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PRELIMINARY SIZING METHOD FOR UNMANNED
AIRCRAFT USING MULTI-VARIATE OPTIMISATION

by, A. TURNBULL MSc.

SUMMARY

This report presents a computerised design synthesis and optimisation package for an unmanned aircraft (UMA) system. It has been developed by interfacing the design synthesis for an UMA with the multi-variate optimisation package RQPMIN.

The background and objectives of the research programme are initially outlined, with the design synthesis relationships then being described. The optimisation process is briefly explained, along with the program architecture and implementation. A User's Guide for the package is provided in the appendices together with an optimisation example.

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Notation

ABSVIS	absolute viscosity
ACCALT	acceleration phase altitude (m)
ACCFBM	acceleration phase fuel burn (kg)
ACCMF	acceleration phase final Mach number
ACCMI	acceleration phase initial Mach number
ACCTIM	acceleration phase time taken (sec)
AF	normal acceleration load factor
ALFA	angle of incidence (rads)
ALT	altitude (general) (m)
ALTAPP	altitude for attained turn point performance (m)
ALTF	climb phase final altitude (m)
ALTI	climb phase initial altitude (m)
ALTMAX	altitude for max. Mach number point performance
ALTSPP	altitude for sustained turn point performance (m)
APPMAS	apparent mass factor
AR	aspect ratio (gross wing)
ARN	aspect ratio (nett wing)
ATRF	max. attained turn 'g' load
ATRFPI	max. attained turn 'g' load in point performance
ATTROT	attained turn rate (deg/sec)
AVIDEN	avionics packaging density (kg/m^3)
AVILEN	avionics bay length (m)
AVIM	avionics mass (kg)
AVIVOL	avionics bay volume (m^3)
BYPASSR	bypass ratio
CCK1	lift induced drag coefficient
CD	drag coefficient
CDV	lift induced drag factor ($=CCK1 \times CL^2$)
CD0	zero lift drag coefficient of vehicle
CD0P	zero lift drag coefficient of parachute
CL	lift coefficient
CLAFB	lift curve slope of nose and forebody
CLANW	lift curve slope of nett wing
CLAS	lift curve slope of wing section
CLAWBC	lift curve slope of wing-body combination
CLBOT	buffet onset lift coefficient
CLDES	design lift coefficient
CLFBM	climb phase fuel burn (kg)
CLMACN	climb phase Mach number
CLMAXS	max. lift coefficient of wing section
CLMAXW	max. lift coefficient of wing
CLMAXWB	max. lift coefficient of wing-body combination
CLRAT	climb phase throttle setting
CLT	lift coefficient in manoeuvre phase turns
CLTIM	climb phase time taken (sec)
CMN	Mach number (used in range analysis)
COMPLEN	equipment compartment total length (m)
CRALT	cruise phase altitude (m)
CRFBM	cruise phase fuel burn (kg)
CRMAC	cruise phase Mach number
CRRAN	cruise phase range (nm.)
CRRAT	cruise phase throttle setting

CRTIM	cruise phase time taken (sec)
CV	velocity (general) (m/s)
CWCB	nett wing root chord (m)
CWCC	gross wing centre-line chord (m)
CWCT	tip chord (m)
CWMA	gross wing aerodynamic mean chord (m)
CWMG	gross wing geometric mean chord (m)
CWMN	nett wing mean chord (m)
C1	fuselage mass factor
DAALT	high speed dash phase altitude (m)
DAFBM	high speed dash phase fuel burn (kg)
DAMAC	high speed dash phase Mach number
DATIM	high speed dash phase time taken (sec)
DELFBM	fuel burn over a general interval (kg) (climb or acceleration phases)
DELM	small interval of Mach number (acceleration phase)
DELTA	relative pressure
DELTAH	small interval in altitude (climb phase)
DELTAY	leading edge sharpness parameter
DELTIM	time taken over a general interval (sec)
DENS	density (local)
DM	vehicle mass (kg)
DRAGTOT	total vehicle drag
DSLST	sea-level static thrust (N) (used in engine performance module)
EACCFBM	estimate of acceleration fuel burn (kg)
ECDS	engine zero lift drag contribution
ECLR	engine clearance (m)
ECLRM1	minimum engine clearance (m)
ECLRM2	maximum engine clearance (m)
ECLFBM	estimate of climb fuel burn (kg)
ECRFBM	estimate of cruise fuel burn (kg)
EDAFBM	estimate of high speed dash fuel burn (kg)
EFLOW	air flow (kg/s)
EFLOWP	air flow (lb/s)
EIM	engine and installation mass (kg)
ELECDEN	electrics density (kg/m^3)
ELECLEN	electrics bay length (m)
ELECVOL	electrics bay volume (m^3)
EM	engine mass (kg)
EMANFBM	estimate of manoeuvre phase fuel burn (kg)
ENGD	engine diameter (m)
ENGDR	reference engine diameter (m)
ENGLEN	engine length (m)
ENGLR	reference engine length (m)
ENGMAX	engine diameter and clearance (m)
ENGVOL	engine volume (m^3)
ERETFBM	estimate of return fuel burn (kg)
ESM	electric system mass (kg)
ETHRG	gross engine thrust (N)
ETHRKG	nett engine thrust (kg)
ETHRN	nett engine thrust (N)
ETHRP	nett engine thrust (lbs)
ETOTFBM	estimate of total fuel used (kg)

FB	fuselage width (m)
FBM	general fuel burned (kg)
FBWK	interference factor (effect of the wing on the body in calculating the lift curve slope)
FCDS	fuselage zero lift drag contribution
FCLEN	nose cone length (from geometrical diameter) (m)
FCLENR	nose cone length (from maximum required diameter)
FCM	flying controls mass (kg)
FCVOL	nose cone volume (geometry)
FCVOLR	nose cone volume (required)
FECLR	factor on engine diameter for clearance
FEDK	factor for calculating engine diameter
FELK	factor for calculating engine length
FFAT	fraction of fuel for attained turn point performance
FFLOW	fuel flow (kg/s)
FFLOWA	average fuel flow (kg/s)
FFLOWP	fuel flow (lb/s)
FFMN	fraction of fuel for max. Mach number point performance
FFSEP	fraction of fuel for specific excess power point performance
FFST	fraction of fuel for sustained turn point performance
FH	fuselage height (m)
FINRAT	fineness ratio
FL	fuselage length (m)
FLAGENG	engine installation flag
FLAGFUS	fuselage cross section flag
FLAGPAY	payload flag
FLAGPP	point performance flag
FLAGRS	recovery system flag
FM	fuselage mass (kg)
FSA	fuselage surface area (m^2)
FSM	fuel system mass (kg)
FTCG	fuel tank CG position (measured from nose) (m)
FTDEN	fuel density (kg/m^3)
FTLEN	fuel tank length (m)
FTVOL	fuel tank volume (m^3)
FUELMAS	fuel mass that can be carried by fuselage (kg)
FUELVOL	fuel volume that can be accommodated in fuselage
FUSVOL	fuselage volume (available from geometry)
FUSVOLR	fuselage volume (required from sortie and mass analysis)
FWBK	interference factor (effect of the body on the wing in calculating the lift curve slope)
FXSA	fuselage cross sectional area
GOFI	factor for nose cone geometry
GNAPP	load factor (attained turn point performance)
GNS	load factor (sustained turn manoeuvre phase)
GNSPP	load factor (sustained turn point performance)
IBUFF	buffet onset C_L output flag
LOALT	loiter leg altitude (m)
LOFBM	loiter leg fuel burned (kg)

LOMAC	loiter leg Mach number
LOTIM	loiter leg time taken (min)
MANALT	manoeuvre leg altitude (m)
MANFBM	manoeuvre leg fuel burned (kg)
MANMAC	manoeuvre leg Mach number
MANTIM	manoeuvre leg time taken (sec)
MAR	effect of aspect ratio on design Mach number
MASSA	mass of vehicle at attained turn point performance
MASSM	mass of vehicle at max. Mach number point performance
MASSP	mass of vehicle at specific excess power point performance
MASSS	mass of vehicle at sustained turn point performance
MAXNUM	max. achievable Mach number
MAXSPD	max. achievable speed (m/s)
MDES	design Mach number
MD2	drag divergent Mach number
MSWEEP	effect of sweep angle on design Mach number
NDEG	number of degrees in manoeuvre phase turns
NTURN	number of turns in manoeuvre phase
PAYDEN	payload density (kg m^{3})
PAY1	first payload mass (kg)
PAY2	second payload mass (kg)
PAYLEN1	first payload bay length (m)
PAYLEN2	second payload bay length (m)
PAYVOL1	first payload bay volume (m^{3})
PAYVOL2	second payload bay volume (m^{3})
PRESS	local pressure
QWLD	leading edge sweep angle (deg)
QWLR	leading edge sweep angle (rad)
QWTD	trailing edge sweep angle (deg)
QWTR	trailing edge sweep angle (rad)
QW2D	mid chord sweep angle (deg)
QW2R	mid chord sweep angle (rad)
QW4D	1/4 chord sweep angle (deg)
QW4R	1/4 chord sweep angle (rad)
RAN	range (km)
RATAPP	throttle setting: attained turn point performance
RATING	throttle setting (general)
RATSPP	throttle setting: sustained turn point performance
RCDK	technology factor (induced drag)
REF	Reynolds number
REQMM	required max. Mach number for point performance
REQSEP	required specific excess power for point performance
RESFBM	fuel reserves (kg)
RESFF	fuel reserve as fraction of total fuel
RESLEN	reserve fuel tank length (m)
RESVOL	reserve fuel tank volume (m^{3})
RETALT	return phase altitude (m)
RETFBM	return phase fuel burn (kg)
RETMAC	return phase Mach number
RF	fuselage shape factor (zero lift drag coefficient)

RM	recovery mass (kg)
RN	engine installation factor (zero lift drag coeff.)
ROT	rate of turn (deg/sec)
RRE	Reynolds number correction factor (zero lift drag coefficient)
RSPA	recovery system canopy area (m^2)
RSPM	recovery system canopy mass (kg)
RSPSD	recovery system packing density (kg/m^3)
RSPSL	recovery system compartment length (m)
RSPSV	recovery system volume (m^3)
RSTIM	recovery system total installation mass (kg)
RT	tail unit factor (zero lift drag coeff.)
RTHR	thrust reverser factor (zero lift drag coeff.)
RUC	undercarriage factor (zero lift drag coeff.)
RW	wing factor (zero lift drag coefficient)
SECAPP	Mach number: attained turn point performance
SECMAC	Mach number (general)
SECSEP	MAch number: specific excess power point performance
SECSPP	MAch number: sustained turn point performance
SEP	specific excess power
SFC	specific fuel consumption
SIGMA	relative density
SKINT	skin thickness (mm)
SLSTR	sea-level static thrust ratio
STHR	specific thrust
STRF	max. sustained turn 'g' load
STRFP1	max. sustained turn 'g' load in point performance
STU	tail unit area
SUSROT	sustained turn rate
TARM	tail arm (m)
TC	thickness-chord ratio
TCLEN	tail cone length (from geometry)
TCLENR	tail cone length (from analysis)
TCVOL	tail cone volume (from geometry)
TCVOLR	tail cone volume (from analysis)
TEANG	trailing edge included angle
TEMP	temperature
TFMEC1	technology factor (fuselage mass)
TFMEC2	technology factor (wing mass)
THETA	relative temperature
THRR	reference engine sea-level static thrust (N)
TOTFBM	total fuel burned (kg)
TOTHR	sea-level static thrust of required engine (N)
TOTMAS	total vehicle mass (kg)
TRC	taper ratio of gross wing
TUM	tail unit mass (kg)
UNCN	taper ratio of nett wing
VDEAS	design diving speed (m/s)
VMAXI	max. speed obtainable in 1'g' flight
VMAXII	max. speed obtainable in 1'g' flight (improved value)
VRD	descent speed of vehicle in recovery (m/s)
VSOUND	speed of sound (m/s)

WA	wing area (gross) (m^2)
WAN	wing area (nett) (m^2)
WCDS	wing zero lift drag contribution
WM	wing mass (kg)
WS	wing span (gross) (m)
WSN	wing span (nett) (m)
XCGMOM	vehicle CG position (measured from nose) (m)
XKINVIS	kinematic viscosity

1. Introduction

A one year research programme, as part of a PhD study, was started in October 1987. It had as its main objectives the development of a computerised design synthesis for an unmanned air vehicle (UMA), and the integration of this synthesis with existing RAE multi-variate optimisation (MVO) algorithms.

MVO has been under investigation at RAE Farnborough since before 1970, and started with a design synthesis for transport aircraft (Ref. 11). By 1980, (Ref.5) a synthesis and optimisation program had been developed for a high-wing, aft-tail, single engined combat aircraft and a twin-engined version was produced a year later. Cannard-delta layouts are currently under investigation (Ref. 3). With the increasing interest in UMA's a need for the development of a design synthesis for such vehicles emerged. The synthesis was not to be aimed at the design of one particular type of UMA but was to be a method suitable for any configuration, within laid down limits (see Baseline Configuration).

The method was to use simple mass, aerodynamic, packaging and performance estimations for the vehicle. The vehicle was then assumed to fly a specified sortie with the fuel consumption and performance estimations being calculated for each leg of the given mission. The mass of the synthesized aircraft was to be optimised with respect to specified configuration variables, such as wing area, aspect ratio, fuselage length and diameter, and sea-level static thrust. The optimisation problem was subject to a set of user-specified constraints, on overall geometry and performance for example, and was to be solved by manipulating the configuration variables within simple upper and lower limits. The final results should give the optimised mass, and the total configuration of the vehicle at the start and finish of the optimisation process.

Some of the initial background work, simple mass and drag estimations, was started during a preliminary peroid during a Msc course, but the majority of the work was completed for the PhD programme.

2. Baseline Configuration

The research programme started with a parametric study in order to investigate the geometric sizes, weights, performance, aerodynamics and equipment packaging requirements for current UMA's. This information led to the initial design of two baseline configurations, shown in Fig. 1, and was also used to validate the mass, aerodynamic and performance estimations.

The following requirements were laid down at the beginning of the programme for the initial layout:

1) flight speeds of up to high subsonics (Mach 0.8)

2) modest sustained turn load factors (≈2-3 'g')

3) consideration to be given to payloads of instrumentation

4) either air- or ground-launched

5) jet engine propulsion

6) wing planforms limited to trapezoidal configurations

The baseline vehicle under consideration consists of a trapezoidal wing planform, with similar shaped aft tail and a single fin situated on the upper surface of the fuselage afterbody. The fuselage itself is one of simple shape, with a cylindrical central body and defined nose and tail cones.

The fuselage internal layout (fig. 1) consists of nose cone; payload bay; avionics compartment; fuel tank; electrics bay; recovery systems and tail cone. The propulsion system is either internal, located behind the electrics bay, or externally podded below the fuel tank.

3. Summary of Design Relationships

A flow chart of the design synthesis can be seen in fig 2 which shows the structure of the program and where the following routines are included.

3.1 Geometric Model

The synthesis starts by sizing all the geometry-dependent variables of the vehicle configuration under consideration. These are items such as:

wing geometry: (both gross and net)
- wing span
- centreline chord
- geometric mean chord
- aerodynamic mean chord
- tip chord
- leading edge sweep
- mid-chord sweep
- trailing edge sweep

fuselage geometry:
- surface area
- body volume
- cross-sectional areas
- skin thickness

These geometric dependent variables are all calculated from the set of user-specified input data (external variables).

The engine geometry is scaled from a set of datum dimensions for a currently used turbojet engine, MICROTURBO TRS-18 and is based on the maximum static, sea-level thrust. Engine installation clearances are also calculated, being a factor of the engine diameter.

The relationships used for this geometric model can be seen in Appendix A.

3.2 Mass Estimation

The synthesis then makes use of these goemetric variables to estimate the mass of the empty vehicle. Various methods of estimating the mass of the airframe and components were investigated, with the degree of accuracy depending upon the complexity of the method used. Thus methods were chosen which depended mainly upon the simplicity of the input data required, and reasonable accuracy.

The difficulty in gaining a reasonable mass prediction for a vehicle of this sort came in obtaining information regarding the masses of the vehicle components. The methods used were mainly empirically based studies of manned civil and military aircraft. This led to difficulties when predicting the airframe and propulsion system masses, since the airframe estimated mass appeared to be well below the actual mass, with the propulsion estimation well above. This resulted in a new propulsion estimation method being produced (Appendix B) and technology factors being introduced into the airframe estimation methods.

The overall mass estimation module can be divided into three sub-sections of analysis, these are: structural, propulsion and systems.

3.2.1 Structures.

The structural group is composed of the wing, fuselage and tail unit. The wing mass estimation is a complex equation which takes into account the wing geometry and thickness, the design mass of the vehicle, normal acceleration factor and design diving speed. The fuselage estimation gives reasonable results for the simple circular cross sectional body under consideration but would need to be improved upon for more complex shapes. The estimation takes into account the skin and stringer masses and the mass of the frames used, these are defined by the major fuselage geometry. The tail unit estimation is very crude, with the mass being determined in terms of the design diving speed and tail unit area. This area is assumed to be one third the size of the wing area. A formula such as the wing estimation would improve the estimation but this would increase the amount of general input data considerably.

3.2.2 Propulsion.

The propulsion system mass has been estimated from a plot of engine static sea-level thrust against engine mass containing data for twenty current turbojet engines. The plot can be conveniently split into two thrust regimes, above and below 1.56 kN, and straight lines were fitted to the data in each. These lines were joined by a cubic spline in order to avoid discontinuities in the first and second derivatives. An installation factor of 1.3 is included in the engine mass estimation which allows for the mass of a simple intake system, although this could be an area for further detailed work. Overall the results are acceptable, with average errors of about 10%.

3.2.3 Systems.

The following systems were investigated: fuel, flying controls, electrical and recovery systems. The formulae used were, again, simple empirical calculations, with all, except the recovery system, being a factorisation of the aircraft design mass. The recovery system mass has been found by using the basic drag coefficient equation (Appendix B) leading to the canopy size, from which the canopy mass is calculated and then the total installed mass.

3.3 Packaging Calculations

With the vehicle mass predicted, volume accounting calculations can be carried out. This calculates the volumes and lengths of the compartments required for the given components and systems. These values are used in the problem constraints to confirm that the estimated fuselage volume (geometrical) is greater than the volume required to accommodate all the systems, components and fuel tank to be installed in the fuselage. The calculations are based on the previous mass predictions and density figures obtained from current UMA information and Ref 3. The relationships used here can be found in Appendix C.

The longitudinal CG of the vehicle is also calculated with the moment arms of the components being determined by assuming that the fixed masses of each are installed at specific points along the fuselage centre-line.

3.4 Aerodynamics

The next stage of the design synthesis was to calculate basic aerodynamic coefficients, such as: zero lift drag coefficient, lift induced drag coefficient; lift-curve slope, design lift and buffet onset lift coefficients. These coefficients are used in the sortie analysis and can be calculated for the configuration at each stage of the sortie. The relationships for these can be seen in Appendix D.

3.4.1 Drag Estimation

This consists of calculating the zero lift drag coefficient and the lift induced contributions of various components of the vehicle under consideration. The zero lift drag estimation used was based on the method in Ref. 5. This is a very elementary approach which sums the individual drag contributions of various components of the vehicle. A correction factor for the effect of Reynolds number on skin friction drag has been included, along with a factor to take into account the compressibility effects on the drag at high subsonic Mach numbers.

The lift induced drag coefficient was taken from Ref 5., this assumes a parabolic variation between the drag and lift coefficients and uses major geometric properties ie 1/4 chord sweep angle, taper ratio and aspect ratio, in the calculations.

3.4.2 Lift Estimation

The lift-curve slope for the vehicle is calculated using Ref 7. This method initially estimates the lift-curve slope of the wing and body separately. These are then used to determine the lift-curve slope of the wing-body combination by considering associated interference effects. These interference effects are estimated using the configuration geometry (fuselage diameter and wing span).

For the wing lift-curve slope to be calculated the trailing edge included angle of the wing section is required. As this is dependent upon wing section and the thickness chord ratio, one family of wing sections has been chosen and the variation of trailing edge angle with thickness chord ratio formulated for this.

The lift-curve slope calculations were dependent on various parameters, such as Mach number, leading edge sweep angle, ratio of exposed to gross wing areas, nett wing aspect ratio, mid-chord sweep angle, fuselage diameter and length. The latter having all been calculated in the geometry module of the synthesis.

3.4.3 Design and Buffet Onset Lift Coefficients

These coefficients are calculated using Ref 9 and are used at a later stage of the synthesis when the mission profile is investigated and also the buffet onset lift coefficient is used as a constraint, (see Appendix I). Both these coefficients are calculated using major geometric properties obtained from the geometry module.

3.5 Engine Performance Module

The engine performance module is based on the WILLIAMS INTERNATIONAL FJ44 turbofan engine. For a given altitude, throttle setting and Mach number the nett thrust, airflow and fuel flow for the FJ44 engine can be obtained. These values are then scaled using the sea-level thrust ratio of the required engine to the reference engine. The details of this module can be found in Appendix F. The scaled values are used during the analysis of the mission profile.

4. Basic Mission Profile

The lift, drag and engine performance methods are used in the vehicle performance estimation. This consists of a sortie analysis in which the synthesized vehicle is assumed to fly a specified mission profile, with its performance and fuel consumption being estimated for each sortie leg. This is followed by point performance analysis which includes the calculation of the maximum achievable sustained and attained turn 'g' load factors; maximum Mach number obtainable in 1'g' flight and specific excess power.

The mass of the vehicle prior to the sortie analysis includes the fuel mass which is assumed to be that obtained from the previous sortie analysis. Estimates of the fuel required for each leg are initially given as external variables and are totalled to give the fuel mass prior to the first sortie analysis. This fuel mass is then modified, for the current airframe, by iterations of the sortie analysis.

A basic mission profile was proposed to exercise the program, as follows:

1. Climb (at full throttle to 3000m)

2. Cruise (from end of climb phase at best cruise speed for 50 nautical miles at 3000m)

3. Loiter (for 30 minutes at 3000m)

4. Accelerate to Mach 0.8

5. 5 minute highspeed dash at Mach 0.8

6. Manoeuvre phase (two 360 degree 4 'g' turns)

7. Return with 10% reserve fuel

Initially the payload was to be 25 kg with the option of increasing the mass to 50 kg to test the optimisation. The vehicle configuration to be initially considered was one with a circular cross-sectioned body, one payload bay, internal engine and a hand-packed recovery system.

5. Optimisation and Implementation

The mass of the synthesized vehicle was optimised using the optimisation package RQPMIN (Ref 10). This package is designed to solve any constrained or unconstrained problem. This is a problem of minimising a function with respect to certain problem variables, while having to satisfy user-specified constraints as well as simple upper and lower bounds.

The design synthesis and optimisation is automatically implemented by a modular computer program, written in standard FORTRAN 77. The modular program consists of the RQPMIN optimisation package with the synthesis being incorporated as the last subroutine in the program. The whole program has a total length of 8500 lines and consists of 108 modules.

To run the program the user must provide the last subroutine and four certain input data files. These files are described fully in Appendix J. The subroutine contains the equations which make up the problem of interest, ie. the design synthesis, with the function to be minimised and the constraint equations, which can be either equality or inequality constraints. The main input file contains the starting values of the independent variables, their upper and lower bounds and markers to state which function is to be optimised and which are the constraints. A second input data file contains the values of the external variables, ie. payload mass, sortie leg definitions, design diving speed and package densities. The engine and aerodynamic performance data are included in the final two files which are read at the beginning of the program. These contain equation coefficients for engine performance charts and the aerodynamic calculations.

6. Results

The program has been exercised using all the variables and constraints, described in Appendix I, in order to try to obtain convergence of the configuration. It was found that convergence could be achieved when using the following set of constraints:

1. fuel required
2. fuselage diameter
3. sustained turn load factor
4. wing geometry

These constraints are all described fully in Appendix I.

It was felt that a constraint by which RQPMIN could control the fuel carried, via independent variables, was required. This constraint is function number 2 and states that the fuel which can be accommodated in the fuselage should be equal to that required by the sortie analysis.

When the length constraint was included in the set of applied constraints convergence was never achieved. The reason for this is discussed fully later but seems to be that the length constraint uses the minimum fuselage daimeter to calculate the required length and this would only work satisfactorily when the actual fuselage diameter is near the minimum.

The volume constraint was not included as convergence was never reached when using this together with the fuselage length and diameter constraints. However, with the diameter being fixed by the engine size, the fuselage length constraint is in effect a volume constraint and as such a separate volume constraint is not required.

The volume constraint has been applied without the fuselage length constraint, but satisfaction of the volume and fuselage diameter constraints, does not necessarily mean that the fuselage length will be adequate to fit all the compartments. This is because the volume constraint takes no account of the nose and tail volumes. However this constraint could be used if the user required only the central fuselage length to be satisfied, for example, assuming no equipment is carried in the nose and tail cones.

The buffet onset lift coefficient constraint, as with the other constraints, can be switched on and off as the user wishes. This constraint was not exercised extensively but it seemed that the coefficient calculated from geometry was rather low, this is proving difficult to satisfy. The region of buffet onset is very difficult to predict, this being a region for further detailed work.

The turn load factor constraints seemed reasonably easy to satisfy, with high maximum 'g' loads being predicted.

