	7.50	8.65	8.78	7.45	9.66	10.00
Relative unit production cost	1.000	1.017	1.039	1.042	1.110	1.214	1.221

TABLE 3 AIRCRAFT SYNTHESISED TO MEET A RANGE OF REQUIREMENTS

obtained with a slightly larger wing, having lower leading-edge sweep and increased thickness and aspect ratio. The decrease in lift-dependent drag leads to a further decrease in the optimum cruise Mach numbers.

Having considered the performance of the aircraft when employed on the specified sortie it is necessary to determine what influence other possible subsidiary roles might have on the design. Assuming that the fuel capacity needed to fulfil the prime role is sufficient for the other proposed roles, it is only necessary to consider the point performance of the aircraft. For example, it is reasonable to expect a strike aircraft to have an adequate supersonic performance at altitude, in the clean state. The inequality constraint IC24 was introduced to specify a maximum Mach number of not less than 2 for the aircraft, at an altitude of 10 km, when using the maximum available reheat thrust. Details of the optimised configuration are shown in column D of Table 3. All the constraints previously in the active set for aircraft C remain, with the addition of a further fuselage cross-sectional area constraint (IC9), and the new performance constraint (IC24). To reduce supersonic drag leading-edge sweep is increased, and wing thickness to chord ratio and aspect ratio are decreased. As constraint IC19, the maximum Mach number on dry thrust at 100 metres altitude, still remains active a suitable compromise has been reached for the values of these independent variables to just satisfy both inequalities. A slight increase in wing area is necessary to maintain the take-off run requirement (IC23). Engine size is decreased but unit production cost rises because of the increased wing mass and fuel consumption. The latter affects the structural mass via the stressing mass that is defined from the take-off mass. The drop in zero-lift drag causes the optimum cruise Mach numbers in the sortie to be higher than for aircraft C.

The ability of the aircraft to climb or

accelerate was next considered by specifying that it should be able to achieve a specific excess power (SEP) of at least 100 m/s at 0.9 Mach number and 10 km altitude, when in the clean condition and using the maximum reheat thrust available (IC25). Some details of the optimum configuration for this set of requirements are shown in column E of Table 3. The additional requirement necessitates an increase in engine size, and the requirements for maximum Mach number (IC19 and IC24), that had previously determined engine size, cease to be members of the active set of constraints. Another constraint on the fuselage cross-sectional area (IC7) becomes active. To achieve the required SEP the wing thickness to chord ratio is substantially reduced. As there is more than adequate thrust to achieve the supersonic performance required the sweep of the wing is greatly reduced to reduce wing mass. Wing area is also reduced just sufficiently to counteract the effect of the increased thrust in improving the take-off performance. Aspect ratio is also reduced, to lower wing mass, as sufficient thrust is available to match the corresponding increase in lift-dependent drag.

The potential of the aircraft in manoeuvring flight was specified in two further stages of requirements. Firstly a sustained turn rate (STR) of at least 5.5°/s was set for the clean aircraft, using full reheat thrust and operating at 0.9M and 10 km altitude, via constraint IC26 (details in column F of Table 3). To achieve this STR substantial increases in engine size and wing area are necessary, and all the other previously active performance constraints are easily satisfied (and hence are removed from the active set of constraints). However the increase in wing area is limited by ride quality considerations and the associated constraint (IC21) becomes active. A decrease in aspect ratio and thickness to chord ratio and an increase in wing sweep all help in reducing the lift-curve slope, an important factor in improving ride quality. The increase in wing area and decrease in aspect ratio result in a larger wing chord at the body side and cause the constraint on the minimum separation distance between the wing trailing edge and the tailplane leading edge to become active (IC1). One of the fuselage cross-sectional area constraints (IC8) ceases to be active. The increase in wing area leads to an increase in zero-lift drag (greater wetted area) and a decrease in lift-dependent drag, and hence lower values for the optimum cruise Mach numbers in the sortie.