7. Discussion

The program has been exercised using a range of constraints and variables (Appendix I). It has been found however that constriants 2, 3 and 4 are effectively trying to achieve similar results. This being the case, using these constraints together increases computer time and could lead to unsatisfactory results.

Constraints 3 and 4 (volume and length) seem only to work sensibly when the actual fuselage diameter is close to the minimum ENGMAX (defined by the engine diameter). The reason for this is that if the actual fuselage diameter is greater than ENGMAX then the length of the various components will become shorter and thus the overall fuselage length will drop below the minimum specified by the length constraint.

Convergence has been achieved using the fuel constraint (2) without either of the length or volume constraints. This seems satisfactory as the fuel required tends to drive the overall optimisation as wished.

The results from the runs show different final points are reached depending upon the starting values. An example of this is when the wing is started with a low aspect ratio. This variable will be decreased or increased, along with the others, until the program reaches an end point but will always stay reasonably close to the starting value, similarly with a high aspect ratio wing. The variables which always seem to be altering the most are the wing area and the sea-level static thrust. These will invariably be reduced to their lowest bound, independent of the starting value, whereas other independent variables will only be changed slightly. This is probably the result of there being many local optimum points occurring in 'hollows' within the surface described by the design relationships and differing starting values are leading to different optimum points.

The program can end, completed, in 4 ways:

(i) converged A
(ii) converged B
(iii) converged C
(iv) no further progress possible

(These are discussed in Ref 10)

When the program ends with the message 'no further progress possible' the applied constraints are not always totally satisfied by the final solution. But with the 3 convergence cases the constraints are satisfied. The user should note, however, that reference 10 states that:

> "none of the three converged messages guarantees that RQPMIN has found a valid solution. Nor does the 'no further progress possible' message necessarily mean that the program has failed."

So it is up to the user to decide whether the solution given is valid.

An example of this is that a constraint could, in practical terms, be satisfied to the user, but RQPMIN will try to find a more accurate solution and fail.

The program requires the constraints to be satisfied to a high degree of accuracy and this could be the probable cause of the none convergence in many runs, as the program seems to stop after a certain number of feasibility or minimisation steps without convergence.

8. Conclusion

The design synthesis for an UMA has been successfully linked with the optimiser RQPMIN. Convergence has been achieved with a small number of constraints, however much more exercising of the program is need to try all the constraints, in various combinations. It is also possible to hold independent variables constant throughout the process, this could be tried in order to obtain convergence .

One optimised solution could not be found for a single set of external variables. This depended upon the starting values of the independent variables. It is for the user to decide whether the solution found is viable, the total configuration details are given in the output.

Further work which can be foreseen is: extending the present synthesis to square-bodied vehicles, looking into the aerodynamics of these and the internal stores and packaging layouts; better aerodynamic and mass estimations included in the synthesis; and investigation into more radically alternative configurations such as a flying wing type vehicle (Aquila) with internal wing stores or large thickness/chord winged vehicles again with internal stores.

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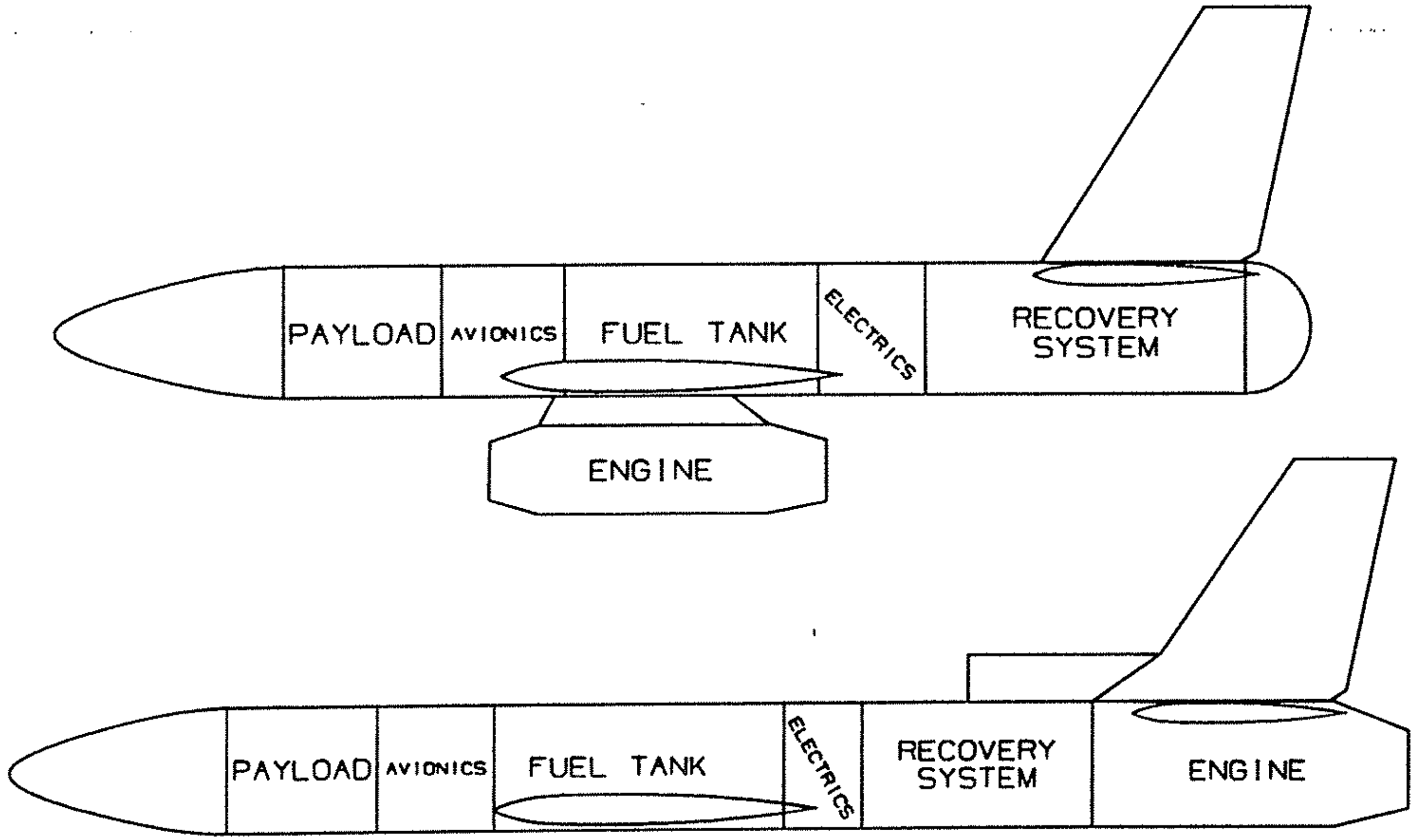
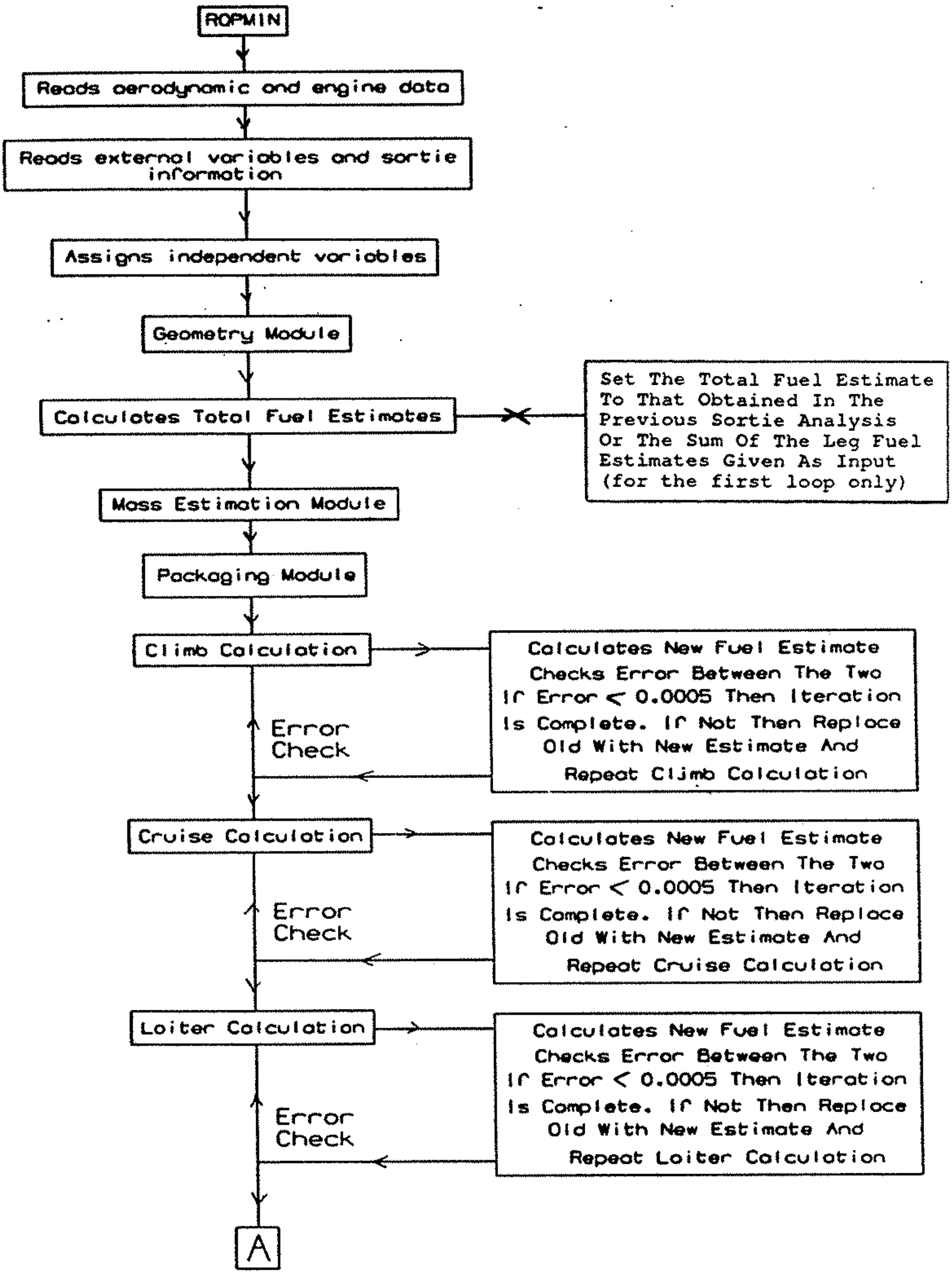
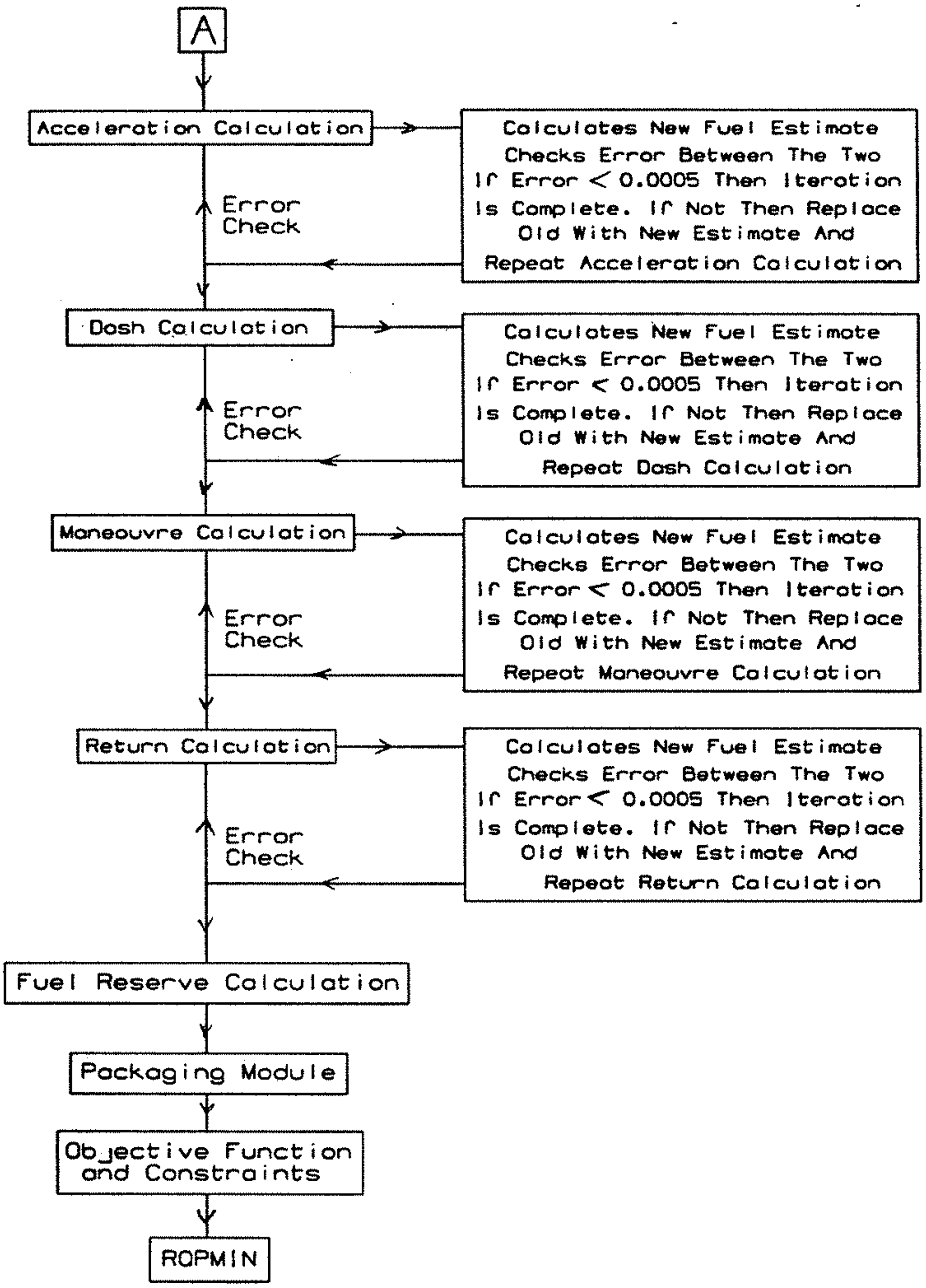


Figure 1. Baseline Configurations

Figure 2 Flow Chart





A
Acceleration Calculation
Calculates New Fuel Estimate
Checks Error Between The Two
If Error < 0.0005 Then Iteration
Is Complete. If Not Then Replace
Old With New Estimate And
Repeat Acceleration Calculation
Error Check
Dash Calculation
Calculates New Fuel Estimate
Checks Error Between The Two
If Error < 0.0005 Then Iteration
Is Complete. If Not Then Replace
Old With New Estimate And
Repeat Dash Calculation
Error Check
Manoeuvre Calculation
Calculates New Fuel Estimate
Checks Error Between The Two
If Error < 0.0005 Then Iteration
Is Complete. If Not Then Replace
Old With New Estimate And
Repeat Manoeuvre Calculation
Error Check
Return Calculation
Calculates New Fuel Estimate
Checks Error Between The Two
If Error < 0.0005 Then Iteration
Is Complete. If Not Then Replace
Old With New Estimate And
Repeat Return Calculation
Error Check
Fuel Reserve Calculation
Packaging Module
Objective Function and Constraints
ROPMIN

APPENDIX A.

GEOMETRY MODULE

1. Wing Geometry

The wing planform shown in fig. A.1.1. is defined by independent and external and external variables. Using the notation in Fig. A.1.1. the following dependent variables are obtained:

Gross wing

span, $WS = (AR \times WA)^{1/2}$

centre-line chord, $CWCC = \dfrac{2 \times WA}{WS \times (1+TRC)}$

geometric mean chord, $CWMG = 0.5 \times CWCC \times (1+TRC)$

aerodynamic mean chord, $CWMA = \dfrac{2}{3} \times CWCC \times \left(TRC + \dfrac{1}{(1+TRC)}\right)$

tip chord, $CWCT = TRC \times CWCC$

LE sweep, $QWLR = \tan^{-1}\left(\tan(QW4R) + \dfrac{CWCC}{2 \times WS} \times (1-TRC)\right)$

mid chord sweep, $QW2R = \tan^{-1}\left(\tan(QW4R) - \dfrac{CWCC}{2 \times WS} \times (1-TRC)\right)$

Nett wing

span, $WSN = WS - FB$

root chord, $CWCB = CWCC \times \left(1- \dfrac{FB}{WS} \times (1-TRC)\right)$

taper ratio, $UNCN = \dfrac{CWCT}{CWCB}$

mean chord, $CWMN = 0.5 \times (CWCB + CWCT)$

area, $WAN = CWMN \times WSN$

aspect ratio, $ARN = \dfrac{WSN}{CWMN}$

Also trailing edge sweep,

$$QWTR = \tan^{-1}\left(\tan(QWLR) - 2 \times \frac{(CWCC-CWCT)}{WS}\right)$$

If QWTR is negative then sweep is forward
positive then sweep is backward

2. Fuselage Geometry

2.1 The geometrical size of the fuselage is obtained, again, from independent and external variables and can be seen in fig A.2.1:

surface area, $$FSA = 2.56 \times FL \times \frac{(FH+FB)}{2}$$

cross sectional areas,

1/ circular; $$FXSA = \pi \times \left(\frac{FH+FB}{4} \right)^2$$

2/ elliptic; $$FXSA = \pi \times \frac{(FH+FB)}{4}$$

3/ rectangular; $$FXSA = FH \times FB$$

2.2 The nose cone is defined from the diameter of the fuselage and assumes a linear variation of cross-sectional area with axial distance. The value of the constant GOFI has been obtained by calculating the average value for a series of current UMAs ie ASAT, Mirach.

length, $$FCLEN = \frac{\pi}{GOFI} \times \left(\frac{FB+FH}{2} \right)^2$$

volume, $$FCVOL = \frac{GOFI \times FCLEN^2}{2}$$

2.3 In order to simplify the calculations the tail unit length for both internal and external engine is assumed to be half the fuselage diameter in size with the volume to be the same as that of a hemisphere.

length, $$TCLEN = \frac{FB+FH}{4}$$

volume, $$TCVOL = \frac{2}{3} \times \pi \times (TCLEN)^3$$

Geometric fuselage volume:

$$FUSVOL = (FL-FCLEN-TCLEN) \times FXSA + TCVOL + FCVOL$$

This assumes a constant cross sectional centre body.

3. Engine geometry

The engine dimensions are also calculated in the geometry module. Both the engine diameter and engine length are taken as scaled values of a currently used turbojet engine, the MICROTURBO TRS-18. The reference engine dimensions are scaled by a factor of the sea-level thrust ratio.

3.1 Diameter:

$$ENGD = ENGDR \times \left(\frac{TOTHR}{THRR}\right)^{FEDK} \qquad (m)$$

Where FEDK is a constant and has been obtained by averaging the value calculate using a data bank of current turbojets. The value obtained is 0.1269885.

3.2 Length:

$$ENGL = ENGLR \times \left(\frac{TOTHR}{THRR}\right)^{FELK} \qquad (m)$$

Where FELK is a constant and has been obtained in a similar way to that of FEDK.

3.3 Engine clearance.

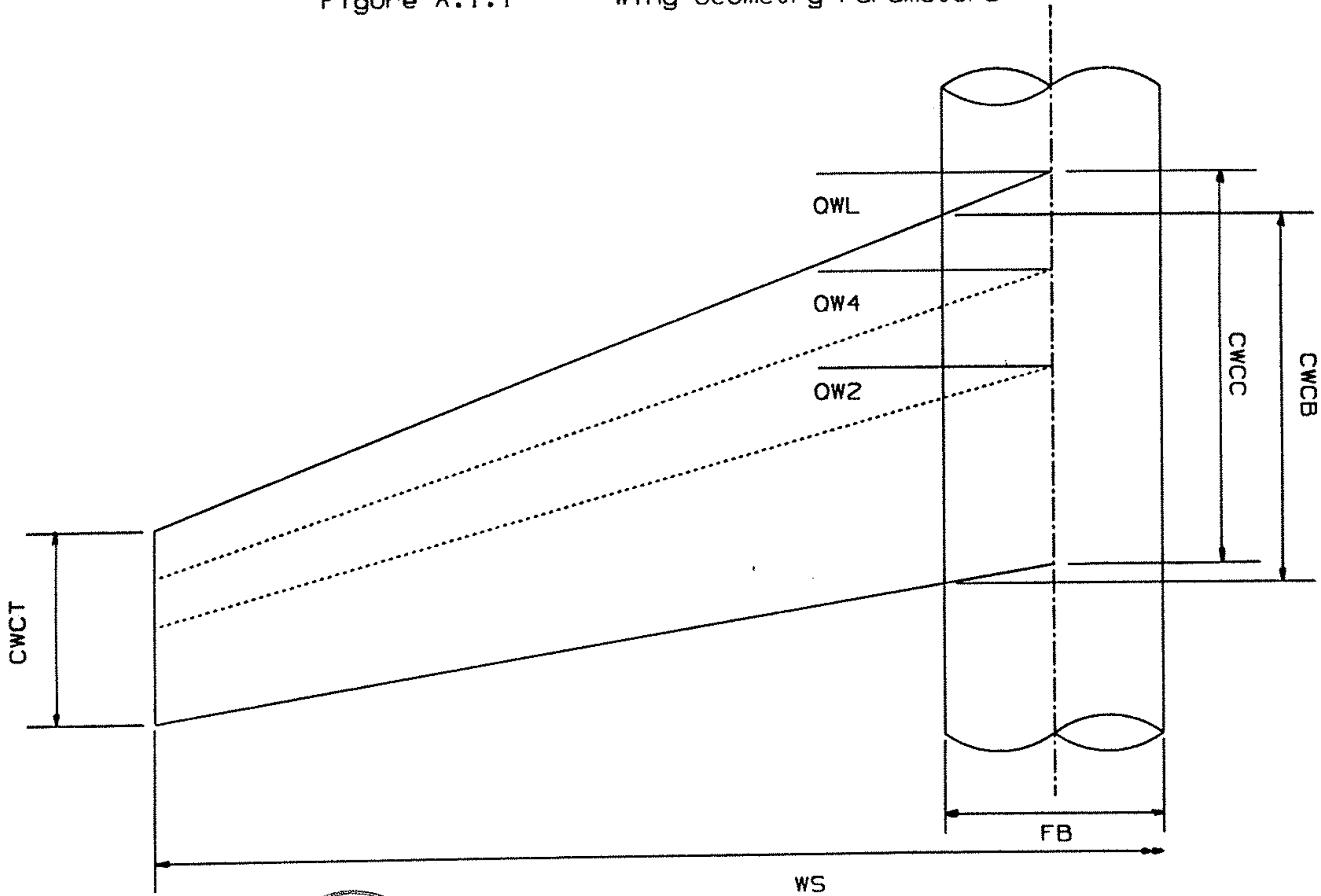
The engine clearance is calculated as a fraction of the engine diameter.

$$ECLR = ENGD \times FECLR \qquad (m)$$

where FECLR = fraction of engine diameter required and is an external variable.

The clearance size is limited by upper and lower bounds, which are also given as external variables; ECLRM2 and ECLRM1 respectively.

Figure A.1.1 Wing Geometry Parameters



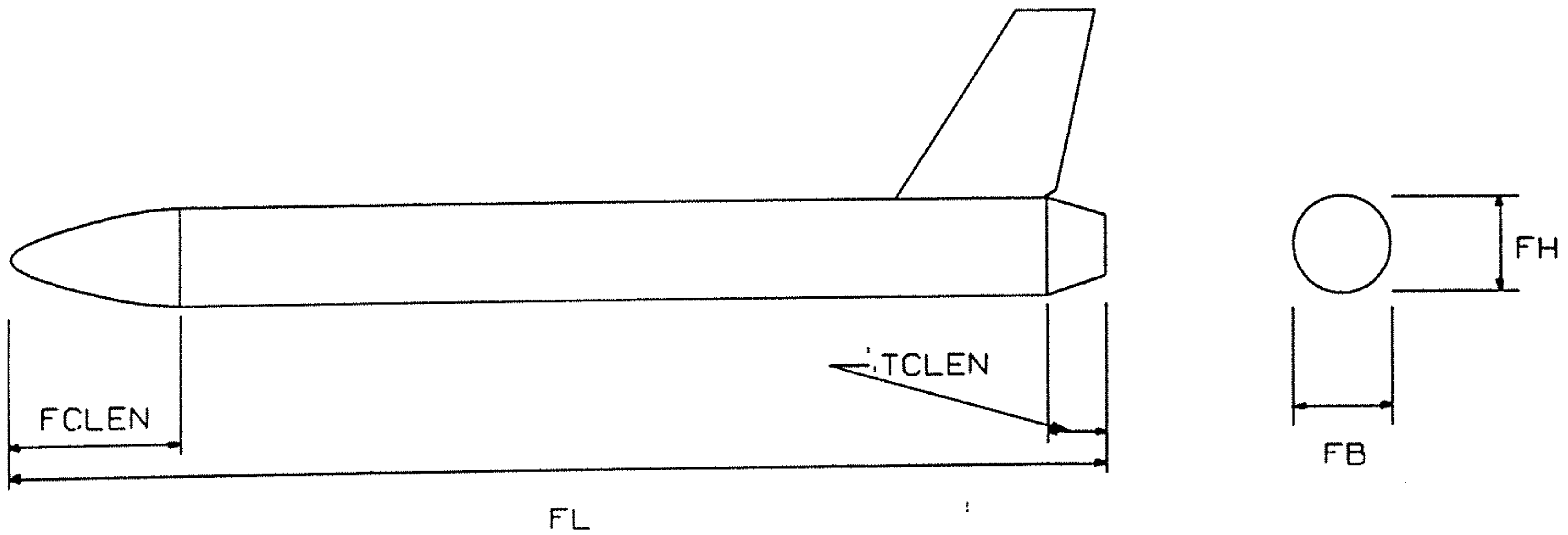


Figure A.2.1 Fuselage Geometry

APPENDIX B.

Mass Estimation Module.

The formulae used in the mass estimation module have been chosen on the basis of two requirements: minimum of input data and good accuracy. The summary of formulae, which follows, is divided into structural, systems and propulsion groups.

1. Structure Group.

This group is composed of the wing, fuselage and tail unit. All the following formulae are empirical and are taken from Ref. 1.

1.1 Wing Mass.

As can be seen from the formula the important factors in the wing mass estimation are wing geometry and thickness, the design mass of the aircraft, the normal acceleration factor and the design diving speed.

$$WM = C1 \times \left[\left(\frac{WS \times WA}{\cos(QW4R)} \right) \times \left(\frac{1+2.TRC}{3+3.TRC} \right) \times \left(\frac{DM \times AF}{WA} \right)^{0.3} \times \left(\frac{VDEAS}{TC} \right)^{0.5} \right]^{0.9} \quad (kg)$$

The constant factor, C1, is a coefficient relating to the configuration and type of aircraft. This can range from around,

0.0265 to 0.034

depending on the complexity of the wing in question. For an unmanned aircraft with a fairly clean wing, no undercarriage cutouts and simple flying controls a value of 0.028 has been used.

1.2 Fuselage Mass.

The estimation used for the fuselage mass is very simple and crude. The formula gives reasonable results for vehicles with constant cross-section bodies, but does not allow for complex body shapes. However, for the vehicle in question, which assumed a constant circular cross-section, the formula gives reasonable results. The formulae take into account skin mass, stringer mass and frame mass, it also gives the equivalent skin thickness.

Skin mass.

$$FM1 = 0.0542 \times FSA^{1.07} \times VDEAS^{0.743} \times CK2 \qquad (kgs)$$

where

$$CK2 = 0.22 + 0.36 \times \left(\frac{TARM}{FB+FH} \right) - 0.14 \times \left(\frac{TARM}{FB+FH} - 2 \right)^{1.5}$$

a function of fuselage length to diameter ratio, with the last term being zero if $\frac{TARM}{FB+FH} < 2$

This skin mass gives an equivalent skin thickness of:

$$SKINT = 0.2 \times CK2 \times FSA^{0.07} \times VDEAS^{0.743} \times 10^{-4} \qquad (m)$$

Stringer mass.

$$FM2 = 0.012 \times FSA^{1.45} \times VDEAS^{0.39} \times AF^{0.316} \times CK2 \qquad (kgs)$$

Frame mass.

$$FM3 = 0.12 \times (FM1 + FM2)^{1.07} \qquad (kgs)$$

This gives a total fuselage shell mass of:

$$FM = FM1 + FM2 + FM3 \qquad (kgs)$$

1.3 Tail Unit.

The formula provided here is one of a group which deals with the tail unit mass. Each covers one of three general configurations: 'T' tail, tailless and conventional low tail. The formula is very simple and provides relatively crude results. It is believed that a considerable increase in accuracy could be achieved if these equations were expressed in a form similar to that used in the wing mass estimations. However this would introduce considerably more input data relating to the tail. An assumption is also made that the tail area is 0.3×wing area.

Conventional tail.

$$TUM = 3.9 \times VDEAS \times STU^{1.2} \times 10^{-2} \qquad (kgs)$$

where STU is the tail unit area.

2. Systems Group.

This group covers four main systems; electrics, flying control, fuel and recovery. The first three system masses being taken as a factor of the design mass.

2.1 Electrical Systems.

$$ESM = 0.094 \times DM^{0.84} \qquad (kgs)$$

2.2 Flying Control System.

$$FCM = 0.16 \times DM^{0.75} \quad \text{(kgs)}$$

2.3 Fuel System.

$$FSM = 0.068 \times DM^{0.8} \quad \text{(kgs)}$$

This includes residual fuel.

2.4 Recovery System.

This mass estimation is taken from Ref. 2 and calculates the canopy area from the basic drag coefficient equation.

$$\text{canopy area,} \quad RSPA = \frac{2 \times DM \times G}{DENS \times CD0P \times VRD^2} \quad (m^2)$$

Typical values used for the drag coefficient, CD0P, and the descent velocity, VRD, were 0.75 and 6 m/s respectively.

A typical canopy material has a mass of about 0.0454 kg/m2. Factoring to allow for the true canopy area, the rigging lines and case, the total mass of the parachute is about 1.6 times the mass of the material based on flat area.

$$\text{total mass,} \quad RSPM = 0.0454 \times 1.6 \times RSPA \quad \text{(kgs)}$$

The total installed weight of the parachute system must include the means used to deploy the canopy and this can add upto 12% to the basic weight.

$$\text{installed mass,} \quad RSTIM = 112 \times RSPM/100 \quad \text{(kgs)}$$

3. Propulsion Group.

A data bank of nineteen current turbojet engines, giving engine sea-level thrust and engine mass, has been obtained. These can be seen in Fig. B.3.1 along with the two straight line 'fits' used to represent the data. The intersection point has been faired to provide continuity for the first and second derivatives of the mass function when the thrust is being altered in the optimisation process. This gives the following equations:

Engine Mass:

For: $TOTHR <= 1.5695$ kN

$$EM = 0.792 + (25.887 \times TOTHR) \quad \text{(kgs)}$$

For: $TOTHR >= 1.5895$ kN

$$EM = 28.765 + (8.177 \times TOTHR) \quad \text{(kgs)}$$

For: 1.5696 < TOTHR < 1.5895

$$EM = 0.01 \times \left[1 + 2 \times \left(\frac{3.X}{0.08}^{2} - \frac{1}{3} \times \frac{3.X}{0.08}\right)^{4}\right] + \frac{(y_1 + y_2)}{2} \qquad \text{(kgs)}$$

where

$$X = \frac{y_2 - y_1}{2} = 13.9865 - (8.855 \times TOTHR)$$

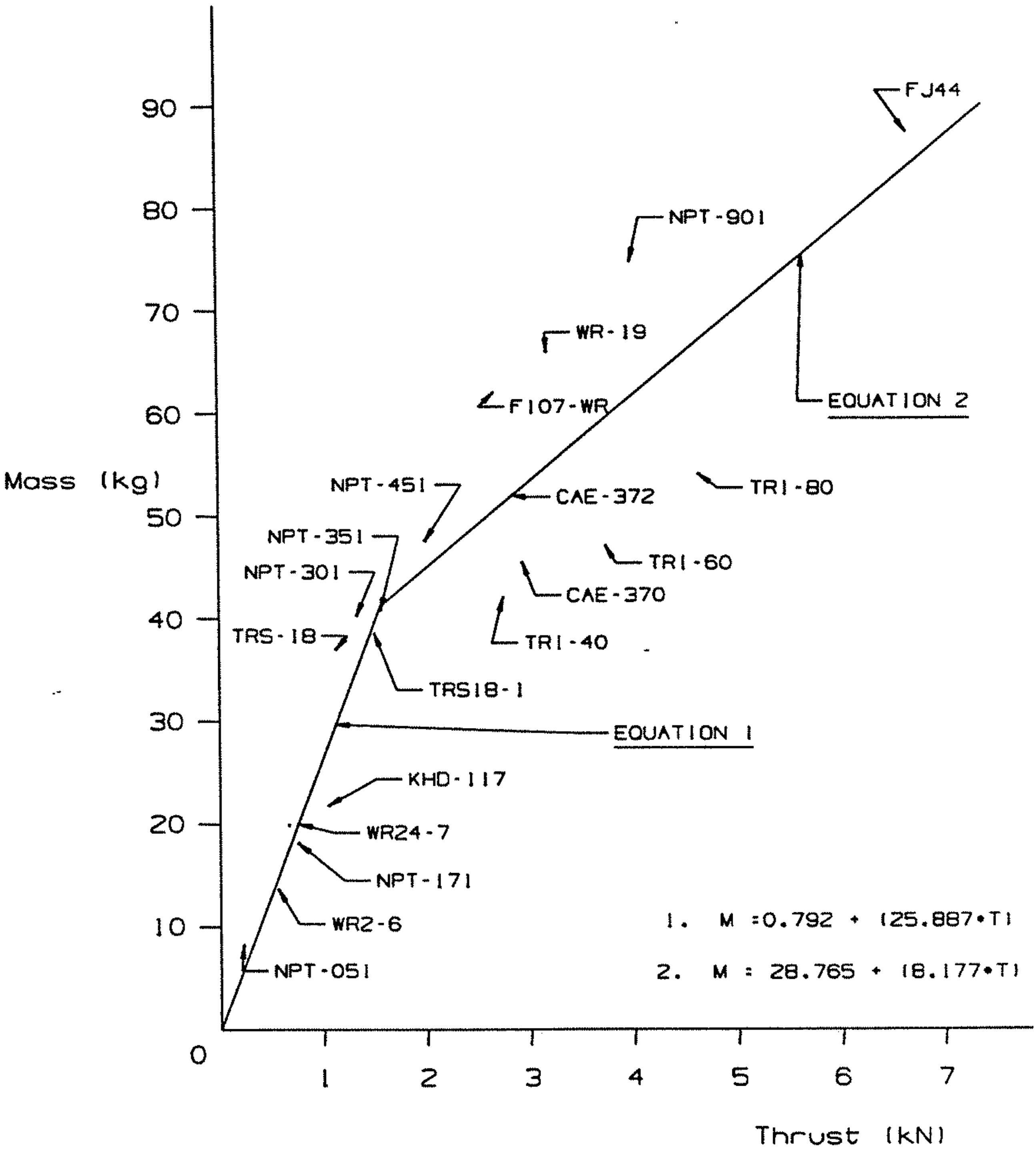
$$\frac{y_1 + y_2}{2} = 14.779 + (17.032 \times TOTHR)$$

Engine Installation.

An installation factor of 1.3 has been included giving a total installed propulsion system mass:

$$EIM = 1.3 \times EM \qquad \text{(kgs)}$$

Figure B.3.1 Engine Masses



APPENDIX C.

Packaging Module

This module deals with required fuselage geometry in order that the equipment and system are accommodated in the fuselage. This is defined using the dependent variables of the mass and geometry module.

1. Fuselage dimensions

These equations use 'engmax' which is the diameter required by the fuselage in order that the engine can be accommodated internally. This is calculated from the engine diameter and clearances required for operation. The fuselage volume and length equations can be seen later in the appendix.

1.1 Cross sectional area

$$FXSAR = \pi \times \left(\frac{ENGMAX}{2}\right)^2 \qquad (m^2)$$

1.2 Nose cone

length, $$FCLENR = \frac{\pi \times ENGMAX^2}{GOFI} \qquad (m)$$

volume, $$FCVOLR = \frac{GOFI \times FCLEN^2}{2} \qquad (m^3)$$

The value of GOFI used was 0·727 and includes any constant values required in the formula.

1.3 Tail cone

length, $$TCLENR = \frac{ENGMAX}{2} \qquad (m)$$

volume, $$TCVOLR = \frac{2}{3} \times \pi \times TCLENR^3 \qquad (m^3)$$

2. Fuselage compartments

The basic fuselage layout can be seen in Fig C.2.1. The equipment compartment volumes and lengths are calculated from the system masses, calculated cross sectional area and installed packaging densities, obtained from Ref 3. The fuel tank size is calculated from the fuel usage in the sortie analysis. The density values are external variables which can be user specified in the input file.

2.1 Recovery System

The package density of a recovery system depends primarily on the method of parachute packing. If the canopy and rigging lines are hand packed a density of about 320.0 kg/m^3 can be expected, compared with a value of about 560.0 kg/m^3 if machine packed. (Ref 4)

volume, $RSPSV = \frac{RSTIM}{RSPSD}$ (m^3)

length, $RSPSL = \frac{RSPSV}{FXSAR}$ (m)

2.2 Payload

A typical payload density used here is 640.8 kg/m^3. It is possible to have a second payload bay, this is handled by a 'flag' in the external variable input file which is set to 0 (integer value) for one payload or 1 for two payloads.

volume, $PAYVOL = \frac{PAY}{PAYDEN}$ (m^3)

length, $PAYLEN = \frac{PAYVOL}{FXSAR}$ (m)

2.3 Electrical and fuel systems

A typical density value for both electric and fuel systems is 480.6 kg/m^3. Both systems are taken together as one.

volume, $ELECVOL = \frac{(ESM+FSM)}{ELECDEN}$ (m^3)

length, $ELECLEN = \frac{ELECVOL}{FXSAR}$ (m)

2.4 Avionics

Density values of about 640.8 kg/m^3 were found for avionic systems.

volume, $AVIVOL = \frac{AVIM}{AVIDEN}$ (m^3)

length, $AVILEN = \frac{AVIVOL}{FXSAR}$ (m)

2.5 Engine

The volume of the engine compartment is defined from the engine length, calculated in the geometry module and the required cross-sectional area, calculated from the engine diameter plus the clearance required.

$ENGVOL = ENGLEN \times FXSAR$ (m^3)

2.6 Fuel tankage

The fuel density was taken as 800.0 kg/m^3 and the main fuel tank sizes given by:

volume: $$FTVOL = \frac{TOTFBM1}{FTDEN} \qquad (m^3)$$

where TOTFBM1 = fuel burned in sortie analysis (not including reserve fuel)

length: $$FTLEN = \frac{FTVOL}{FXSAR} \qquad (m)$$

The reserve fuel mass is calculated as a fraction of the total fuel mass, given by:

$$RESFBM = RESFF \times \frac{TOTFBM1}{(1-RESFF)} \qquad (kgs)$$

Giving reserve tank sizes as:

volume: $$RESVOL = \frac{RESFBM}{FTDEN} \qquad (m^3)$$

length: $$RESLEN = \frac{RESVOL}{FXSAR} \qquad (m)$$

3. Fuselage length and volume

The required fuselage length and volume are defined by the above compartment calculations and take into account whether the engine is externally or internally stored. A flag in the input file can be set to 1 for an external engine or 0 internal engine system.

Length:

$$FLBAS = RSPSL + PAYLEN + ELECDEN + AVILEN + FCLENR + TCLENR$$

If engine external:
length, $$COMPLEN = FLBAS \qquad (m)$$

If engine internal:
length, $$COMPLEN = FLBAS + ENGLEN \qquad (m)$$

Volume:

$$FVBAS = RSPSV + PAYVOL + ELECVOL + AVIVOL + FCVOLR + TCVOLR$$

If engine external:
volume, $$FUSVOLR = FVBAS \qquad (m^3)$$

If engine internal:
volume, $$FUSVOLR = FVBAS + ENGVOL \qquad (m^3)$$

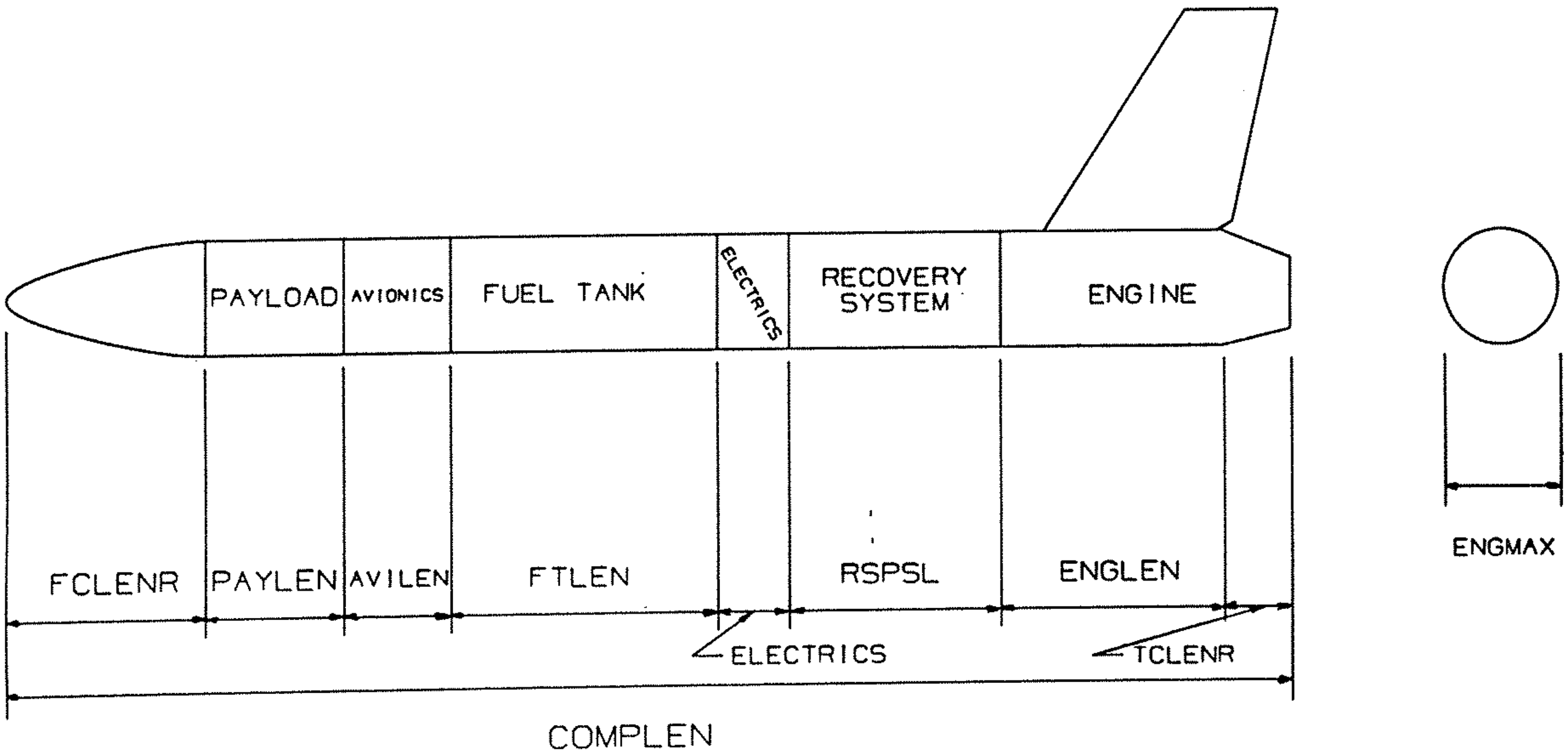


Figure C.2.1 Fuselage Layout

APPENDIX D.

Aerodynamics Module.

The aerodynamics section of the program is split into six separate modules. The modules are: lift-curve slope; zero lift drag; lift induced drag; design C_L and design Mach number; buffet onset lift coefficient and maximum usable lift coefficient. Each module is independent of the others and can be used when ever needed in the program, such as for the sortie analysis and point performance calculations (described later).

1. Lift-curve slope.

The lift-curve slope for the vehicle is calculated using the methods obtained from ref. 7 (section 4.3.1.2). This consists of estimating the lift-curve slope of the wing and body separately and then determining the wing-body combination from these and associated interference effects given by:

$$\text{CLAWBC} = (\text{KN+FBWK+FWBK}) \times \text{CLANW} \times \frac{\text{WAN}}{\text{WA}}$$

where WAN = nett wing area (m^2)
WA = gross wing area (m^2)
CLANW = nett wing $C_{L\alpha}$ (see 1.1)
$\text{FBWK+FWBK} = \left(\frac{\text{FB}}{\text{WS}} + 1\right)^2$
$\text{KN} = \frac{\text{CLAFB}}{\text{CLANW}} \times \frac{\text{SNREF}}{\text{WAN}}$
CLAFB = nose lift-curve slope
SNREF = reference area for the nose lift-curve slope, taken as the front cross-sectional area.

1.1 Wing lift-curve slope.

This is calculated based on ref 7 (section 4.1.3.2) and uses the nett wing area as the reference area.

$$\text{CLANW} = \frac{2 \times \pi \times \text{ARN}}{2 + \left[\frac{\text{ARN}^2 \times \text{BETA}^2}{K^2} \times \left(1 + \frac{\tan^2(\text{QW2D})}{\text{BETA}^2}\right) + 4\right]^{1/2}}$$

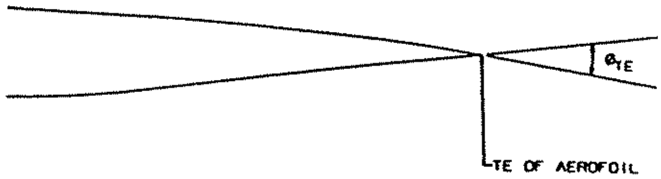
where ARN = nett wing aspect ratio
QW2D = mid-chord sweep

$\text{BETA} = (M^2-1)^{1/2}$ if $M^2-1 > 0$
$= (1-M^2)^{1/2}$ if $M^2-1 < 0$ (M = Mach No.)

$$K = \frac{1}{2\times\pi} \times [6.28+4.7\times (TC) \times (1+0.00375\times TEANG)]$$

TC = thickness chord ratio
TEANG = trailing edge included angle in degrees

The coefficient, K, above, represents the wing section lift curve slope and takes into account the section shape. The section trailing edge included angle is required for this calculation (see below)



As this angle depends upon the wing section used, and the thickness chord ratio of that section, one family of wing section has been chosen and the variation of TEANG with the thickness chord ratio calculated for this. The section chosen was the NACA 64- family and the variation is given by:

$$TEANG = 33.33 \times TC \qquad \text{(Ref. 7 section 2.2.1.8)}$$

1.2 Nose lift-curve slope.

This is calculated using ref. 7 (section 4.2.1.1) and is based on the fuselage front area.

$$CLAFB = \frac{2 \times APPMAS \times S_o}{\pi \times (FB/2)^2}$$

where
APPMAS is an apparent mass factor. This is represented in ref 7 (section 4.2.1.1, fig. 4.2.1.1-20a) in terms of the body fineness ratio, and is given by the equation below:

$$APPMAS = 0.5265 + (0.824\times10^{-1}\times FR) + (-0.5262\times10^{-2}\times FR^2) + (0.1137\times10^{-3}\times FR^3)$$

where FR = fineness ratio of the fuselage

Also, S_o is the cross-sectional area of the fuselage at the station where the flow becomes viscous. However, since the fuselage cross section is constant:

$$S_o = FXSA \qquad \text{(from the geometry module).}$$

2. Zero lift drag.

The zero lift drag coefficient estimation used was based on the method in ref. 8, and was achieved by summing the individual drag contributions of the various components of the vehicle under consideration.

$$CD0 = \frac{\Sigma\ (C_{Di} \times S_i)}{WA}$$

where $(C_{Di} \times S_i)$ is the drag area of each component
WA is the wing area (m^2)

In this method the sum of the component drag areas is given by:

$$\Sigma\ (C_{Di} \times S_i) = RRE \times RUC \times [RT \times (WCDS + FCDS) + ECDS]$$

The various contributions to this equation are described below.

2.1 Wing.

Uncorrected drag area for smooth wings.

$$WCDS = 0.0054 \times RW \times [1 + (3 \times TC \times \cos^2(QW4D))] \times WA$$

where RW = 1.0 for cantilever wings
= 1.1 for braced wings

2.2 Fuselage.

Uncorrected parasite drag for streamlined shapes.

$$FCDS = 0.0031 \times RF \times FL \times (FH+FB)$$

where RF = shape factor
= 1.0 cylindrical bodies,
= 1.3 rectangular bodies,
= 1.15 one side of the body rectangular, the other side rounded.

2.3 Tail unit.

It has been found, (8), that in practical cases the tailplane contribution amounts to approximately 24% of the fuselage plus wing drag areas, hence:

$$RT = 1.24$$

2.4 Engine installation.

The parasite drag of the engine is related to the installed thrust, this takes into account the variation of nacelle or intake scoop size and shape.

$$ECDS = 1.72 \times RN \times RTHR \times \frac{5+BYPASSR}{1+BYPASSR} \times \frac{TOTHR}{STHR \times p_o}$$

where p_o = sea-level pressure
RTHR = 1.0 thrust reversers installed
= 0.82 no thrust reverser
RN = installation factor
= 1.50 all engines podded
= 1.65 two engines podded, one buried in fuselage tail
= 1.25 engines buried in nacelles attached on the side of the fuselage
= 1.00 engines fully buried, with intake scoops on fuselage
= 0.30 engines fully buried, with wing root intakes.

2.5 Undercarriage.

Undercarriage drag can be accounted for by the factor, RUC. Here, since the majority of unmanned aircraft have no undercarriage a drag penalty in not necessary, so RUC=1.0 (see ref.8 for vehicles with undercarriages).