The second stage in specifying manoeuvre performance was the addition of a constraint (IC27) for an attained turn rate (ATR) of at least 10°/s under the same conditions as for the sustained turn rate requirement. Details of the optimum synthesised aircraft are given in the last column of Table 3 (aircraft G). Attained turn rate is dependent on the maximum lift that the aircraft can produce, including the resolved component of gross thrust that supplements aerodynamic lift, and hence the required ATR is met mainly by an increase in wing area. Associated with this is a slight increase in sweep and decrease in aspect ratio to prevent ride quality deteriorating below the level specified (IC21). Engine size is decreased, as the increased wing area also assists the sustained turning performance, until constraint IC26 is just satisfied (*ie* it remains active). The further increase in wing area and decrease in aspect ratio again cause the wing chord at the fuselage side to increase, and the fuselage length is increased in order to satisfy the constraint on wing-tailplane separation (IC1). The longer fuselage can accommodate more fuel and as a result fuel that was accommodated in the wing box of the exposed wing is stored in the fuselage centre section, in spite of the bending relief offered by wing storage. The variable for fuel volume utilisation in the wing (IV7) goes onto its lower bound (0.0), whereas in all the previous cases (A to F) it had been on its upper bound (1.0). Another of the fuselage cross-sectional area constraints (IC4) ceases to be active.

The above discussion has concentrated on the effects of the additional performance requirements on the set of active constraints for the optimum configurations. The shape of the fuselage and the wing planform change considerably under the successive influence of the members of this set of requirements. Unit production cost increases by 22% from configuration A to configuration G. The particular values of the performance parameters that are used to form the constraints were chosen to show a progressive increase in unit production cost, the objective function in the optimisation. In a practical study a designer is likely to be interested in the effect on the optimum design configuration of variations in some of the requirements.

Perturbations in Requirements

Perturbations in two of the basic requirements for the strike aircraft are considered. In both cases the optimum configuration reference E in Table 3 was used as a datum, with unit production cost again used as the objective function. In the first set of optimisations the penetration range in the sortie was varied ±50% relative to that defined in Fig 6. A selection of the results are shown as

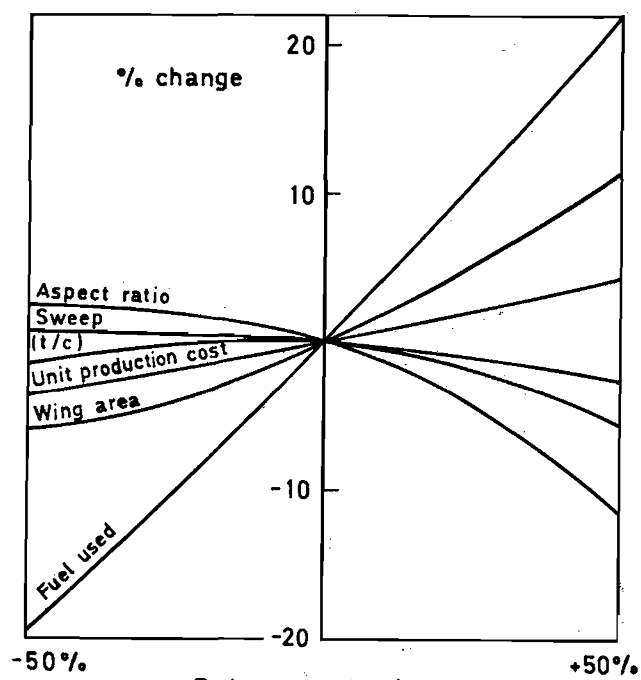


FIGURE 7
EFFECT OF PENETRATION RANGE ON CONFIGURATION

percentage changes, relative to the values for aircraft E, in Fig 7. While some parameters, such as the objective function and the fuel used, have an approximately linear variation with the external variable (penetration range), the other parameters show a significantly nonlinear variation. Although the variation of the objective function with the external variable will be continuous there may well be discontinuities in slope or even value in the variation of the optimum values of the independent variables with the external variable, as has been shown by Edwards[18]. At the shortest range the active set of constraints includes the one on approach speed for landing (IC22) but not the one for take-off (IC23), the reverse of the situation for the datum range and highest range. This occurs because of the higher proportion of the take-off mass that remains at landing for the shortest penetration range. Examining the departure of the wing area variation from a linear form it is apparent that relatively more wing area is required for the shorter ranges. The decrease in aspect ratio may be related to the variation of lift-dependent drag with penetration range. Because of the lower fuel use the shorter range aircraft have a higher average mass, relative to take-off mass, and hence a greater lift is required relative to the initial value. The change in aspect ratio is determined by the balance of the zero-lift and lift-dependent drag components to produce the optimum cruise conditions.