2.6 Correction for Reynolds number.

The following approximation accounts for the effect of the Reynolds number on turbulent skin friction drag and miscellaneous drag items.

$$RRE = 47 \times REF^{-0.2}$$

where $REF = \frac{CV \times FL}{XKINVIS}$
CV = speed of vehicle (m/s)
XKINVIS = coefficient of kinematic viscosity

3. Lift induced drag.

The lift dependent contribution to the drag coefficient, CDV, has been taken from the lower speed region of the method in ref. 6 (section 3.3.2). This assumes the lift dependent contribution to have a parabolic variation with the lift coefficient, C_L, given by:

$$CDV = \frac{CCK1 \times CL^2}{\pi \times AR}$$

where AR = aspect ratio
CCK1 is given below;

The parameter CCK1 is defined by the major geometric variables and in the subsonic speed range, $0 < M < 0.8$, is assumed not to vary with Mach number. The constant value is given by:

$$CCK1 = RCDCK \times FKID0$$

where; for $FKISK4 \geq 0.6159$

$$FKID0 = [FKISK4 \times (0.8137 + 0.767 \times FKISK4 - 0.742 \times FKISK4^2)]^{-1}$$

otherwise;

$$FKID0 = \frac{RCDCK}{FKISK4}$$

The variable RCDCK is a factor to allow for the effect of advanced technology in the aerodynamic performance. The constant FKISK4 is defined as:

$$FKISK4 = FKISK3 \times [FKISK1 \times (1-FKISK2) + FKISK2]$$

with
$$FKISK1 = 0.85 - 0.306 \times QW4D \times (1+0.535 \times QW4D) + (\log_{10}(AR)-0.5441) \times [-0.179-1.38 \times QW4D+ 2.433 \times QW4D^2-1.722 \times QW4D^3+0.431 \times QW4D^4]$$

$$FKISK2 = 0.515 - 0.1 \times \tan^{-1}[50 \times (0.91-TC)]$$

$$FKISK3 = 0.8738 - 0.795 \times TRC - 1.654 \times TRC^2 + 1.145 \times TRC^3$$

unless $TRC \geq 0.3833$

when $FKISK3 = 1.0$

4. <u>Design lift coefficient and Mach number.</u>

4.1 The design lift coefficient of a configuration is obtained from a method used in ref. 9 and is defined as a function of wing sweep, aspect ratio and thickness chord ratio. The equation used is:

$$X = 0.184 \times AR + 0.032$$

where $X = \frac{CLDES \times (AR)^{1/2}}{\cos(QW4D)}$ (for a wing with no camber)

Giving: $$CLDES = \frac{(0.184 \times AR + 0.032) \times \cos(QW4D)}{(AR)^{1/2}}$$

4.2 The design Mach number is obtained by calculating the two-dimensional drag divergence Mach number, MD2, using fig. D.4.1, and correcting this for sweep and aspect ratio effects, MSWEEP and MAR respectively, giving:

$$MDES = MD2 + MSWEEP + MAR$$

A series of straight line fits were obtained for fig D.4.1, with linear interpolation taking place for the intervening values of C_{LDES}.

MSWEEP is given by:

$$MSWEEP = 0.001533 \times QW4D + 0.0000266 \times QW4D^2 - 0.028$$

if QW4D > 30.0 degrees, otherwise

$$MSWEEP = -0.000133 \times QW4D + 0.0000511 \times QW4D^2$$

MAR is given by:

$$MAR = 0.13059 \times (AR)^{-1} + 0.03095 \times (AR)^{-2}$$

The values of MSWEEP and MAR have been obtained from ref. 9 and can be seen in figure D.4.2.

5. Buffet onset lift coefficient.

The buffet onset lift coefficient is given graphically by fig. D.5.1, obtained from ref.9, as a function of thickness chord ratio, sweep, C_{LDES} and MDES. Linear interpolation takes place for values of $(TC)^{2/3}$ between the shown data. The value DELTAM is given by:

$$DELTAM = SECMAC - MDES$$

where SECMAC is the vehicle configured Mach number.

6. Maximum lift coefficient.

The maximum lift coefficient for the vehicle is calculated by the method given in ref. 7 (section 4.3.1.4), which defines the C_{LMAX} of the section, wing and body separately and the combines them to obtain the wing-body C_{LMAX}. The calculations all use the major configuration geometry obtained from the independent variables and dependent variables of the geometry module.

6.1 Wing section maximum lift coefficient. (Ref. 7 section 2.2.1)

This assumes that a 6-series section is being used and the C_{LMAX} is given in terms of the thickness chord ratio by:

$$CLMAXS = 3.072 + (-13.22 \times TC) + (0.2567 \times TC^2) + (-0.02081 \times TC^3) + (0.7656 \times TC^4) + (-0.1060 \times 10^{-4} \times TC^5)$$

For the 6-series wing section the variation of leading edge sharpness parameter, DELTAY, with thickness chord ratio is given by:

$$DELTAY = 20.75 \times TC \qquad \text{(see fig D.6.1)}$$

6.2 <u>Wing maximum lift coefficient.</u> (Ref. 7 section 4.1.3.4)

This section is split into two parts depending on aspect ratio:

6.2.1 <u>Low aspect ratio wings.</u>

The wing is assumed to have a low aspect ratio if:

$$AR \leq \frac{4}{(COEFF1+1) \times \cos(QWLR)} \qquad \ldots(1)$$

where the coefficient COEFF1 is given as a function of taper ratio by:

$$COEFF1 = -0.7813\times10^{-2} + (4.646\times TRC) + (-14.03\times TRC^2) + (14.48\times TRC^3) + (-5.078\times TRC^4)$$

For low aspect ratio wings satisfying (1) the maximum lift coefficient is given by:

$$CLMAXW = CLMAXBASE + DELTACLMAX$$

where CLMAXBASE is the max. lift coefficient for the vehicle at Mach 0.2 and DELTACLMAX is a correction factor for speeds above Mach 0.2.

<u>CLMAXBASE</u>

This is given by fig. D.6.2 as a function of DELTAY, above and

$$\frac{(COEFF1+1) \times AR \times \cos(QWLR)}{BETA}$$

<u>DELTACLMAX</u>

This is an increment on CLMAXBASE for Mach numbers over 0.2 and is given by fig. D.6.3, in terms of Mach number and

$$(COEFF2+1) \times AR \times \tan(QWLR)$$

where COEFF2 is another taper ratio correction factor obtained from:

$$COEFF2 = -0.06175 + (2.919\times TRC) + (4.397\times TRC^2) + (-16.09\times TRC^3) + (9.766\times TRC^4)$$

6.2.2 High aspect ratio wings.

These are defined as wings satisfying:

$$AR > \frac{4}{(COEFF1+1) \times \cos(QWLR)}$$

and the max. lift coefficient is given by:

$$CLMAXW = \left(\frac{CLMAXW}{CLMAXS}\right) \times CLMAXS + DELTACLMAX$$

where CLMAXS is the wing section C_{LMAX}, above;

$\left(\frac{CLMAXW}{CLMAXS}\right)$ is given by fig. D.6.4, for a Mach number of 0.2, in terms of DELTAY and QWLR.

DELTACLMAX is again an incremental factor to correct for speeds over Mach 0.2 and is given by fig. D.6.5 as a function of Mach number, DELTAY and leading edge sweep.

6.3 Wing-body maximum lift coefficient. (Ref. 7 section 4.3.1.4)

The wing-body C_{LMAX} is obtained by factoring the wing-alone C_{LMAX}.

$$CLMAXWB = \left(\frac{CLMAXWB}{CLMAXW}\right) \times CLMAXW$$

The factor $\left(\frac{CLMAXWB}{CLMAXW}\right)$ is obtained from fig. D.6.6 and is a function of the fuselage diameter to wing span ratio and (COEFF2+1)×AR×tan(QWLR), from earlier.