A second set of optimisations was made in which the weapon load was varied ±50%, some of the results of which are shown in Fig 8. Again some variables have a near linear variation with change in the weapon load, but others are highly nonlinear. The membership of the active set of

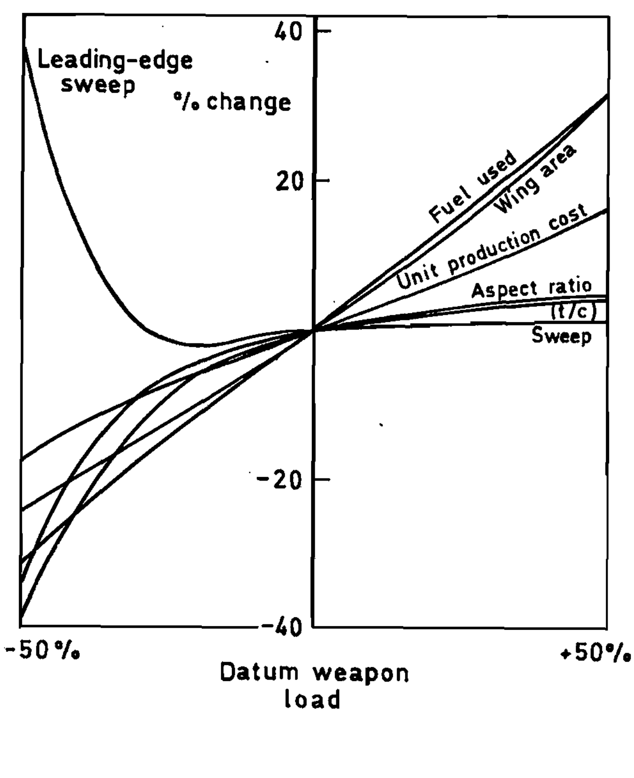


FIGURE 8 EFFECT OF WEAPON LOAD ON CONFIGURATION

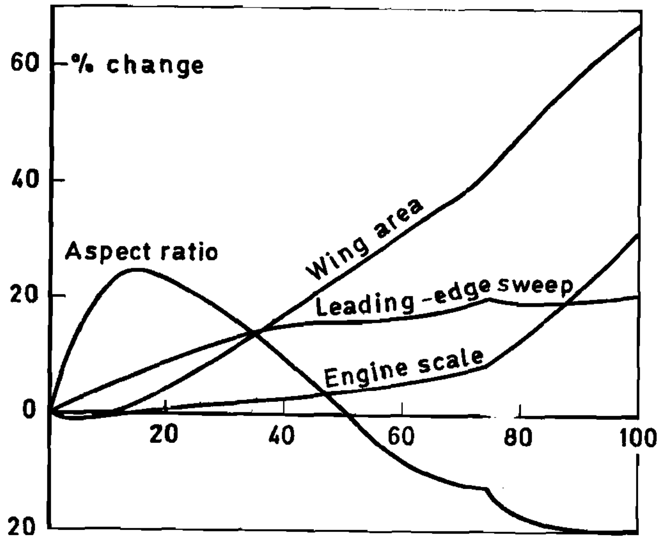
constraints varies markedly as the weapon load is decreased from the datum. The constraints on three of the fuselage sections and the constraint on SEP (IC4, IC8, IC9 and IC25 respectively) remain active throughout. The constraint on take-off ground roll (IC23) is replaced by the constraint on the speed for landing approach initially, in a similar manner to that noted when penetration range was varied. However at the smallest weapon load considered this situation is reversed and, in addition, the constraints on STR at 0.9M at 100 metres altitude and maximum Mach number at 10 km altitude become active. These constraints lead to a great change in the optimum configuration to a low aspect ratio, thin wing with a more highly swept leading edge.