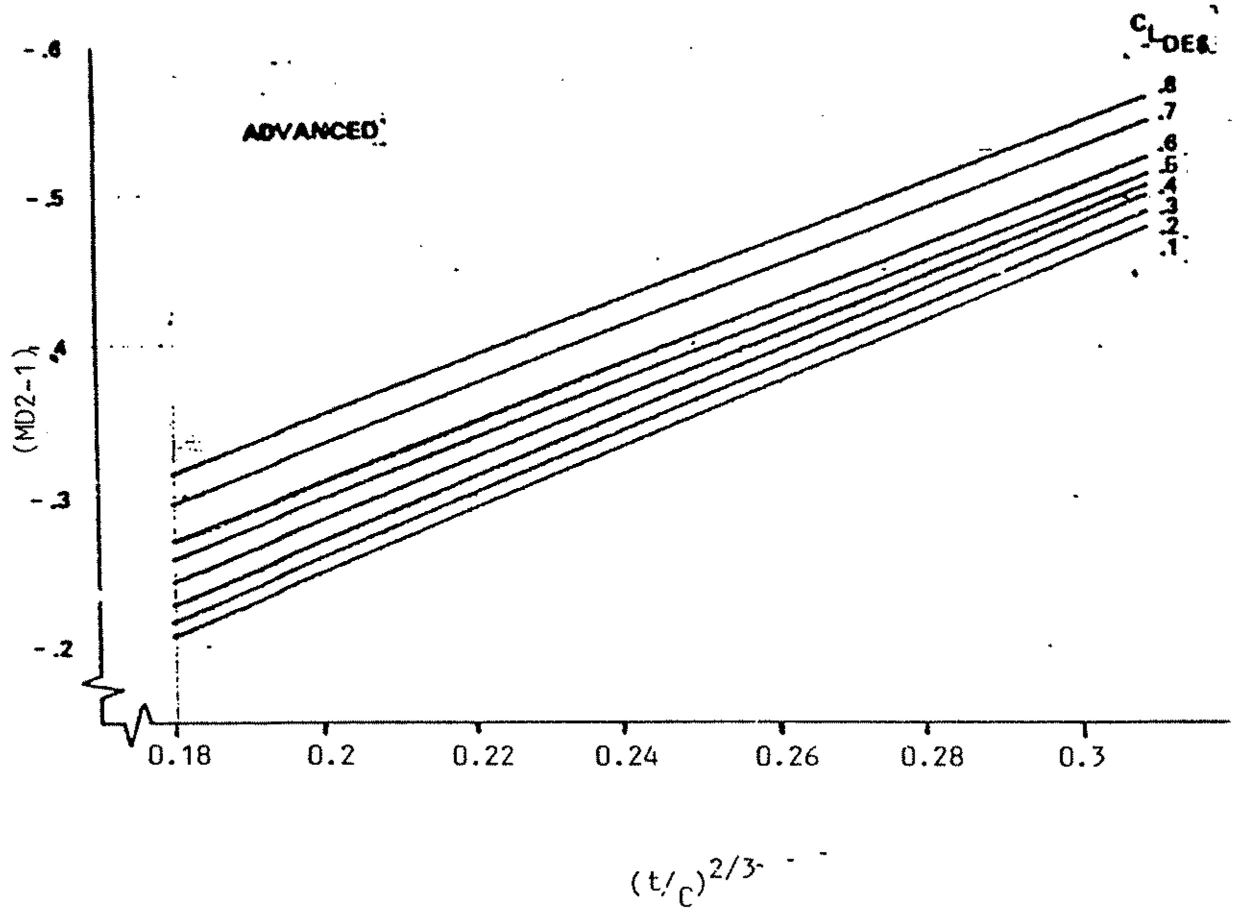


Fig. D.4.1 Drag Divergent Mach Number

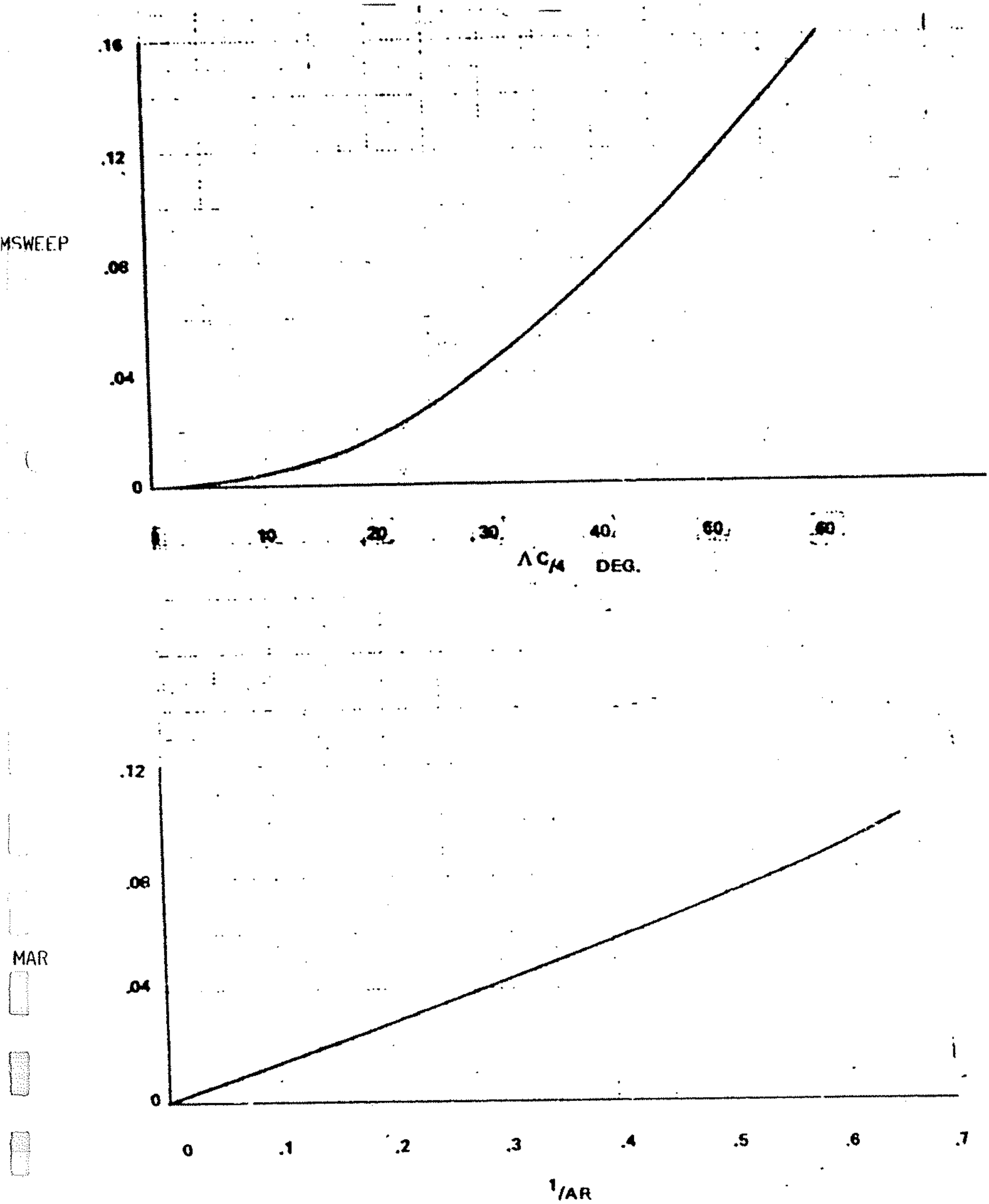


Fig. D.4.2 MD2 - Correction parameters

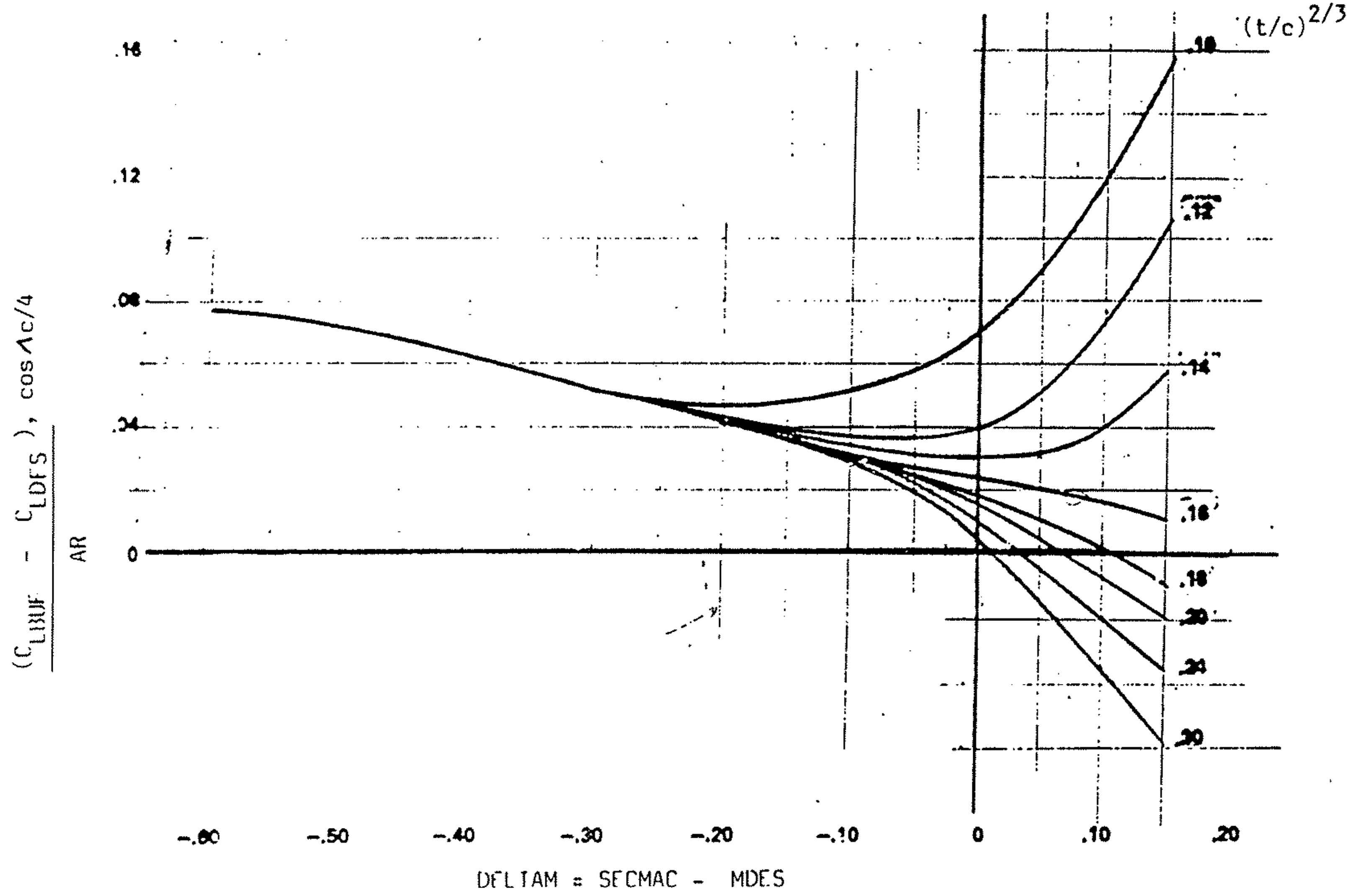


Fig. D.5.1 Buffet Onset

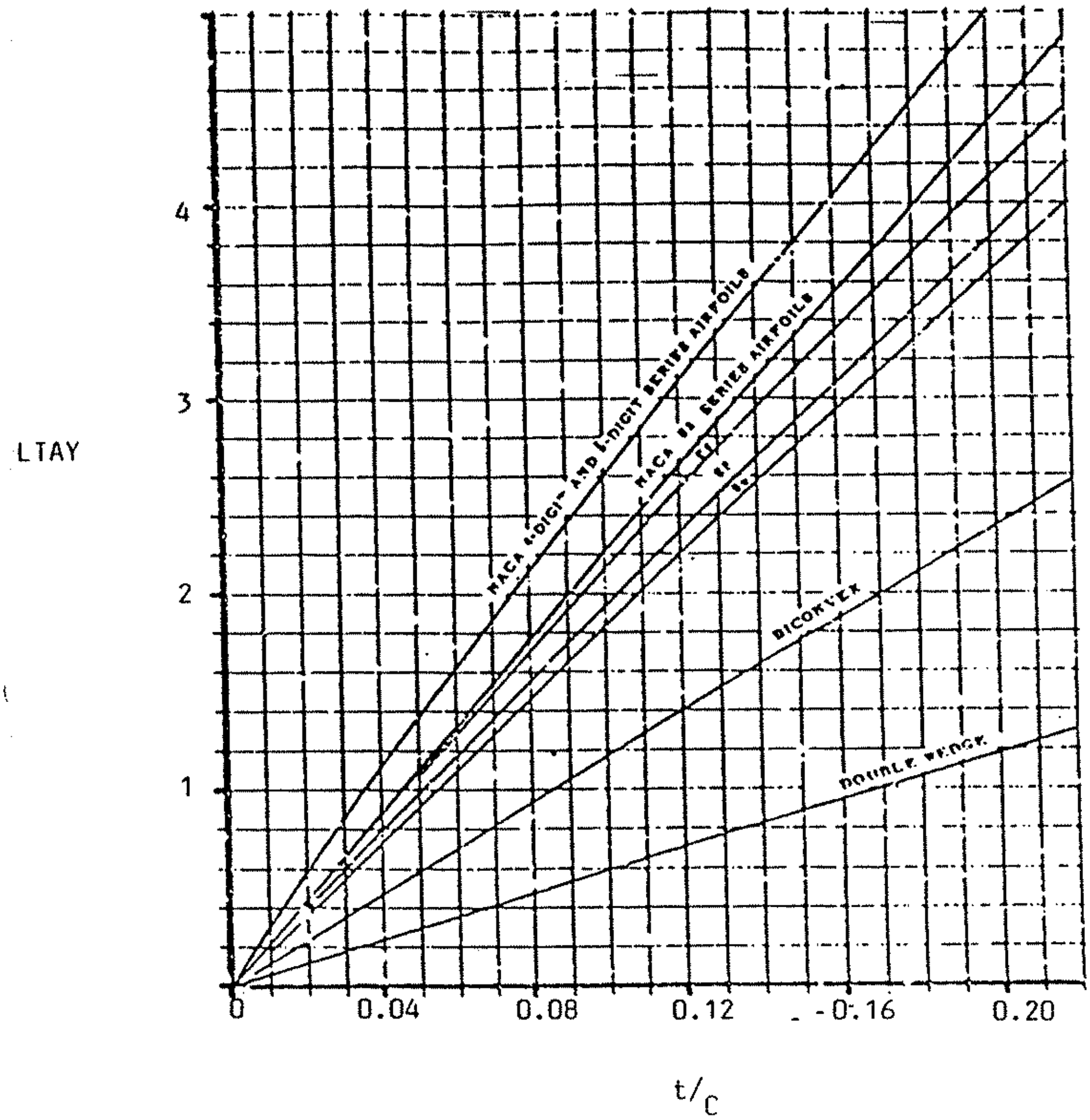


Fig. D.6.1 LE Sharpness Parameter

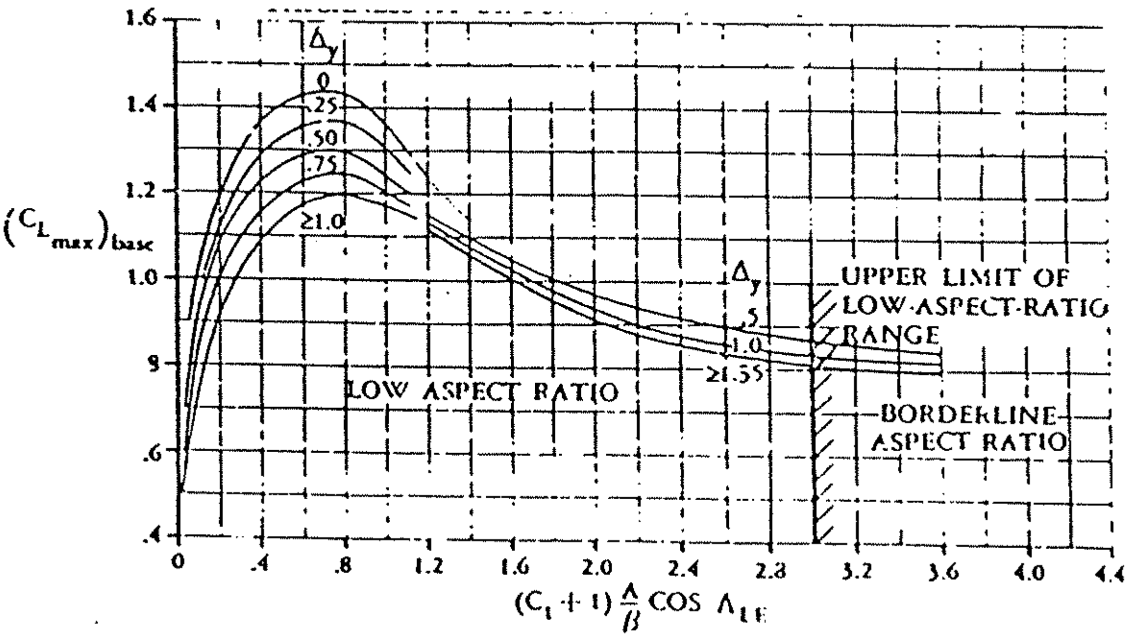


Fig. D.6.2 Maximum Wing Lift

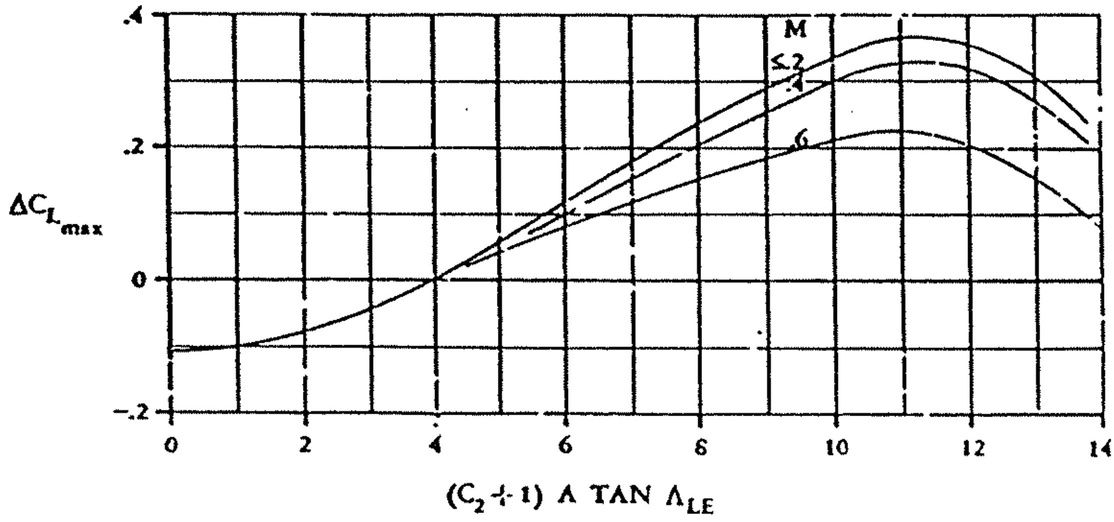


Fig. D.6.3 Max. Lift Increment

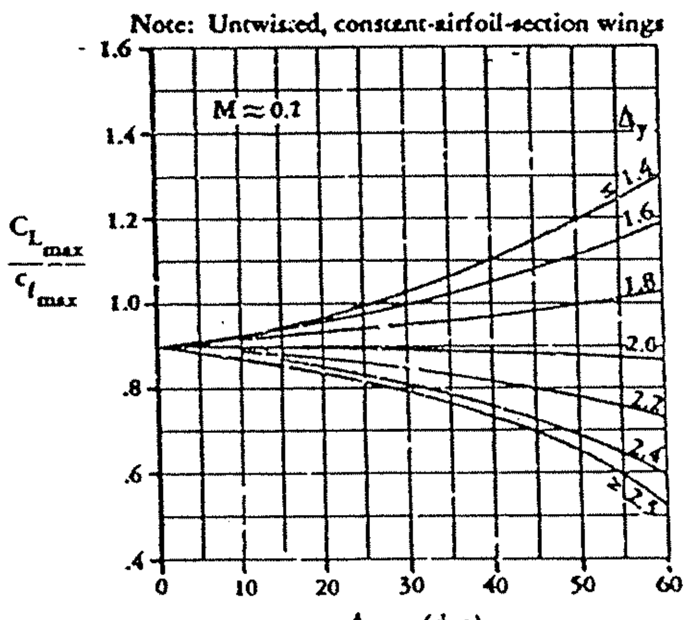


Fig. D.6.4 Max. Lift of High Aspect Ratio Wings

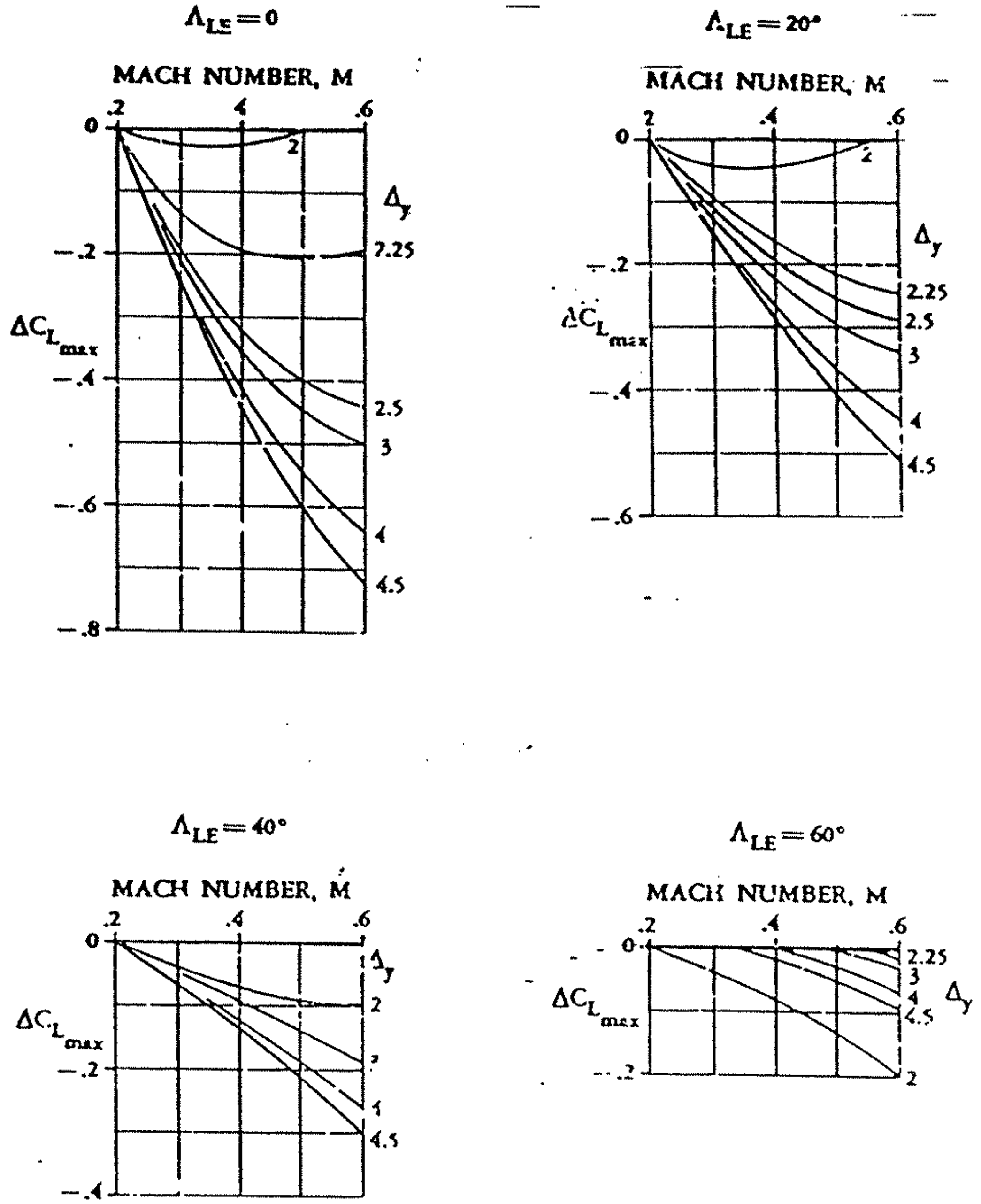


Fig. D.6.5 Mach Number Correction for High Aspect Ratio Wings

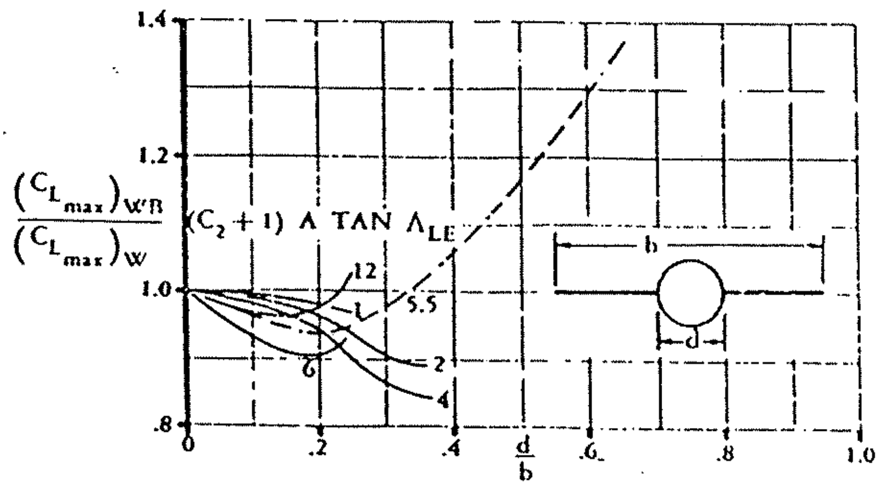


Fig. D.6.6 Wing-Body Max Lift

APPENDIX E.

Atmospheric Properties.

Atmospheric properties for altitudes upto 11000m, in the troposphere, and from 11000m to 20000m, the stratosphere, can be calculated from one module. The only input for this module is the altitude in metres. This is converted to feet and the following properties are calculated.

1. Troposphere. altitude < 11000m

temperature,	$temp = 288.16 - (0.0019812 \times alt)$
relative temperature,	$theta = \dfrac{temp}{288.16}$
relative pressure,	$delta = theta^{5.2561}$
pressure,	$press = delta \times 101325.0$
relative density,	$sigma = theta^{4.2561}$
density,	$dens = sigma \times 1.225$
speed of sound,	$vsound = 340.294 \times theta^{0.5}$
absolute viscosity,	$absvis = 0.1458 \times 10^{-5} \times \dfrac{temp^{1.5}}{(temp+110.4)}$
kinematic viscosity,	$xkinvis = \dfrac{absvis}{dens}$

2. Stratosphere. 11000m < alt < 20000m

temperature,	$temp = 216.65$
relative temperature,	$theta = 0.751874$
relative pressure,	$delta = 0.22415 \times e$
relative density,	$sigma = 0.29707 \times e$

where,

$$e = 2.7182818^{\left(\dfrac{-32.1741}{(3092 \times 216.65) \times (alt-36089)}\right)}$$

pressure,	$press = delta \times 101325.0$
density,	$dens = sigma \times 1.225$
speed of sound,	$vsound = 295.07$

APPENDIX F.

Engine Performance.

The engine performance module is based on the WILLIAMS INTERNATIONAL FJ44 turbofan engine. The performance data for this engine has been turned from graphical form to equations, using a curve-fit routine and the equation coefficients are stored in a data file, 'ENGINE.DAT', which is read at the beginning of the program and stored in various arrays.

For a given altitude, throttle setting and Mach number the nett thrust for the Williams FJ44 engine can be obtained. This nett thrust is the scaled using the sea-level thrust ratio. This is a ratio of the sea-level static thrusts of the required engine and the reference FJ44 engine. The required engine's sea-level static thrust is obtained as an independent variable under the control of the optimiser.

As well as the nett thrust obtained, the module calculates the airflow and the fuel flow for the FJ44 engine. These are also scaled using the same ratio factor as for the thrust. The gross performance data is also calculated using the nett values taking into account the airflow of the engine:

$$ETHRG = ETHRN + EFLOW \times CV$$

where ETHRG = gross thrust
ETHRN = nett thrust
EFLOW = air mass flow of the engine
CV = speed

The FJ44 is a modern two-spool, axial flow turbofan engine, with a medium bypass ratio and produces 1800 pounds (=8006.8 N) of take-off thrust at sea-level. Performance data equations have been obtained for the altitudes 0, 5000, 10000 and 20000 metres and for three throttle settings; 70%, 90% and 100%. These throttle settings equate to cruise, climb and take-off configurations. The graphical data can be seen in figures F.1 to F.12.

Interpolation takes place in order to obtain performance values at any configuration of altitude, throttle setting and Mach number, with the Mach number range being from 0 to 0.8. The performance values are used for obtaining sortie leg performances of fuel used and time taken and also in the point performance calculations.

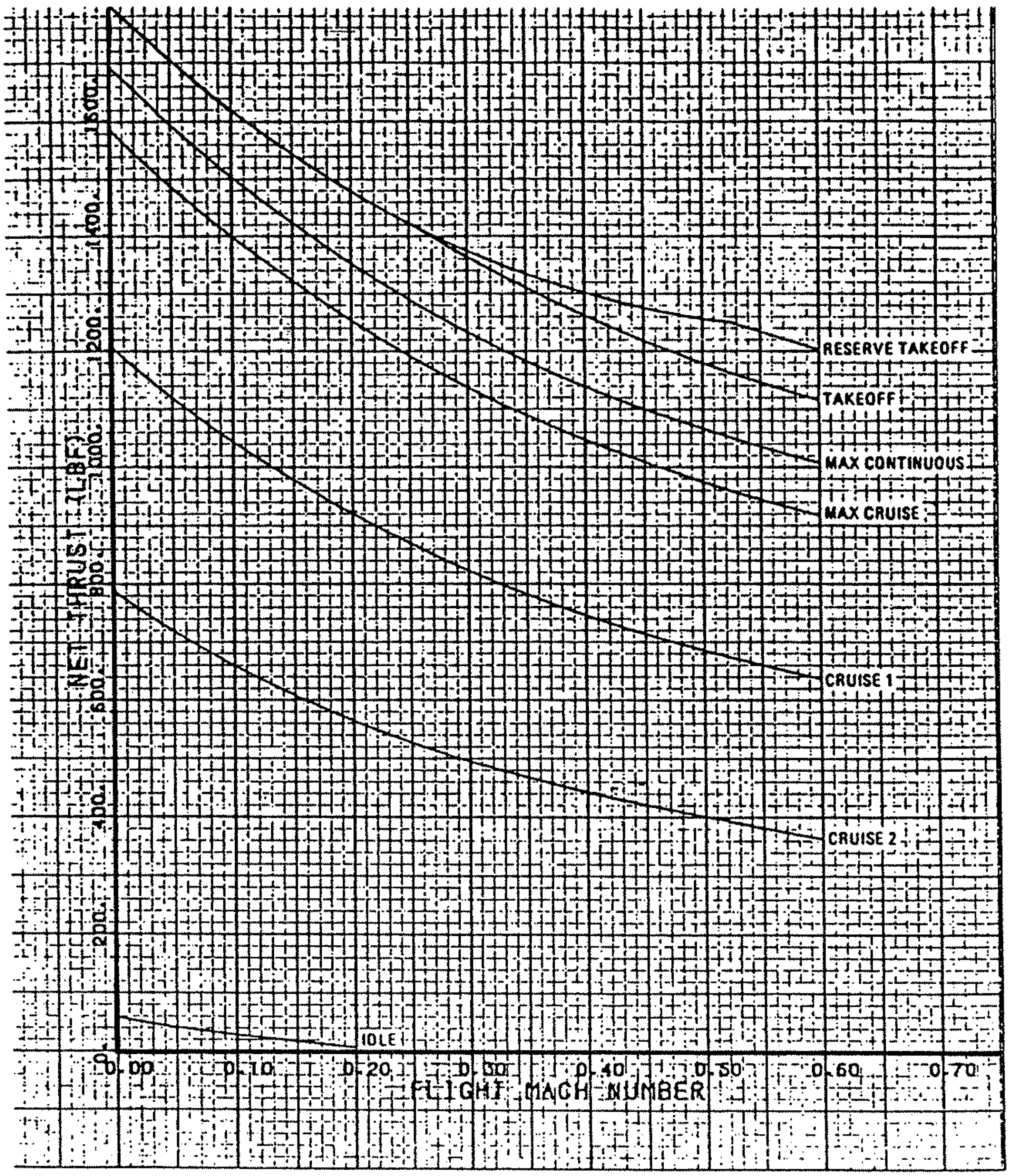


Fig. F1 Net Thrust - Sea Level, Standard Day.

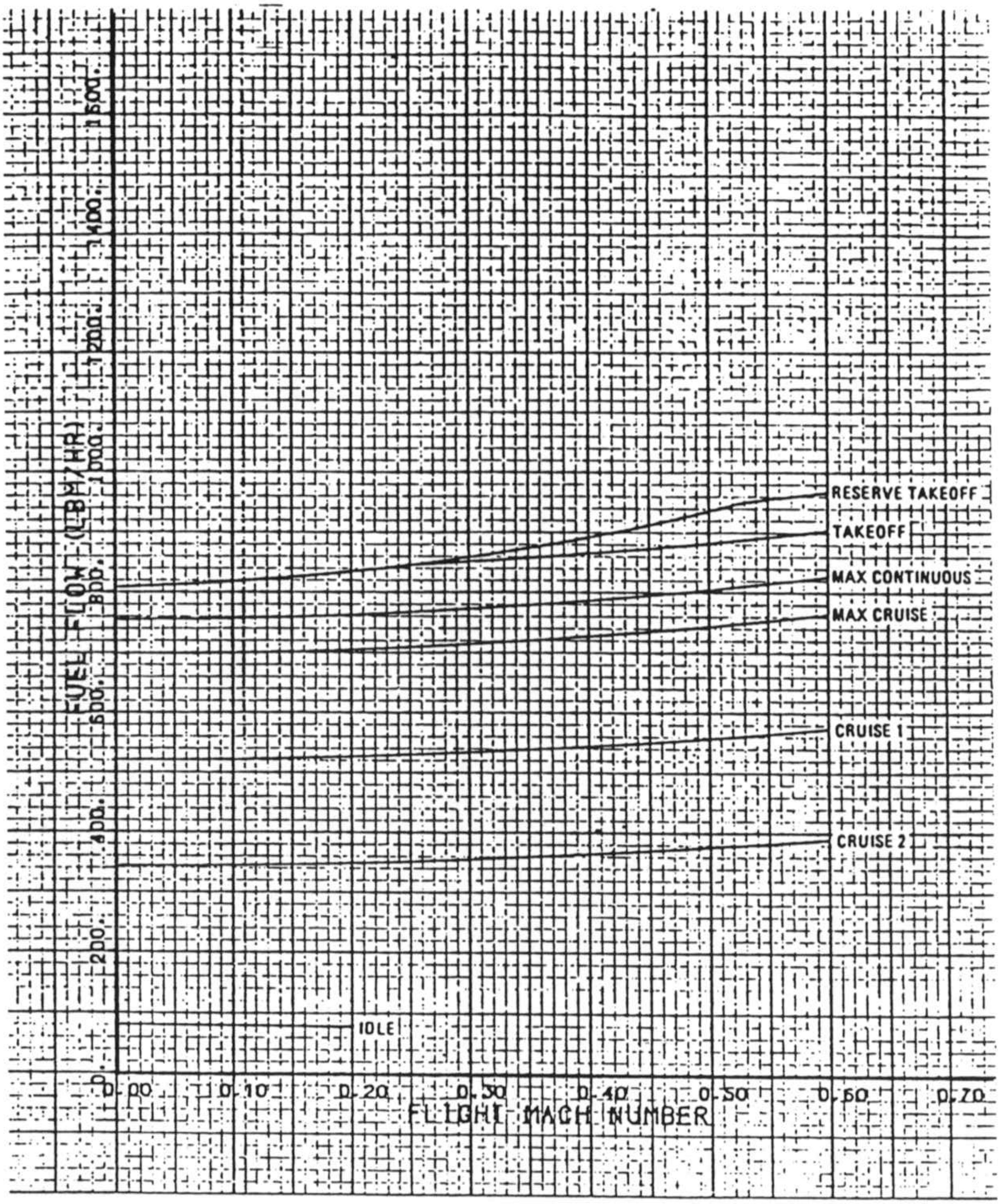


Fig. F2 Fuel Flow - Sea Level, Standard Day.

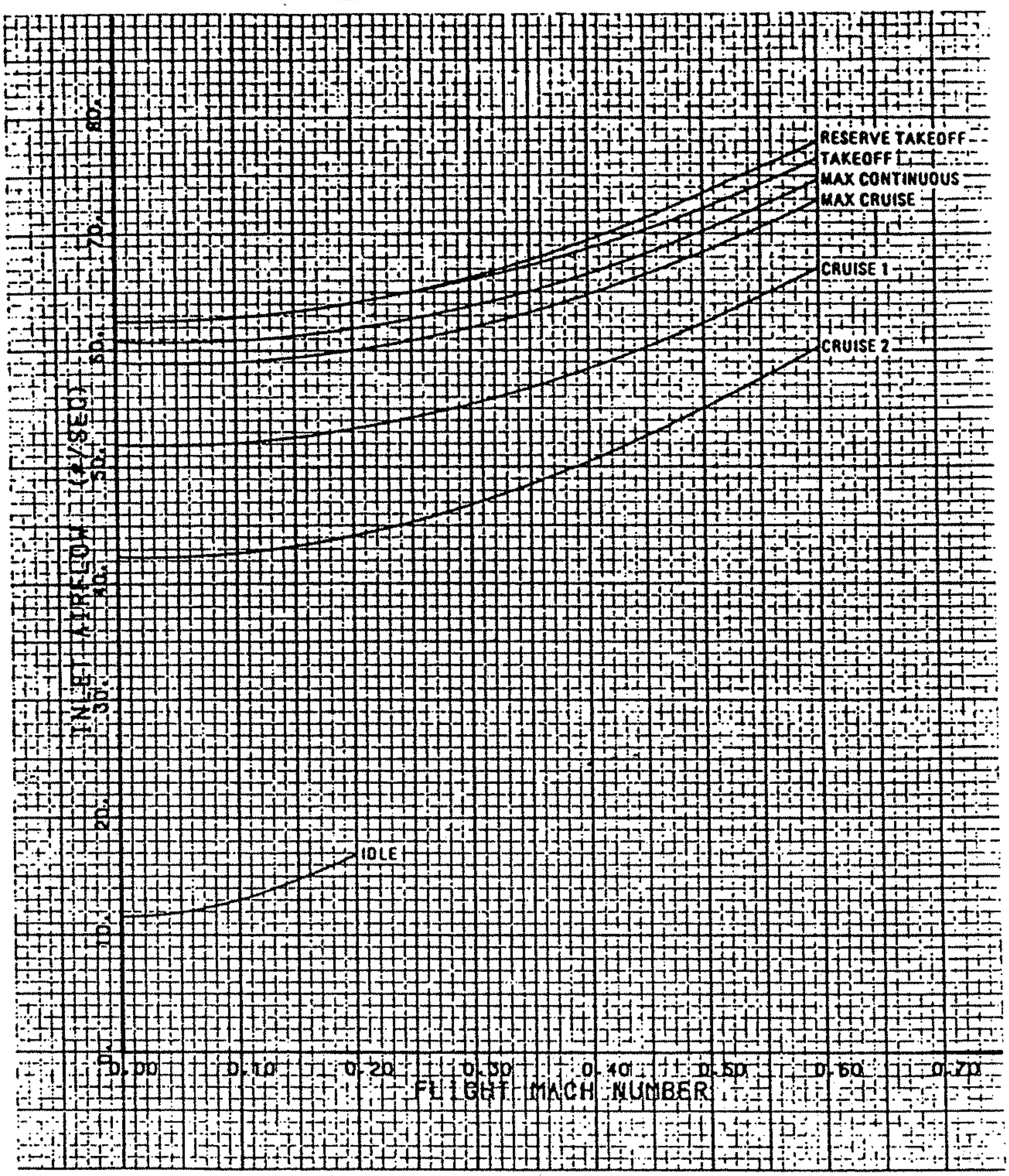


Fig F3 Inlet Airflow - Sea Level, Standard Day.

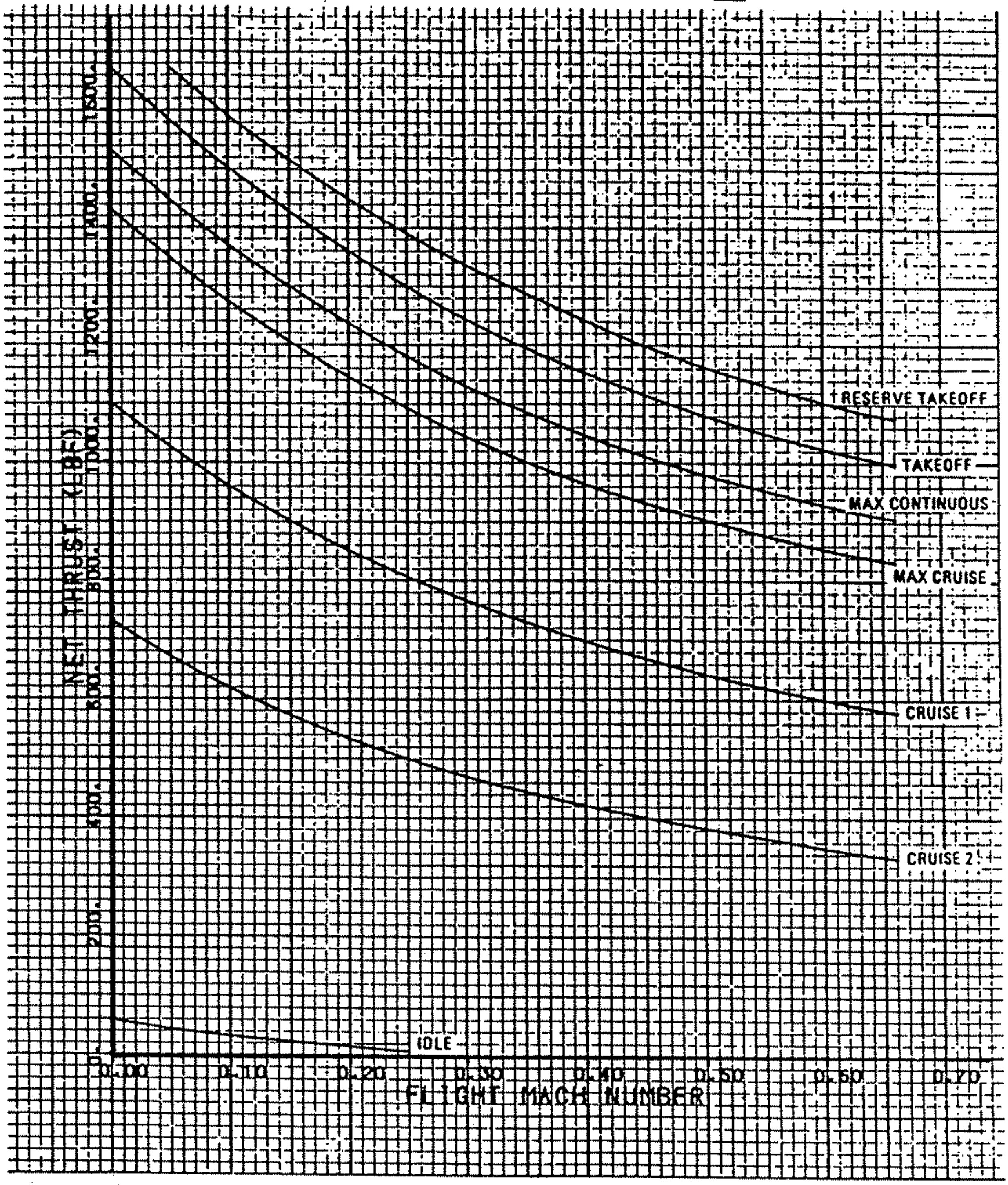


Fig. F4 Net Thrust - 5000 Feet, Standard Day.

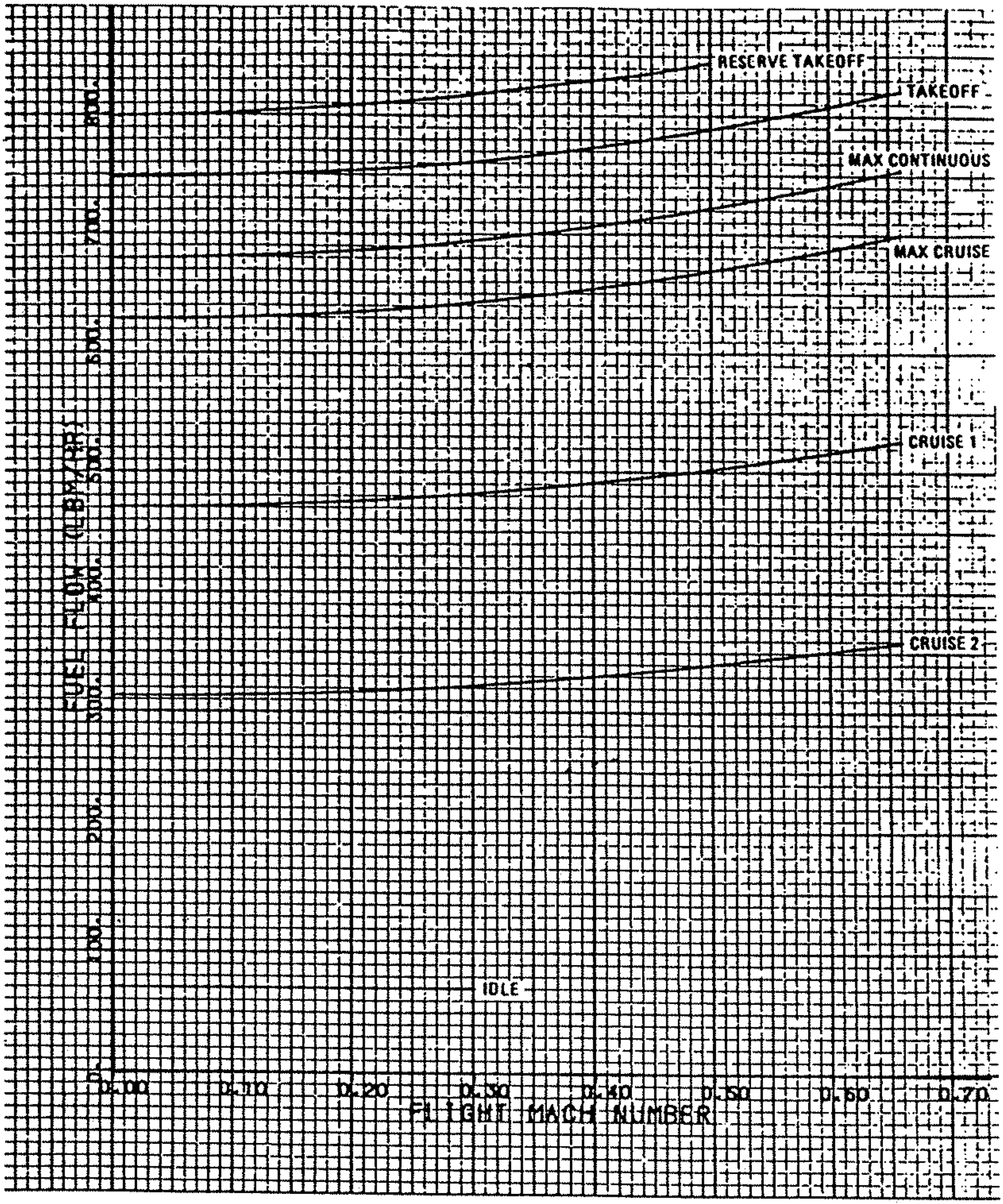


Fig. F5 Fuel Flow - 5000 Feet, Standard Day.

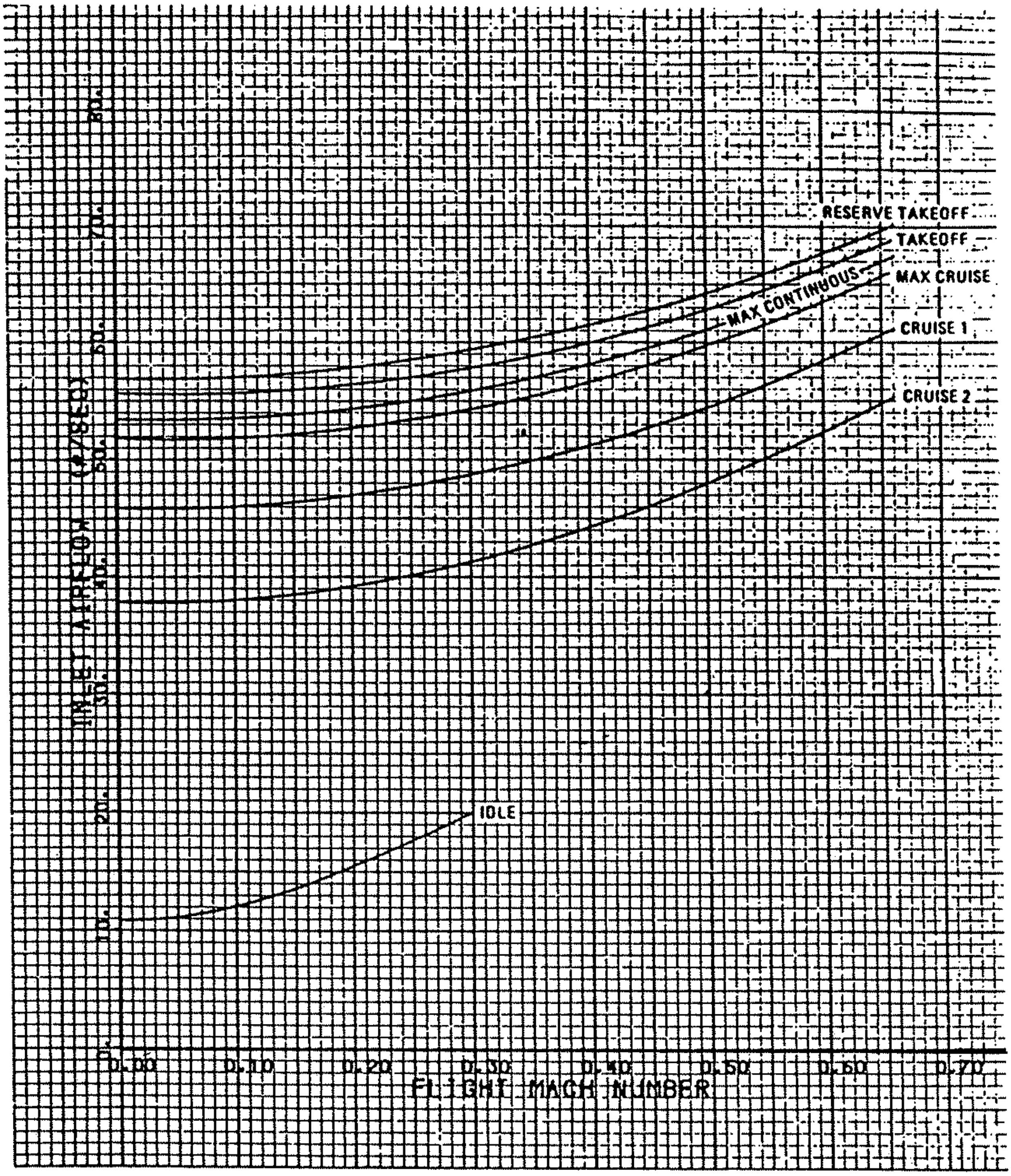


Fig. F6 Inlet Airflow - 5000 Feet, Standard Day.

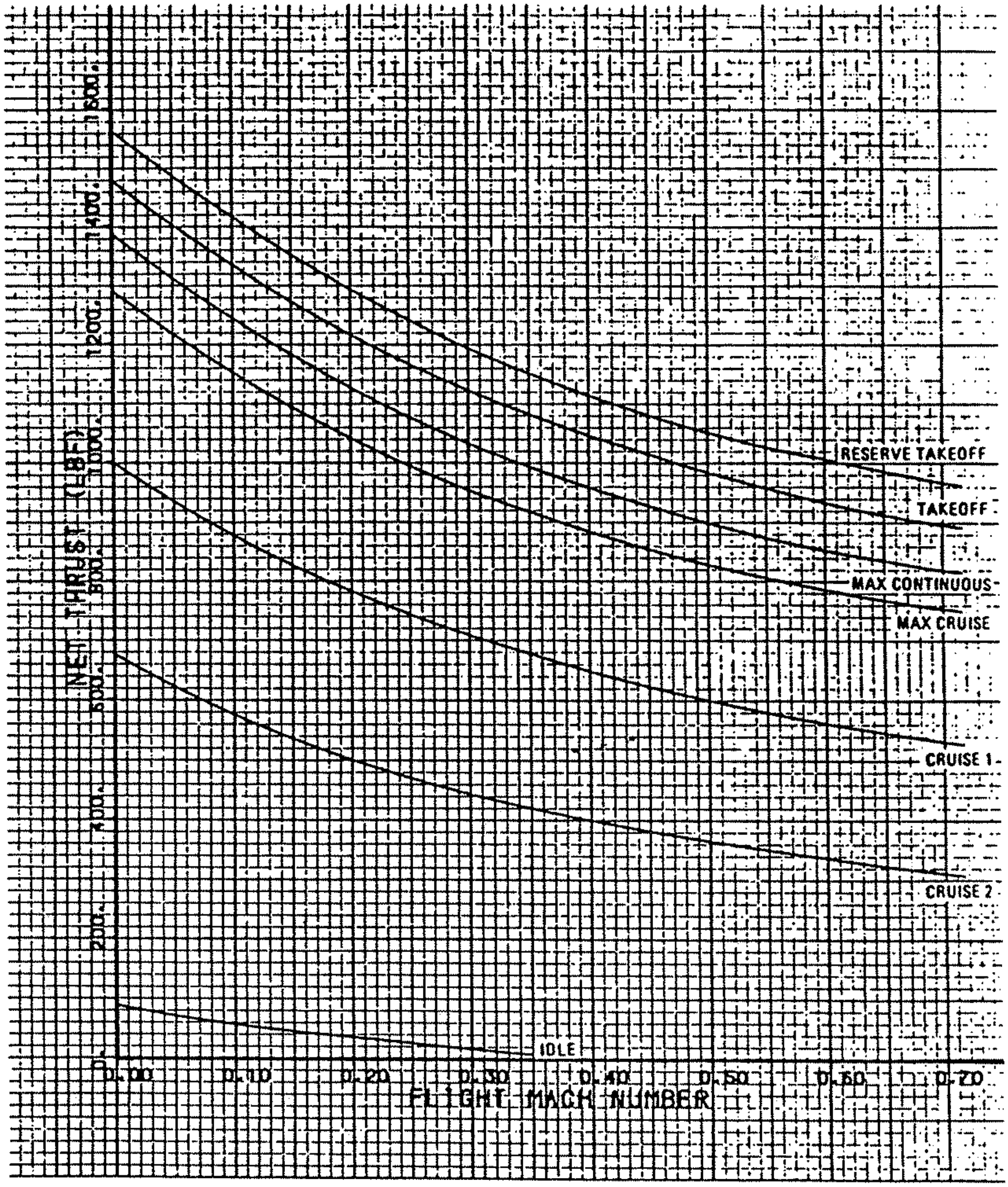


Fig. F7 Net Thrust - 10,000 Feet, Standard Day.

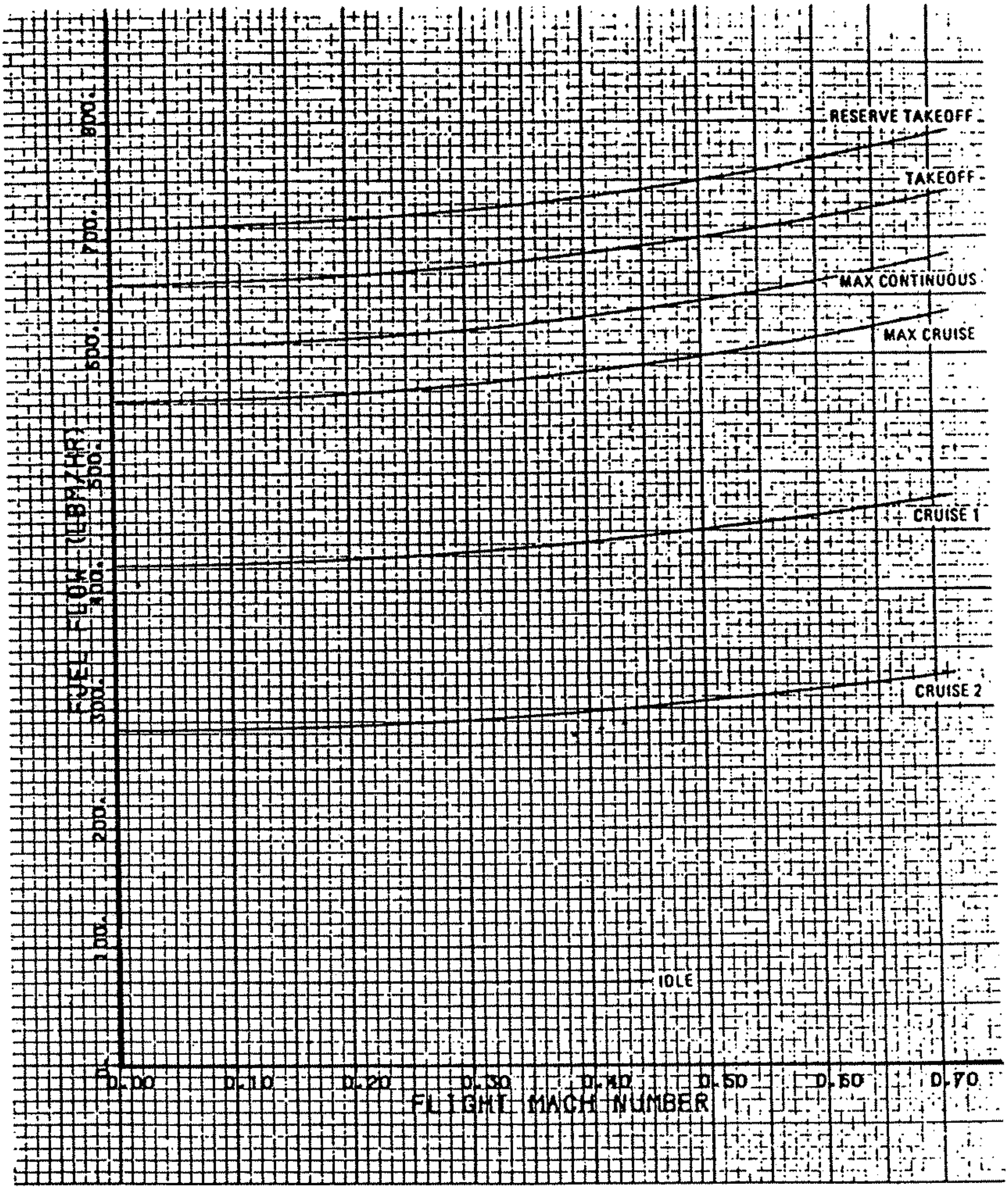


Fig. F8 Fuel Flow - 10,000 Feet, Standard Day.

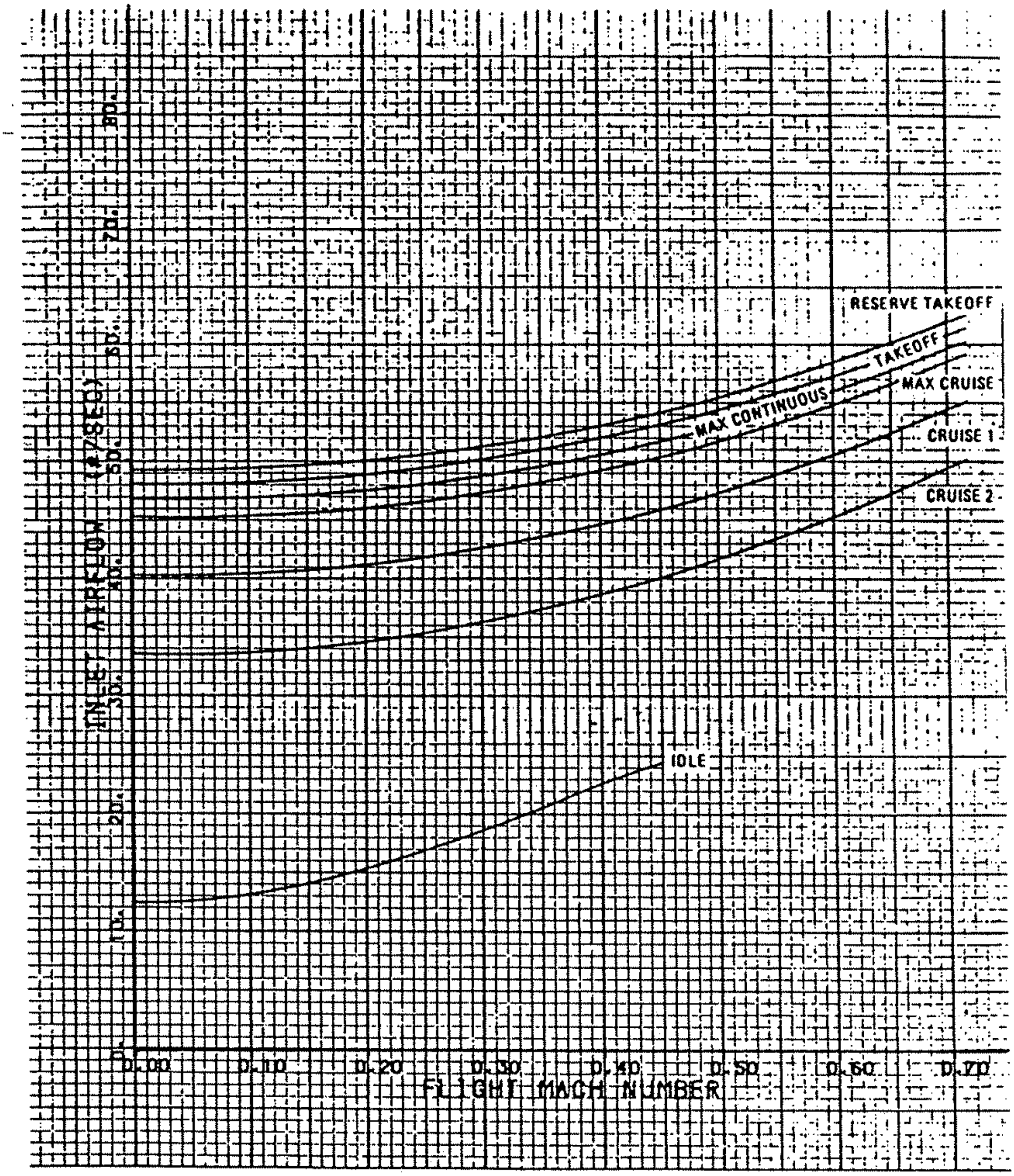


Fig. F9 Inlet Airflow - 10,000 Feet, Standard Day.

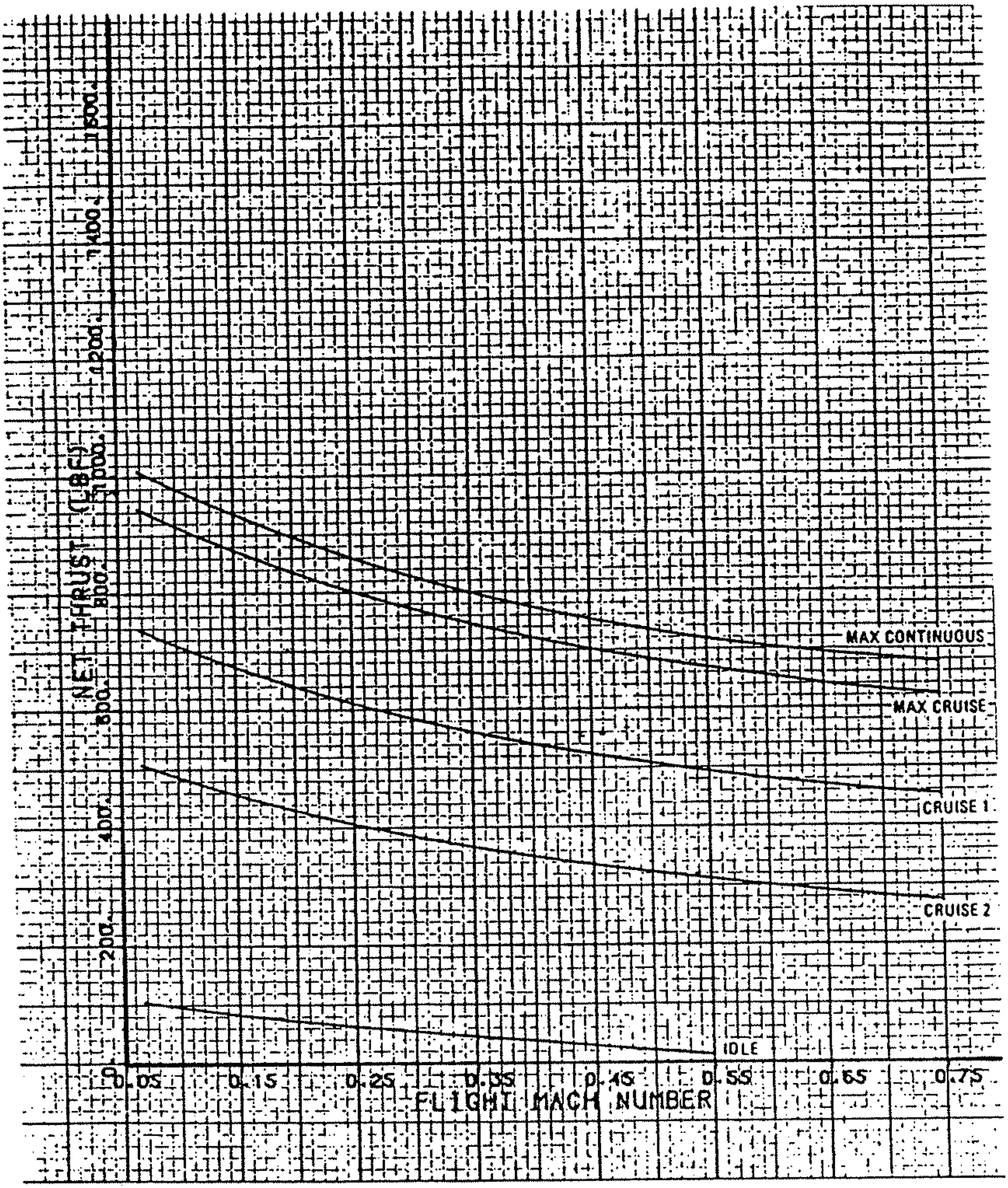


Fig. F10 Net Thrust - 20,000 Feet, Standard Day.