An important conclusion from these calculations is that the complete reoptimisation of a configuration, when the requirements are varied, often leads to a highly nonlinear variation, or even to a variation with discontinuities, in the optimum values of the independent variables with the external variable (requirement). It is thus unwise to use derivatives of the objective function at a datum configuration to estimate the effect of perturbations in a requirement or any other external variable.

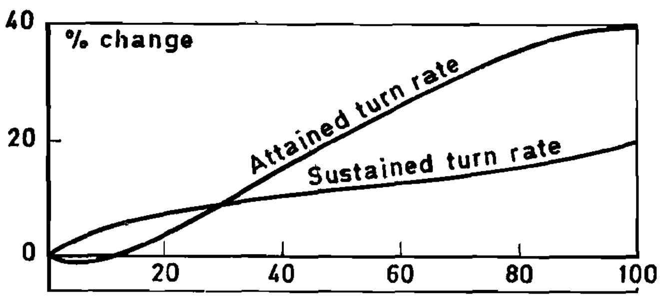
Use of a Compound Objective Function

The above type of analysis provides information that can form the basis for choosing a configuration to meet the requirements for aircraft E, with some allowance for modifications to payload or penetration range. A possibly more difficult decision to make is what degree of manoeuvre performance to provide. Aircraft reference F and G in Table 3 are optimised for particular turn-rate requirements but what is the best combination of STR and ATR? There are many views as to the relative importance of these parameters (*eg* see Fletcher and Burns[20]) but one measure of manoeuvrability that has been used is the product ($STR^2 \times ATR$). This has been incorporated in the compound objective function described in section II.

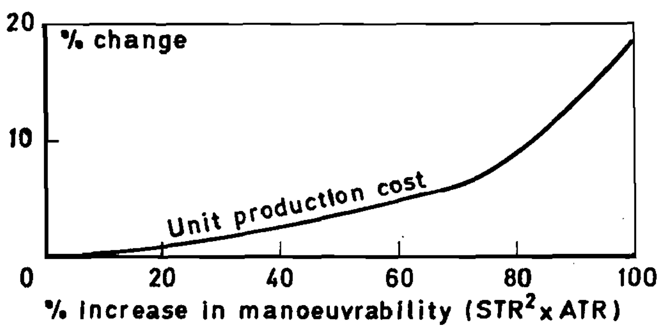
The set of requirements to define aircraft E were again used as a datum so that with the compound objective function the optimum configuration shown in Table 3 would be generated when the weighting constant on manoeuvrability was zero. Increasing the value of the weighting constant on manoeuvrability produced the change in unit production cost with manoeuvrability shown in Fig 9a. This line corresponds to the locus of configurations having the maximum manoeuvrability for a given cost or the minimum cost for a given manoeuvrability. Fig 9b shows the optimum combination of STR and ATR to produce a given level of manoeuvrability. The corresponding optimum values of some of the design variables are shown in Fig 9c. The initial increase in manoeuvrability is obtained for a very small increase in unit production cost by increasing aspect ratio to reduce the lift-dependent drag and hence increase STR. The cost increase associated with the heavier wing that results is partially offset by a small decrease in wing area and engine size. This trend continues until the constraint on ride quality (IC21) becomes active, when aspect ratio reaches a peak and wing area and engine size begin to increase. The increase in the latter two variables causes the constraint on take-off ground roll (IC23) to cease to be active. With the increase in wing area also,



c Optimum values of some of the design variables



b Optimum combinations of sustained and attained turn rates



a Optimum combination of unit production cost and manoeuvrability

FIGURE 9
USE OF MANOEUVRABILITY/UNIT PRODUCTION COST AS A COMPOUND OBJECTIVE FUNCTION

aspect ratio is decreased and leading-edge sweep is increased to maintain the ride quality. There follows a gradual increase in unit production cost with improved manoeuvrability until the constraint on wing-tailplane separation distance (IC1) becomes active. This leads in turn to the length of the intake diffuser becoming greater than the minimum necessary for a given size of engine (*ie* constraint IC16 becomes inactive). The extra weight associated with additional fuselage length necessitates a larger increase in wing area and engine size to improve the manoeuvrability, as can be seen from the kinks in the curves in Fig 9c and the steepening of the unit production cost curve. Fuel that was accommodated in the wing is now stored in the extra fuselage volume (*ie* IV7 goes from its upper to its lower bound). The constraint on SEP at 0.9 Mach number and 10 km altitude (IC25) ceases to be active in view of the large engine needed to provide an increase in the STR component of manoeuvrability.