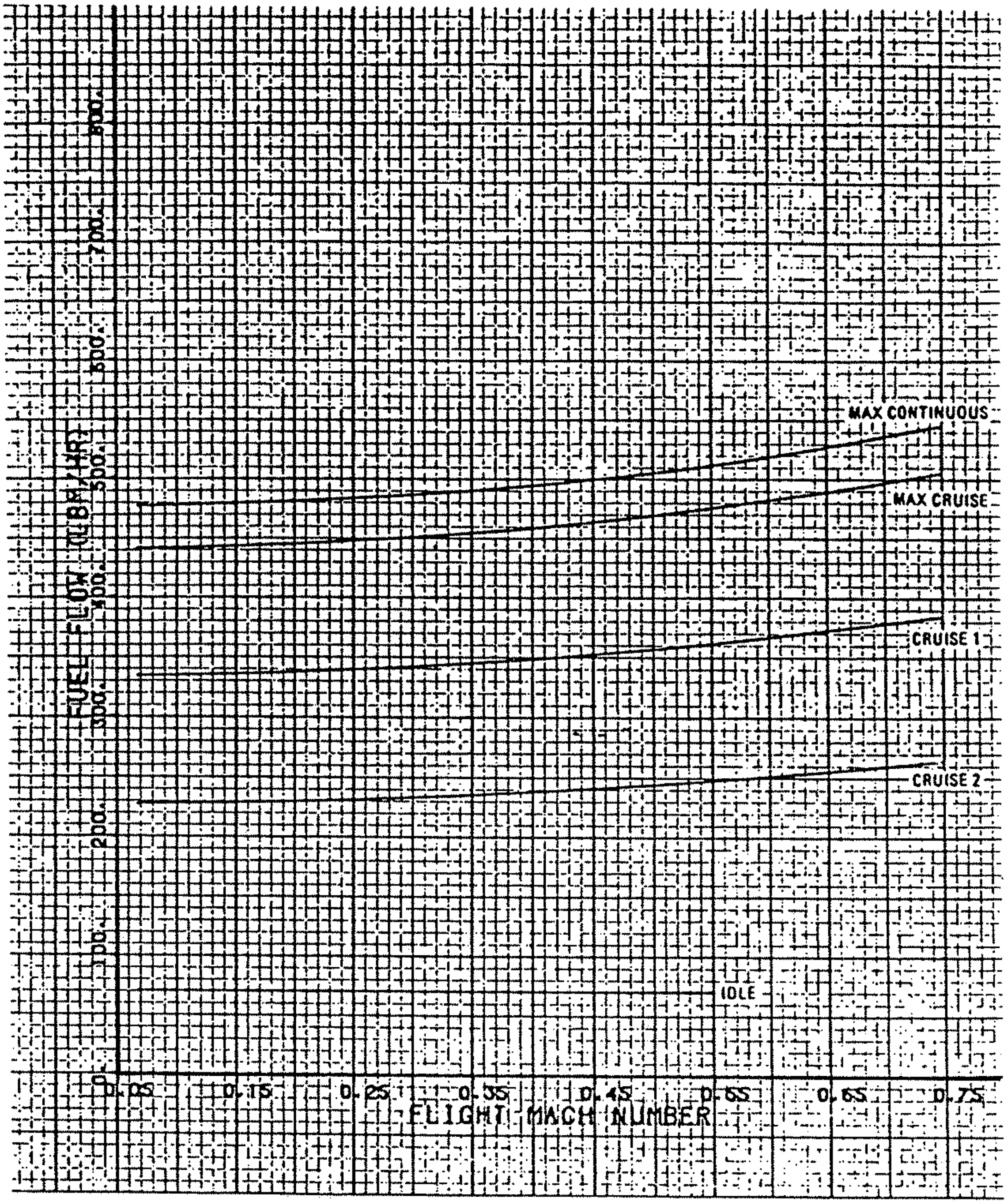


Fig. F11 Fuel Flow - 20,000 Feet, Standard Day.

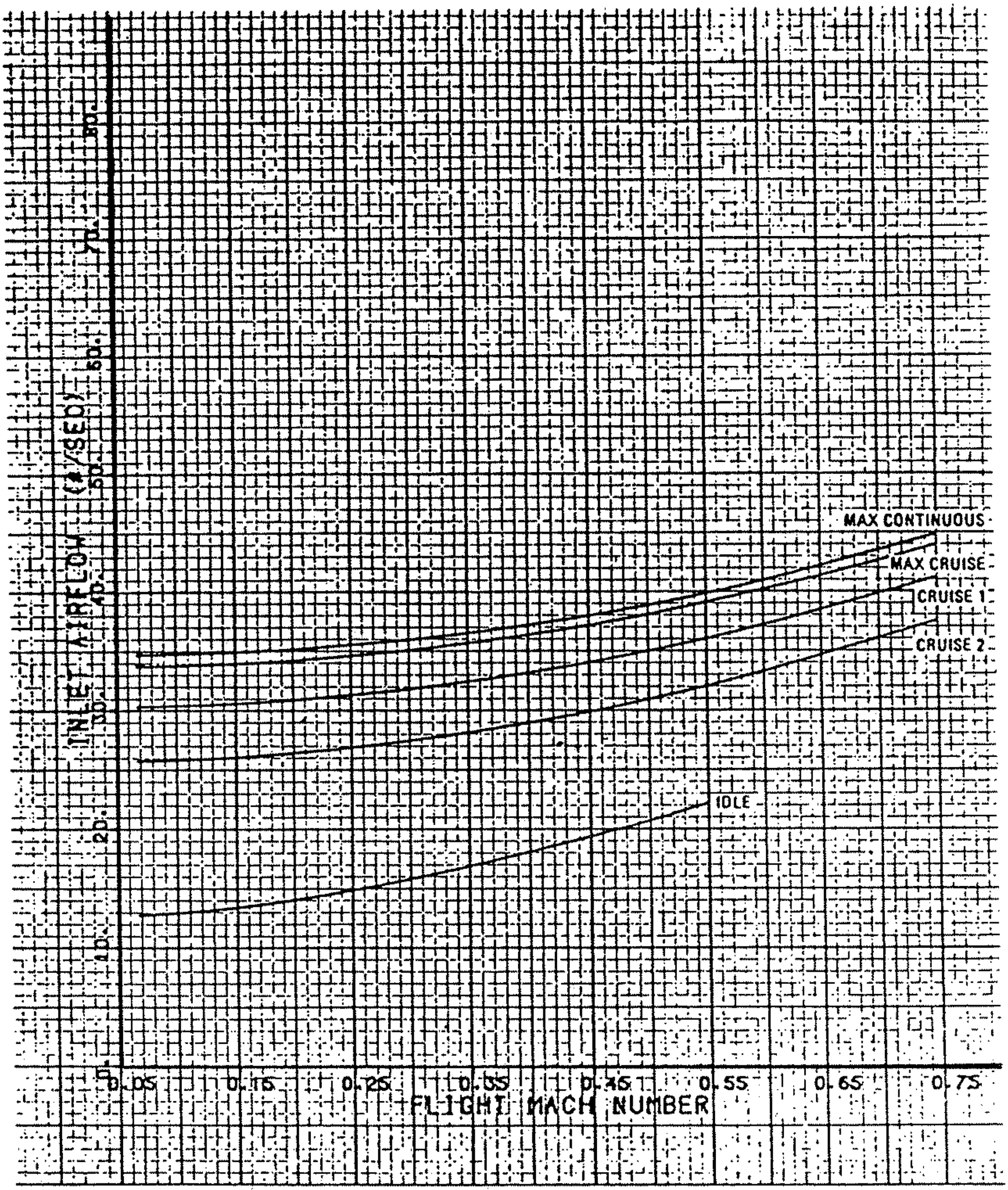


Fig. F12 Inlet Airflow - 20,000 Feet, Standard Day.

APPENDIX G.

Sortie Analysis.

The aim of the sortie analysis for the synthesized vehicle is to determine the fuel required and the mass of the vehicle at particular points of the sortie. This is important for the calculation of the point performance of the synthesized aircraft, described later.

As part of the external variables read into the program, a sortie of seven legs is defined. This sortie consists of: climb; cruise; loiter; acceleration; dash; manoeuvre and return phases, with the method and variables used for each leg analysis being described later. The analysis was simplified by considering the loiter, dash and return phases similar to the cruise phase only at differing configurations of altitude, speed and throttle setting. The total fuel used for the sortie is taken as the sum of the fuel used for each leg, thus giving the overall aircraft mass. Fuel reserves are taken as a fraction, user specified, of the total fuel mass.

1. Climb phase.

The climb performance is achieved by calculating the specific excess power of the vehicle, which for small climb angles and a constant true airspeed climb, can be assumed to be the vehicle's rate of climb.

The inputs required for this module are: initial and final altitudes, Mach number and throttle setting for the climb. These are designed in the design data file (external variables).

1.1 Analysis:

Initially, the climb altitude is divided into 50 equal segments, DELTAH.

$$DELTAH = \frac{(ALTI - ALTF)}{50.0} \qquad (m)$$

For each of these segments the specific excess power and the fuel flow rate is found. The fuel flow rate being found using the engine performance module (Appendix F), and specific excess power using:

$$SEP = \frac{(ETHRG - DRAGTOT)}{(9.81 \times DM)} \times CV \qquad (m/s)$$

These are calculated at the centre of each segment and are assumed constant throughout the segment. The fuel used is then calculated for the segment in question:

$$DELFBM = \frac{DELTAH}{SEP} \times FFLOW \qquad (kg)$$

and also the time taken:

$$DELTIM = \frac{DELTAH}{SEP} \qquad \text{(sec)}$$

This is achieved for each of the fifty segments, taking the reducing vehicle weight into account each time, until the climb altitude has been reached. Then:

$$\text{Total fuel used} = \Sigma \text{ Fuel used on each segment} \qquad \text{(kg)}$$

$$\text{Total time taken} = \Sigma \text{ Time taken for each segment} \qquad \text{(sec)}$$

2. Range calculations.

The range equation detailed here is used in the analysis of the cruise, loiter, high speed dash and return phases of the sortie. The inputs required for this module are altitude, throttle setting, Mach number and range. These are either given as external variables or can be calculated from the design data which define the leg to be analysed. Each phase is dealt with separately after the following analysis of the range equation.

2.1 Analysis:

The performance is calculated using the basic Breguet Range equation:

$$RAN = \frac{CV}{9.81 \times SFC} \times \frac{LIFT}{DRAG} \times \log_e(m_1/m_2)$$

where RAN = range (m)
SFC = specific fuel consumption (kg/Ns)
CV = velocity (m/s)
m_1 = initial mass (kg)
m_2 = final mass (kg)

If specific fuel consumption, SFC, is in kg/Nhr and range, RAN, is in km then the equation is transformed, giving

$$RAN = 0.36697 \times \frac{CV}{SFC} \times \frac{LIFT}{DRAG} \times \log_e(m_1/m_2)$$

OR

$$RAN = 0.84499 \times \frac{CV}{SFC} \times \frac{LIFT}{DRAG} \times \log_{10}(m_1/m_2)$$

and in terms of Mach number:

$$RAN = 0.84499 \times \frac{(CMN \times VSOUND)}{SFC} \times \frac{LIFT}{DRAG} \times \log_{10}(m_1/m_2)$$

Fuel used for the leg, FBM, is given by:

$$FBM = m_1 - m_2$$

and therefore:

$$FBM = m_1 - \frac{m_1}{10^{\left(\frac{RAN}{0.84499} \times \frac{SFC}{(CMN \times VSOUND)} \times \frac{DRAG}{LIFT}\right)}} \quad \text{(kg)}$$

The specific fuel consumption is obtained from the engine performance module using:

$$SFC = \frac{FFLOW}{ETHRN} \quad \text{(kg/Nhr)}$$

From the aerodynamics module the lift/drag ratio for the vehicle, in the configuration in question, can be calculated:

$$\frac{LIFT}{DRAG} = \left(\frac{CD0 \times DENS \times WS \times CV^2}{2 \times 9.81 \times DM} + \frac{2 \times CCK1 \times 9.81 \times DM}{DENS \times WS \times CV^2}\right)^{-1}$$

2.2 Cruise phase.

The external variables used to define the cruise leg of the sortie are: altitude; throttle setting; range and Mach number. These are used directly in the range module with the fuel burned, FBM, resulting. The time taken for the cruise is also obtained from:

$$CRTIM = \frac{CRRAN}{CV} \quad \text{(sec)}$$

2.3 Loiter phase.

The external variables needed here are: altitude; Mach number and time taken for loiter to take place. From these the loiter speed and the loiter range are calculated:

$$CV = LOMAC \times VSOUND \quad \text{(m/s)}$$

$$RAN = CV \times LOTIM \quad \text{(m)}$$

The throttle setting used is the same as that for the cruise phase. Then the range module is used to obtain the leg performance.

2.4 High speed dash performance.

This phase is similar to that of the loiter with the variables being: altitude; Mach number and time taken. The throttle setting is taken to be 95%. Again the range is calculated from the speed and the time taken and thus the fuel required is obtained.

2.5 Return phase.

This is similar to the cruise phase, variables being altitude, Mach number and range, with the throttle setting the same as in the cruise leg.

3. Acceleration phase.

The time to accelerate, at constant altitude and throttle setting, through an increment in Mach number, is calculated by assuming that over a small interval of Mach number the vehicle horizontal acceleration varies linearly with Mach number. The size of interval chosen was determined as $0.1\times M$, where M is the overall increment in Mach number (5). The fuel used for the acceleration is calculated using the time taken and average fuel flow for the small interval in MAch number. The total time and fuel used being the summation of the time and fuel for each interval.

The input required for this phase is: altitude, initial Mach number and final Mach number.

3.1 Analysis:

The overall increment in Mach number is split into smaller intervals, DELM.

$$DELM = \frac{(ACCMF - ACCMI)}{10}$$

The acceleration of the vehicle, F1, ie. the nett propulsive force per unit mass, is calculated at the beginning and end of each interval, Mach numbers, M1 and M2 respectively.

At M1:

$$F1 = [ETHRG\times\cos(ALFA) - 0.5\times DENS\times CV1^2\times WA\times CD - CV1\times EFLOW] \;/\; DM$$

where

CV1 = velocity (relating to M1)

A similar calculation is made for F2, with the vehicle at Mach number M2. The gross thrust and air mass flow are calculated for the given configuration of altitude, throttle setting and Mach numbers by the engine performance module, with the mass and drag of the vehicle being obtained from the appropriate sections described earlier.

The time to accelerate from Mach number M1 to M2, DT, is given by:

$$DT = A \times \frac{M2 - M1}{F2 - F1} \times \mathrm{Log}_e\,(F2/F1) \qquad \text{(sec)}$$

The average fuel flow over the interval is calculated using the fuel flow from the engine module, thus giving the fuel used for each interval, DELFBM.

$$DELFBM = FFLOW \times DT \qquad (kg)$$

This is completed for each interval upto the accelerated Mach number, with the mass being altered accordingly for the fuel loss. Then the total time to accelerate through the increment is obtained as the sum of the times to accelerate over each small interval making up the increment specified. The fuel is obtained in the same manner.

$$\text{Total fuel used} = \Sigma \text{ Fuel used on each interval} \qquad (kg)$$

$$\text{Total time taken} = \Sigma \text{ Time taken for each interval} \qquad (sec)$$

4. Manoeuvre phase.

The fuel used in this leg is calculated from the time taken for the manoeuvres. The input required is altitude, Mach number, throttle setting, number of turns to be sustained and 'g' load factor for the turns.

The rate of turn of the vehicle is calculated for the given speed and load factor by;

$$ROT = \frac{180 \times G}{\pi \times CV} \times (GNS^2 - 1)^{1/2} \qquad (deg/sec)$$

where GNS = load factor
CV = speed

The time taken for the specified number of turns is then given by;

$$MANTIM = \frac{360 \times NTURN}{ROT} \qquad (sec)$$

The fuel used is given by;

$$MANFBM = FFLOW \times MANTIM \qquad (kg)$$

The fuel flow is calculated from the engine performance module for the altitude, Mach number and throttle setting given as design data.

APPENDIX H.

Vehicle Point Performance Calculations.

Having obtained the mass of the vehicle during the sortie specified in the design data, it is possible to calculate various other aspects of the vehicles performance at several combinations of vehicle mass, Mach number, altitude, throttle setting and type of performance parameter. Details of these performance calculations are included in the design data (external variables) which are read at the start of the program. The following parameters may be calculated: sustained turn load factor; attained turn load factor; maximum Mach number obtainable in level flight and specific excess power.

1. Sustained turn performance. (Ref. 6)

The maximum normal load factor is given by:

$$GNS = \frac{0.5 \times DENS \times CV^2}{(9.81 \times MASSS)/WA} \times \left[\frac{1}{CCK1} \times \left(\frac{ETHRG}{0.5 \times DENS \times CV^2 \times WA} - CD0 \right) \right]^{1/2}$$

These variables are obtained from the respective aerodynamic, mass and geometry modules. The actual maximum value of the normal load factor that can be sustained may be limited to that tolerable by the structural strength of the vehicle; the ultimate load factor (AF×1.5).

The sustained turn rate in degrees per second is then given by:

$$SUSROT = \frac{180 \times G}{\pi \times CV} \times (GNS^2 - 1)^{1/2}$$

2. Attained turn performance. (Ref.5)

The maximum normal load factor that may be produced by the aerodynamic-propulsive system, GNA, is given by:

$$GNA = \frac{1}{9.81 \times MASSA} \left[0.5 \times DENS \times CV^2 \times WA \times CLMAXWB + ETHRG \times \sin\left(\frac{CLMAXWB}{CLAWBC}\right) \right]$$

where CLMAXWB = maximum lift coefficient of the vehicle
CLAWBC = lift-curve slope of the vehicle

These dependent variables being obtained from the relevant modules mentioned above.

The actual load factor that can be achieved may be limited again by the structural strength of the vehicle. The corresponding attained turn rate in degrees per second is then given by:

$$ATTROT = \frac{180 \times G}{\pi \times CV} \times (GNA^2 - 1)^{1/2}$$

3. <u>Maximum Mach number.</u> (Ref. 5)

The maximum Mach number that can be attained in 1'g' flight at a specified altitude is calculated by an iterative procedure based on the variation of the nett propulsive force with the vehicle speed.

An initial estimate of the maximum speed, VMAXI, is used to start the iteration. The nett propulsive force, F1, is calculated by:

$$F1 = ETHRG \times \cos(ALFA) - 0.5 \times DENS \times VMAXI^2 \times WA \times C_D - VMAXI \times EFLOW \quad \ldots.(1)$$

with the dependent variable's values being obtained from the respective modules, above.

An initial estimate for the gradient of the nett propulsive force with vehicle speed, DF, is used for the start of the iteration, given by:

$$DF = DENS \times VMAXI \times WA \times CD - EFLOW$$

An improved maximum speed, V2, is calculated by:

$$VMAXII = VMAXI - \frac{F1}{DF} \quad \ldots.(2)$$

A new nett propulsive force, F2, for speed VMAXII, is calculated using (1), and the gradient being calculated more accurately by:

$$DF = \frac{F2 - F1}{VMAXII - VMAXI}$$

Equation 2 is now used to calculate a new improved speed. These calculations are completed iteratively until convergence is obtained. This is when the modulus of the nett propulsive force is less than 0.000001×TOTHR, where TOTHR is the sea-level static thrust.

4. <u>Specific excess power.</u>

The specific excess power is calculated for 1'g' flight and is given by:

$$SEP = \frac{(ETHRG - DRAGTOT) \times CV}{9.81 \times MASSP}$$

with all symbols being described earlier.

APPENDIX I.

Optimisation Process.

1. Independent Variables.

A total of eight independent variables have been made available in the program, although any of these may be held constant if required by the user (see Appendix J). These variables are all listed below and are mentioned earlier in the appropriate modules in which they are used.

No.	Variable	Unit	Notation
1.	S/L static thrust	N	TOTHR
2.	wing area	m^2	WA
3.	aspect ratio		AR
4.	fuselage length	m	FL
5.	fuselage diameter	m	FB
6.	1/4 chord sweep	deg rad	QW4D QW4R
7.	thickness/chord		TC
8.	taper ratio		TRC

2. External Variables.

The external variables are all read from one input file at the initial stages of the program. These variables are set externally to the optimisation process and are constant throughout. Included here are design data variables, sortie analysis data, point performance data, densities and technology factors.

2.1 Design data.

Variable	Unit	Notation
design diving speed	m/s	VDEAS
normal acceleration factor		AF
payload mass	kg	PAY
avionics mass	kg	AVIM
fuselage flag		FLAGFUS
engine flag		FLAGENG
recovery system flag		FLAGRS
payload flag		FLAGPAY

The 'flags' mentioned above describe certain aspects of the vehicle configuration.

1. FLAGFUS = 1 circular cross-sectional fuselage
   = 2 elliptical cross-sectional fuselage
   = 3 rectangular cross-sectional fuselage

2. FLAGENG = 1 engine podded externally
   = 0 engine internally installed

3. FLAGRS = 1 recovery system included
   = 0 no recovery system installed

4. FLAGPAY = 1 two payloads
   = 0 one payload

2.2 Sortie analysis data.

The external variables presented here define each leg of the mission profile to be analysed.

Leg	Variable	Unit	Notation
climb	initial altitude	m	ALTI
	final altitude	m	ALTF
	Mach number		CLMAC
	throttle setting	%	CLRAT
	fuel estimate	kg	ECLFBM
cruise	altitude	m	CRALT
	Mach number		CRMAC
	throttle setting	%	CRRAT
	range	nm.	CRRAN
	fuel estimate	kg	ECRFBM
loiter	altitude	m	LOALT
	Mach number		LOMAC
	time	min	LOTIM
	fuel estimate	kg	ELOFBM
acceleration	altitude	m	ACCALT
	initial Mach no.		ACCMI
	final Mach no.		ACCMF
	fuel estimate	kg	EACCFBM
dash	altitude	m	DAALT
	Mach number		DAMAC
	time	min	DATIM
	fuel estimate	kg	EDAFBM
manoeuvre	altitude	m	MANALT
	Mach number		MANMAC
	throttle setting	%	MANRAT
	no. of turns		NTURN
	load factor		GNS
	buffet C_L flag		IBUFF
	fuel estimate	kg	EMANFBM

return	altitude	m	RETALT
	MAch number		RETMAC
	fuel estimate	kg	ECLFBM

Also included is a reserve fuel factor, RESFF, this specifies the required reserve fuel supply as a fraction of the total fuel. The buffet onset C_L will be given in the output if the buffet C_L flag, mentioned above, is set to 1 in the input file.

2.3 Point performance data.

The data for the three point performance parameters are listed here. These include the vehicle configuration data ie. altitude, speed and throttle setting and also the fraction of the total fuel at which the performances are to be calculated.

Parameter	Variable	Unit	Notation
sustained turn	altitude	m	ALTSPP
	Mach number		SECSPP
	throttle setting		RATSPP
	fuel fraction		FFST
	required 'g' load		GNSPP
attained turn	altitude	m	ALTAPP
	Mach number		SECAPP
	throttle setting		RATAPP
	fuel fraction		FFAT
	required 'g' load		GNAPP
Max. Mach number	altitude	m	ALTMAX
	fuel fraction		FFMM
	required Mach no.		REQMM
specific excess power	altitude	m	ALTSEP
	Mach number		SECSEP
	fuel fraction		FFSEP
	required SEP	m/s	REQSEP

2.4 Packaging densities.

The data included here is the equipment packaging densities used in the packaging module in order to obtain equipment compartment volumes and lengths. Typical values of these have been obtained from Ref. 3 and these take the installation into account. In the case of the recovery system two values are given, one is for when the system is to be hand packed and the other for machine packaging.

Density	Notation	Typical value (kg/m^3)
payload	PAYDEN	640.8
recovery	RSPSD	320.0 (hand) 560.0 (machine)
avionics	AVIDEN	640.8
electrics	ELECDEN	480.6
fuel system		480.6
fuel	FTDEN	800.0

2.5 Miscellaneous variables.

Included here are technology factors and constants for the recovery system sizing. The technology factors, TFMEC1 and TFMEC2, are required in the mass estimation to account for advanced materials used in the manufacture. the formulae used assume aluminium alloy fabrication and for this the factor is set to one. Suggested factors for non-aluminium alloy manufacture are:

carbon composite	15% reduction	(factor = 0.85)
glass fibre	5% increase	(factor = 1.05)

The factor, RCDCK, in the induced drag calculation, is to allow for the effect of advanced technology in the aerodynamic performance. This also being set initially to one.

The recovery system constants included are the parachute zero lift drag coefficient, CD0P, and the speed of descent of the recovering vehicle, VRD. Typical values of these are 0.75 and 0.6m/s, respectively.

3. Summary of the constraint functions.

The following summary describes the constraint functions which can be applied. Convergence has not been achieved using all twelve constraints, but it is up to the user to decide on which constraints are required. Constraints can be cancelled by deleting them from the optimiser input file (see Appendix J). The design relationships in the constraints have all been explained in the earlier appendices.

3.1 Fuel used.

This constraint states that the amount of volume available in the fuselage for fuel is adequate to accommodate the fuel required, calculated by the sortie analysis.

Notation

func(2) = abs(FUELMAS- TOTFBM) - 0.01

3.2 Fuselage volume.

The volume used in this constraint is the central fuselage volume, this does not include the nose and tail cones. The constraint is satisfied when the geometric fuselage volume is greater than the fuselage volume calculated by the individual compartment volumes in the packaging module. Thus ensuring that the equipment can be accommodated in the central fuselage section.

Notation

func(3) = (FUSVOLR+FTVOL+RESVOL-TCVOLR-FCVOLR) - (FUSVOL-TCVOL-FCVOL)

3.3 Fuselage length.

This constraint ensures that the geometrical fuselage length, FL (independent variable), is greater than the fuselage length calculated by the equipment compartments in the packaging module.

Notation

func(4) = (COMPLEN+FTLEN+RESLEN) - FL

3.4 Fuselage diameter.

The constraint here ensures that the geometrical fuselage diameter (independent variable) is greater than the maximum engine diameter and the required clearances. Thus the engine is able to be installed internally.