Effect of Advances in Technology

The configuration reference E in Table 3 was used as a datum for some calculations of the cumulative effects of advances in four aerospace technologies. The improvements described below have been predicted to be achievable in the near term (mid to late 1980s). The configuration was reoptimised, with unit production cost as the objective function, as each aspect of advanced technology was incorporated.

Wing Design. An increase of 0.05 in the drag rise Mach number of the wing-body combination has been predicted. The maximum lift coefficient of the wing and the critical lift coefficient in the lift-dependent drag calculation were both increased by 10%.

Propulsion. Linear dimensions of the engine were reduced by 7.5%, apart from the reheat fuelling and burning lengths, and the basic mass of the engine was decreased by 17%. The thrust and fuel consumption were unchanged. Production cost was not increased as it was assumed that some aspects of the technical advance would be realised as improvements in manufacture.

Structural Materials. By using composite materials the following mass reductions on components of a fixed size have been predicted: fuselage 17%, wing 13.5% and empennage 21%. The cost per unit mass of the advanced structures was taken as 30% higher.

Weapon Design. The weapons were unchanged in size and have an installed drag 20% lower than current weapons. The mass was also unchanged but it is expected that the effectiveness for the same mass of device will be increased by a factor of the order of 2. It is assumed in the present calculations that this increased effectiveness will be essential for operation in an increasingly hostile environment and that the weapon load will not therefore be reduced to give the present level of effectiveness.

Some results for the optimum configurations are shown in Fig 10 as cumulative percentage changes relative to aircraft reference E (current technology). The active constraints on performance remain unchanged for this set of optimised configurations - take-off ground roll, IC23, and SEP at 0.9 Mach number and 10 km altitude, IC25. For this type of strike aircraft the improvements in wing design have only a small effect on UPC, but lead to substantial changes in wing area and aspect ratio. The higher maximum C_L available for take-off enables a smaller wing to be used, which alters

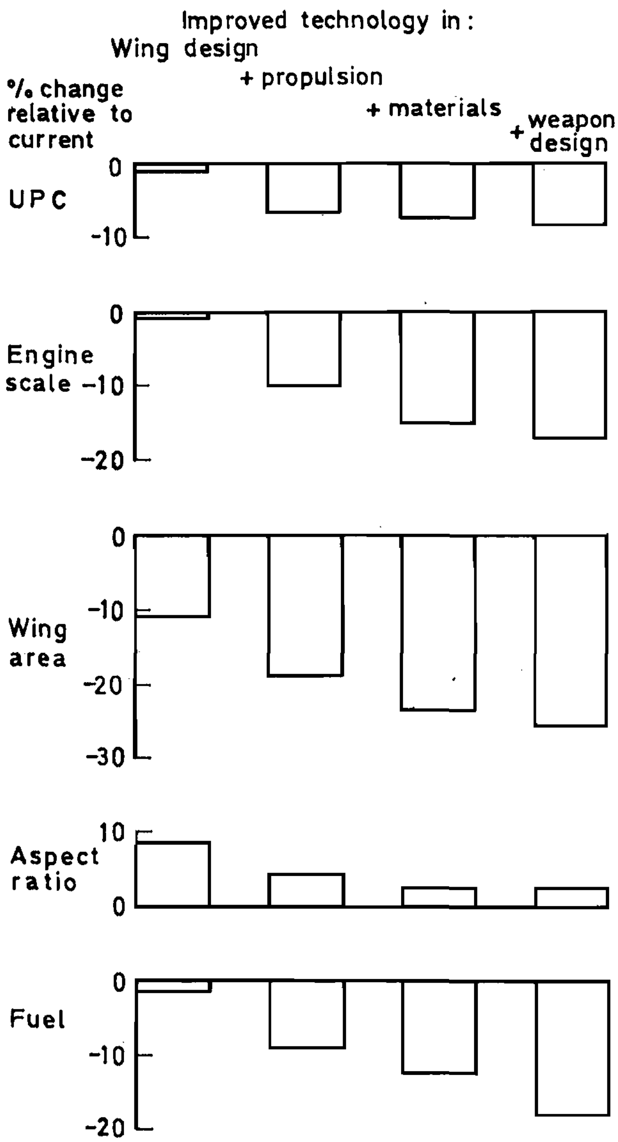


FIGURE 10
CUMULATIVE EFFECTS OF TECHNOLOGICAL ADVANCES

the balance between zero-lift drag and lift-dependent drag. Aspect ratio is increased to reduce the latter component and minimise fuel consumption on the cruise legs of the sortie. Improvements in propulsion have a major effect on the configuration because of the nature of the active constraints, both of which are strongly dependent on thrust to weight ratio. The smaller engine allows a 6% reduction in fuselage length. As the advanced materials produce a 1% decrease in UPC it is evident that the reduction in size of the aircraft is sufficient to offset the increased production cost for the new structures. Engine size is decreased by a further 5%. Similar effects occur when the improved weapon design is incorporated. For this type of requirement the major change resulting from the simultaneous application of the new technologies is a reduction of nearly 10% on linear dimensions; there are only minor changes in the wing planform and fuselage shape. This decrease in size should make the aircraft less easy to detect visually but it may aggravate the problems of aerodynamic interference between the aircraft and its weapons. The decrease in fuel consumption of 18% would be an important contribution to reducing aircraft life-cycle cost.

IV. Concluding Remarks

The applications of the MVO method for combat aircraft described above have shown the dependence of the optimum configuration on the details of the requirements for the sortie and point-performance parameters. The results from such studies cannot therefore be applied out of the context of the particular requirements to which they relate. Thus the relative benefits of the aspects of advanced technology considered above may be significantly different for an aircraft optimised for air combat.

In using the method there is a need for a continuing critical examination of the results in order to understand the combinations of events (*ie* changes in active constraints or variables going off or on to bounds), and the interaction of factors that produce a particular configuration. This implies a large degree of man-program interaction and an intimate knowledge of the method of design synthesis and analysis on the part of the user. If the program is used as a 'black box' to produce optimum configurations it is possible to reach misleading conclusions, *eg* the validity of an estimation method for mass or aerodynamic performance may be doubtful for some extreme combinations of independent variable values. Two important consequences arise; it is not possible to change rapidly the type of configuration synthesised, in view of the need to learn the characteristics of the new model before confidence may be placed in the results, and increasing the complexity of the mathematical model, assuming that this is not limited by computing power, makes the process of understanding the model behaviour and the interpretation of results more difficult.

As a result the use of the MVO method is currently confined to studies at the pre-feasibility stage of aircraft design, as a relatively simple aircraft synthesis is acceptable for this work. For the method to find more widespread use in the future will require an improvement in aids to understanding the changes in the optimum configuration (*eg* by the use of computer graphics), greater computing power to allow a thorough and rapid investigation of the factors determining an optimum design and its behaviour in subsidiary roles, and an extension of modular programming to allow a new mathematical model to be built from standard components (*eg* tailplane or canard) and analysis methods.

Before the MVO method is developed in this fashion there is clearly a need to improve the acceptability of mathematical modelling with optimisation as a general approach to solving problems in aircraft design. Currently this approach tends to be limited to specialist areas that involve complex mathematical analyses (*eg* structural design). It would be valuable if the processes for some aspects of aircraft design traditionally done on the drawing board could be codified and be used as the subjects for simple optimisation studies.

The existing MVO method for combat aircraft provides a design tool to produce consistent sets of fully optimised configurations that completely meet a specified set of requirements. The method is not subjective in that it has no prejudices for particular sets of values for the independent variables and can thus lead to an examination of configurations that otherwise might be neglected.

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