Notation

$$\text{func}(5) = \text{ENGMAX} - \frac{(\text{FB}+\text{FH})}{2}$$

3.5 Sustained turn load factor.

This is included so that the vehicle is able to achieve the load factor specified for the sustained turn performance in the manoeuvre phase of the sortie. The specified 'g' load (external variable) must be less than the maximum 'g' load that the vehicle can sustain.

Notation

func(6) = GNS - STRF

3.6 Buffet onset lift coefficient.

The calculated buffet onset lift coefficient is obtained from the configuration geometry in the appropriate aerodynamic module. The constraint ensures that this coefficient should be greater than the lift coefficient calculated by the attained turn performance of the vehicle.

Notation

func(7) = CLT - CLBOT

where CLT = lift coefficient calculated in the turn
CLBOT = buffet onset lift coefficient

3.2 Wing geometry.

This constraint is taken from Ref. 5 and concerns the combinations of values of wing aspect ratio and 1/4 chord sweep angle. This states that for acceptable behaviour for combat aircraft the following condition had to be achieved:

$$AR \times [\tan(QW4R)]^{0.8} < 3.2$$

Notation

$$\text{func}(8) = AR \times (\tan(QW4R))^{0.8} - 3.2$$

3.8 Point performance constraints.

3.8.1 Attained turn.

The constraint states that the synthesized vehicle, at a specific time, given by the fraction of total fuel used, must be able to achieve a specified attained turn 'g' load.

Notation

func(9) = GNAPP - ATRFP1

with ATRFP1 = calculated maximum attained turn 'g' load

3.8.2 Sustained turn.

This constraint is similar to the one above but concerns the sustained turn performance at a specific point.

Notation

func(10) = GNSPP - STRFP1

3.8.3 Maximum Mach number.

This constraint states that the vehicle should be able to achieve a specified maximum Mach number, given as an external variable, at a specific point in the sortie.

Notation

func(11) = REQMM - MAXNUM

with MAXNUM = maximum Mach number achievable in 1g flight, calculated from the appropriate module.

3.8.4 Specific excess power.

This constraint states that the vehicle should be able to achieve a specified SEP value, given as an external variable, at a specific point in the sortie.

Notation

func(12) = REQSEP - SEP

4. Objective function.

The objective function used in the optimisation process was the total mass of the vehicle at take-off, TOTMAS. This function is named func(1).

APPENDIX J.

Implementation and Operation.

This appendix describes the operating procedure of the program and gives some detail on the optimisation package being used.

1. Optimisation package.

The RQPMIN optimisation package was developed at the Royal Aircraft Establishment, Farnborough, to solve optimisation problems subject to a set of user-specified constraints by manipulating the problem variables within simple lower and upper limits. The problem variables and constraints have already been discussed in Appendix I.

In mathematical terms RQPMIN can solve problems which can be stated as:

minimise: $f(x)$

subject to: $f_j(x) = 0 \qquad (j = 1, \ldots, m)$
$f_j(x) \leq 0 \qquad (j = m+1, \ldots, n)$

and $x_j^L \leq x_j \leq x_j^U \qquad (j = 1, \ldots, p)$

where $f(x)$ is the objective function
$f_j(x)$ is the constraint functions
x_j is the problem variables
x_j^L is the variable's lower limit
x_j^U is the variable's upper limit

The package can handle problems with upto 75 constraint equations and upto 50 variables. It can also be used to solve a problem with no constraints but variable bounds must be specified.

2. Input files.

In order to run the UMA program the user must supply four input data files.

2.1 Data file 1.

This input file is read into the program at the start of the synthesis and is called 'AERO.DAT'. It contains the equation coefficients required to represent the graphical data needed in the calculation of the aerodynamic properties of the vehicle. The properties involved are design Mach number; buffet onset and maximum lift coefficients. The data file can be seen in fig. J.2.1.

The file is split into various sections each covering the representation of the data provided in Appendix D.

The first three lines represent Fig D.4.1, drag divergent Mach number. The first line gives four values of design lift coefficient, with the following two lines giving the coefficients of the equations from which the drag divergent Mach number can be obtained.

eg. For CLDES = 0.1

The equation is; $Y = -2.0962 + 0.1698 \times X$

where
$$Y = (MD2-1)$$
$$X = (TC)^{2/3}$$

The next section represents Fig D.5.1, buffet onset lift coefficient, with the first line giving values of $(TC)^{2/3}$ and the next four the equation coefficients of the curves in Fig D.5.1

eg. For $(TC)^{2/3} = 0.1$

The equation is;

$$Y = 0.07198 + 0.3298 \times X + 1.204 \times X^2 + 1.016 \times X^3$$

where
$$Y = \frac{(CLBUF-CLDES)}{AR} \times \cos(QW4R)$$
$$X = DELTAM = SECMAC-MDES$$

The following set of numbers represent Fig D.6.2, with the first line being values of DELTAY, followed again by the equation coefficients of the curves within the figure.

Fig D.6.3 is represented next, the first line giving the Mach numbers followed by the list of coefficients relating to the equations for these Mach numbers.

Fig D.6.4 is represented next with the first two lines giving values of DELTAY.

eg. For DELTAY = 1.4

The equation is;

$$Y = 0.8925 + 0.2641E-2 \times X + 0.3077E-5 \times X^2 + 0.2244E-5 \times X^3 - 0.193E-7 \times X^4$$

where
$$Y = \frac{CLMAXW}{CLMAXS}$$
$$X = QWLD$$

The following section represents the graphs in Fig D.6.5, the first line being values of the leading edge sweep in degrees. The information following is split into four subsections, one for each graph represented. Within these subsections the first line gives values for DELTAY with the following lines values of the equation coefficients relating to each value of DELTAY.

Fig D.6.6 is represented by the last section with the first line giving values of (COEFF2+1)×AR×cos(QWLR) and then the equation coefficients.

This data file, 'AERO.DAT', is read into the program in free format style, with each data point being separated from the others by a space. Commas or any other characters must not be used to separate data points.

2.2 <u>Data file 2.</u>

This input file is also read into the program, in free format style, at the start of the synthesis and is called 'ENGINE.DAT'. This contains the equation coefficients required to represent the graphical data for the engine performance calculations, described in Appendix F.

Fig J.2.2 shows file ENGINE.DAT for the Williams FJ44 engine. the first line contains the four altitudes at which data is provided. The remainder of the file consists of twelve groups of six lines, each containing three columns of numbers. Each group represents the data given in figures F1 to F12. The first four groups represent the thrust information, for the altitudes 0, 5000, 10000 and 20000m respectively, followed by the fuel flow information and finally the air flow.

The first line of each group gives the throttle settings at which the information is to be calculated. Below each throttle setting value the column of numbers represent the equation coefficients for either the thrust, fuel flow or air flow at the given altitude, in terms of Mach number.

eg. Thrust at: altitude = 0.0m (sea-level)
throttle setting = 0.7 (cruise)

Is given by the equation:

$$Y = 1202 - 1846 \times X + 2659 \times X^2 - 3013 \times X^3 + 1854 \times X^4$$

where Y = net thrust (lbF)
X = Mach number

eg. Fuel flow at: altitude = 0.0m (sea-level)
throttle setting = 0.7 (cruise)

Is given by the equation:

$$Y = 518.3 + 511.3 \times X - 153.8 \times X^2 + 615.4 \times X^3 + 384.6 \times X^4$$

where Y = fuel flow (lb/hr)
X = Mach number

eg. Air flow at: altitude = 5000.0m
throttle setting = 0.9

Is given by the equation:

$$Y = 51.95 - 1.213 \times X + 16.23 \times X^2 + 58.75 \times X^3 - 47.92 \times X^4$$

where Y = air flow (lb/hr)
X = Mach number

These equations cover the engine's thrust, fuel flow and air flow performances for the altitudes 0, 5000, 10000 and 20000 metres. Three throttle settings are included; 75%, 90% and 100%; equating to cruise; climb and take-off conditions. The Mach number range which was used to calculate these coefficients was from 0 to 0.8, Mach numbers above this can be used but the accuracy of the results may be impared. The data file can be seen in fig. J.2.2.

2.3 Data file 3.

This file is also read into the program at the start and is named 'OPTIM.DAT'. It contains all the external variables for the program, such as sortie leg definitions; design diving speed; normal acceleration 'g' factor. It also contains the packaging densities and technology factors required. The definitions of these variables can be seen in Appendix I, with the layout of the file seen in fig. J.2.3a and an example in fig. J.2.3b.

2.4 Data file 4.

This file contains gives details of problem variables, (starting values, upper and lower limits); problem functions (markers to identify the objective and constraints) and control records. An example of an input file is given in fig. J.2.4. The name of this file can be user-specified, but will be referred to as 'INPUT.DAT' for the purpose of this report.

2.4.1 Problem variables.

The first record of the input file must contain the keyword VARIABLES. The first two character places of the subsequent definition record must contain the index number of the variable as understood by the problem routine. The fourth character must contain the variable status. This may take the following values:

- 0 variable is to be kept fixed at its starting value
- 1 variable is to be optimised

Character places 6 to 19 contain a scale factor for the variable. This factor must be such that the variable in question is of the order of one throughout the calculation. Places 21 to 34 must contain the unscaled starting value of the variables, with the places 36 to 49 and 51 to 64 containing the unscaled lower and upper bound, respectively.

2.4.2 Problem functions.

The record after the last variable record must be the FUNCTIONS keyword.

The first two character places of the subsequent definition records must contain the index of the function in question and places 4 and 5 contain the function status:

- -1 function is an inequality constraint
- 0 function is to be minimised
- +1 function is an equality constraint

Character places 7 and 8 contain the derivative status of the function (Ref. 10):

- 0 analytical expression for function derivative is not given - use finite differences
- 1 function is linear - derivative is constant and is calculated by subroutine USERD at start of minimisation
- 2 function is non-linear and an analytical expression for the derivative is given

Places 10 to 23 contain a scale factor for the function (this keeps the function value of order one throughout the process).

2.4.3 Control records.

After the last function definition records, the next record must contain the CONTROLS keyword. These subsequent records contain keywords which control the execution of the package. A description of these records can be found in Ref. 10. The only control records that are needed to run the program are RUN and END. The others are used to tune the performance of the program and their default values should produce reasonable results for the problem at hand.

3. RQPMIN progress reports.

Whilst the program is running, progress reports are produced. The frequency of the reports can be controlled by the keywords in input file 4. The reports are sent to a separate file specified in response to the 'Output file ?' prompt (see later) by the following string:

filename.rep/report

The program will then open the report file and again prompt the user for the name of the output file. The progress reports have differing formats for the different stages of the optimisation:

initialisation context;
feasibility step context;
minimisation step context;
changing to central differences context;
final call to user routine context;

An example of a progress report is found in fig. J.3.1 and a detailed description of each of the above stages is given in ref. 10.

4. RQPMIN output file.

The output file consists of five sections which are always printed as well as two optional sections. An example output file is shown in figure J.4.1.

The sections included are:

1. listing of input file
2. analysis of input file data and control parameters
3. starting point of variables and functions
4. starting point of total vehicle configuration and sortie analysis
5. intermediate output
6. final output of variables and functions
7. final output of total vehicle configuration and sortie analysis

4.1 Listing of input file.

This section is optional and is invoked by typing the '/list' option after the input file specification. Any input file errors detected are printed in this section and reference 10 can be consulted as to the possible errors.

4.2 Analysis of input file data and control parameters.

This section is always printed and consists of a list of all variables, functions and control data as understood by the program.

4.3 Starting point of variables and functions.

This section is always printed and lists the starting values of the independent variables and the problem functions.

4.4 Starting point of total vehicle configuration and sortie analysis.

This section is always printed and lists the starting values of the external and dependent variables which make up the vehicle configuration, sortie analysis and, if required, point performance data.

4.5 Intermediate output.

This section is optional and is invoked by specifying positive integer values to the assignment keywords 'OFROM' and 'OFREQ' in input file 4 (Ref.10). The section lists the best values of the variables and problem functions found up to that point.

4.6 Final output of variables and functions.

This section is always printed when the program stops and ends with a listing of any warnings that were reported during the run. This is of similar style to the starting point section.

4.7 Final output of total vehicle configuration and sortie analysis.

This section is always printed when the program stops and is of similar style to the starting point section for the vehicle configuration and sortie analysis.

5. Operation of the program.

In order to run the program the four input data files described earlier (section 2) must be created. When the executable file is run the following procedure can be followed:

1. In response to the command

   'Input file ?'

   Type 'input .dat/list'

'/list' invokes the listing of the input data and is optional.

2. In response to the command

   'Output file ?'

   Type 'output.rep/report'

This invokes the progress reports to be filed and is optional.

3. If 2. is executed then in response to

   'Output file ?'

   Type 'output.res'

Thus calling the output file 'output.res' and the program runs giving progress reports when specified.

6. Example run.

An example run, giving convergence, of the complete synthesis and optimisation program is detailed below and in figures J.2.3b; J.2.4; J.3.1 and J.4.1.

6.1 External variables.

Figure J.2.3b shows the values of the external variables used, the meaning of which are as follows:

LINE 1

These flags are described in Appendix I, here giving a configuration with simple circular cross-section, on payload bay, internal propulsion system and a hand packed recovery system.

LINE 2

These are design data, with:

a. 10 kg - avionics
b. 200 m/s - design diving speed
c. 25 kg - payload in 1st payload bay
d. 0 kg - payload in 2nd payload bay

LINES 3 to 10

This is the sortie leg information:

a. climb from 100m to 3000m at 95% throttle setting and Mach 0.5
b. cruise at 3000m for 50 nautical miles at Mach 0.6 and a throttle setting of 90%
c. loiter for 30 minutes at 3000m at Mach 0.5
d. accelerate from Mach 0.5 to Mach 0.8, at 3000m
e. high speed dash at Mach 0.8, 3000m for 5 minutes
f. manoeuvre: two 360 degree, 3'g' turns at 2000m
   buffet onset lift coefficient not included in output
g. return at 2000m at Mach 0.6
h. reserve fuel to be 10% of total fuel

LINES 11 to 14

Point performance data:

flag set to 0 (point performance not included in optimisation)

LINE 15

Engine clearance information:

a. fraction of engine diameter
b. lower limit on clearance
c. upper limit on clearance

LINE 16

Packaging densities:

a. payload - 640.8 kg/m^3
b. avionics - 640.8 kg/m^3
c. recovery - 320.0 kg/m^3
d. electrics - 480.6 kg/m^3
e. fuel - 800.0 kg/m^3

LINE 17

Manufacturing factors:

TFMEC1 = 1.0
TFMEC2 = 1.0
RCDCK = 1.0

LINE 18

Recovery system constants:

parachute C_{D0} = 0.75
descent velocity = 0.6 m/s

6.2 Independent variables.

Figure J.2.4 shows the input file 'INPUT.DAT' (data file 4) containing the independent variable used in the example and problem function information. The independent variables used were:

			starting value
01	sea-level static thrust	(N)	1200.0
02	wing area	(m^2)	1.4
03	aspect ratio		6.4
04	fuselage length	(m)	3.75
05	fuselage diameter	(m)	0.39
06	1/4 chord sweep	(degs)	5.0
07	thickness chord ratio		0.1
08	taper ratio		0.55

This file, 'INPUT.DAT', is described fully, in section 2.4 of this appendix, and contains the starting values of the independent variables, the scale factor to be used and the lower and upper bounds for each variable.

Below these, the function information is given, showing that function 1 to be the objective and 2, 5, 6 and 8 are to be applied in equality constraints.

The controls at the end of the file are used in the internal working of the optimiser, with 'RFREQ' setting the progress reports to be printed every 10 loops.

6.3 Example progress report and output file.

Figures J.3.1 and J.4.1 show the example progress report and output file, respectively. The progress report shows the values of the independent variables and constraints for each tenth loop. A description of the various steps can be found in reference 10. The output file shows the detailed results of the process as explained earlier, with the initial and final configurations being given.

The table below shows the values of the major parameters at both the starting point and the convergence point of the optimisation.

TABLE 1. Major configuration parameters. (Example run)

VARIABLE	STARTING VALUE	FINAL VALUE
aspect ratio	6.4	5.929202
wing span (m)	2.993326	2.434995
wing area (m^2)	1.4	1.0
taper ratio	0.55	0.502993
thickness/chord	0.1	0.077210
1/4 chord sweep (deg)	5.0	4.543227
fuselage length (m)	3.75	2.998813
fuselage diameter (m)	0.39	0.356870
fuselage volume (m^3)	0.518724	0.349056
thrust (N)		
(sea-level static)	1200.0	1199.74
fuel masses:		
that can be carried (kg)	129.768	50.961
that is required (kg)		
(including reserves)	66.422	50.977
vehicle mass (kg)	197.420	174.484

Figure J.2.1 AERO.DAT (Data file 1)

```
  0.1      0.6      0.7      0.8
 -2.0962 -1.9808 -1.9692 -1.9692
  0.1698  0.0865  0.0605  0.0405

  0.1      0.12     0.14      0.16      0.18     0.2      0.24      0.3
  0.07198  0.04077  0.03885   0.02267   0.01776  0.01444  0.009049 -0.0007716
  0.3298   0.1613   0.02167  -0.07933  -0.141   -0.1758  -0.235    -0.35
  1.204    1.411    0.8389    0.03333  -0.1756  -0.2855  -0.4241   -0.4926
  1.016    2.525    1.741    -0.1333   -0.2933  -0.4167  -0.358     0.4691

  0.0        0.5       1.35
  0.009164  -0.005156  0.002031
  6.495      6.102     5.21
-10.94     -10.45     -8.601
  9.147      8.597     7.08
 -4.327     -3.813    -3.224
  1.099      0.8748    0.769
 -0.1151    -0.08138  -0.0746

  0.2            0.4            0.6
 -1.935         -1.935         -1.932
  2.060          2.039          2.0
 -0.8542        -0.8333        -0.8002
  0.1753         0.1687         0.1587
 -0.01865       -0.01777       -0.01641
  0.9904E-3     -0.9383E-3      0.8518E-3
 -0.2079E-4     -0.1964E-4     -0.1755E-4

  1.4            1.6            1.8           2.0
  2.4            2.5
  0.8925         0.8987         0.9020        0.8994
  0.9027         0.9019
  0.2641E-2      0.2745E-2      0.3373E-2     0.6E-4
  0.1636E-2     -0.476E-2
  0.3077E-5      0.1819E-4     -0.141E-3      0.1333E-5
 -0.2926E-3      0.103E-3
  0.2244E-5     -0.1958E-6      0.375E-5     -0.6E-6
  0.5892E-5     -0.2427E-5
 -0.193E-7       0.8125E-8     -0.2917E-7     0.6667E-8
 -0.4875E-7      0.4295E-8

  0.0 20.0 40.0 60.0

  2.0   2.25   2.5     3.0    4.0    4.5
  0.11  0.6065 0.5404  0.7552 0.6926 0.4946
 -0.77 -4.477 -3.376  -5.183 -4.287 -2.484
  1.1   8.15   3.462   7.95   4.537 -0.3625
  0.0  -4.833 -0.9167 -4.667 -1.75   1.917

  2.0    2.25          2.5    3.0     4.0     4.5
  0.2083 0.2227        0.4802 0.2227  0.3303  0.2467
 -1.553 -1.29         -3.507 -0.9992 -1.836  -1.003
  2.718  0.85          6.262 -1.038   0.8875 -1.5
 -1.05   0.4996E-5    -4.25   1.917   0.08333 1.833

  0.0 0.0  2.0   3.0    4.0    4.5
  0.0 0.0  0.05  0.0915 0.1335 0.15
  0.0 0.0 -0.25 -0.458 -0.668 -0.75
  0.0 0.0  0.0   0.0    0.0    0.0
  0.0 0.0  0.0   0.0    0.0    0.0

  0.0  2.0  2.25  3.0   4.0   4.5
```

0.0	0.1	0.308	0.1445	0.134	0.1372
0.0	-0.5	-0.55	-0.289	-0.335	-0.392
0.0	0.0	0.0	0.0	0.0	0.0
0.0	0.0	0.0	0.0	0.0	0.0

1.0	2.0	4.0	6.0	12.0
1.003	1.015	1.017	0.9974	0.9979
-0.5516	-0.1394	-0.4408	-1.78	-1.582
-0.5228	-0.8129	-0.1771	7.675	8.796
0.48	0.9625	0.5625	-5.5	-7.071

Figure J.2.2   ENGINE.DAT (Data file 2)

```
    0.0 5000.0 10000.0 20000.0

    0.7     0.9     0.95
 1202.0  1584.0  1696.0
-1846.0 -2075.0 -2283.0
 2659.0  2554.0  3538.0
-3013.0 -2242.0 -4250.0
 1854.0  1125.0  2500.0

    0.7       0.9       0.95
 1102.0  1429.0  1523.0
-1668.0 -1745.0 -1722.0
 2631.0  1819.0  1728.0
-3208.0 -1208.0 -1396.0
 1875.0   625.0   937.5

    0.7       0.9       0.95
 1002.0  1288.0  1392.0
-1411.0 -1477.0 -1668.0
 1830.0  1330.0  2029.0
-1467.0  -433.3 -1750.0
  500.0     0.0   833.3

    0.7       0.9       0.95
  798.2  1049.0  1119.0
 -944.4 -1436.0 -1497.0
  837.5  2195.0  2349.0
 -250.0 -1897.0 -2292.0
    0.0   673.1  1042.0

    0.7     0.9     0.95
  518.3  699.2  754.4
  511.3 -16.58   42.63
 -153.8  319.4 -165.7
  615.4 -457.7  929.5
 -384.6  442.3 -737.2

    0.7     0.9     0.95
  469.7  628.7  680.5
   -4.846 28.67 -24.75
  209.5  -85.0  265.4
 -284.5  633.3  -75.0
  282.1 -500.0  -41.67

    0.7       0.9       0.95
  422.7   561.6   609.9
    8.47   24.33    0.4667
   77.6     6.667 117.7
   21.72  266.7     53.33
   17.68 -166.7   -16.67

    0.7     0.9     0.95
  317.2  423.3  467.2
  177.8  225.1  123.8
 -534.1 -825.6 -418.0
  883.3 1550.0  883.3
 -437.5 -875.0 -479.2

    0.7     0.9     0.95
   51.98  59.05  60.87
   -6.225 -8.012 -3.485
   70.12  77.48  50.95
```

-52.5	-73.75	-28.46
37.5	52.08	28.21

0.7	0.9	0.95
45.97	51.95	53.6
-4.992	-1.213	-9.173
59.62	16.23	83.62
-50.83	58.75	-121.2
37.5	-47.92	99.36

0.7	0.9	0.95
40.13	45.54	47.02
-2.544	-12.9	-4.292
41.03	61.45	37.69
-25.61	-3.667	-18.46
19.95	-30.0	15.38

0.7	0.9	0.95
29.47	35.17	37.47
8.017	-16.44	-32.56
-19.35	68.72	135.2
73.33	-65.0	-175.0
-45.83	31.25	93.75

Figure J.2.3a.

External Variables (Data file 3)

FLAGFUS FLAGENG FLAGRS FLAGPAY

AVIM AF VDEAS PAY1 PAY2

ALTI ALTF CLMAC CLRAT ECLFBM

CRALT CRMAC CRRAT CRRAN ECRFBM

LOALT LOMAC LOTIM ELOFBM

ACCALT ACCMI ACCMF EACCFBM

DAALT DAMAC DATIM EDAFBM

MANALT MANMAC NTURN GNS IBUFF EMANFBM

RETALT RETMAC ERETFBM

RESFF

FLAGPP

ALTSPP SECSPP RATSPP FFST GNSPP

ALTAPP SECAPP RATAPP FFAT GNAPP

ALTMAX FFMM REQMM

ALTSEP SECSEP FFSEP REQSEP

FECLR ECLRM1 ECLRM2

PAYDEN AVIDEN RSPSD ELECDEN FTDEN

RCDCK TFMEC1 TFMEC2

CDØP VRD

Figure J.2.3b.

Data file 3: (OPTIM.DAT) Example.

```
1  0  1  0

10.0  10.0  200.0  25.0  0.0

100.0  3000.0  0.5  0.95  0.5

3000.0  0.6  0.9  50.0  10.0

3000.0  0.5  30.0  30.0

3000.0  0.5  0.8  0.5

3000.0  0.8  5.0  15.0

2000.0  0.5  2.0  3.0  0  1.0

2000.0  0.6  15.0

0.1

0

0.0  0.0  0.0  0.0  0.0

0.0  0.0  0.0  0.0  0.0

0.0  0.0  0.0

0.0  0.0  0.0  0.0

0.1  0.035  0.05

640.8  640.8  320.0  480.6  800.0

1.0  1.0  1.0

0.75  0.6
```

FIG. J.2.4. INPUT DATA (DATA FILE 4)

```
    * data for uma problem

    VARIABLES
    01 1  1.0           1200.0          1000.0          2000.0
    02 1  1.0           1.4             1.0             3.0
    03 1  1.0           6.4             3.0             9.0
    04 1  1.0           3.75            2.0             6.0
    05 1  0.1           0.39            0.2             0.5
    06 1  1.0           5.0             0.0             45.0
    07 1  0.1           0.1             0.06            0.2
    08 1  0.1           0.55            0.0             1.0

    FUNCTIONS
    01  0 00  100.0
    02 -1 00  10.0
    05 -1 00  0.1
    06 -1 00  1.0
    08 -1 00  1.0

CONTROLS
RFREQ = 10
BFSTOL = 10.0
XTOLU = 0.001
XTOLV = 0.001
RUN
END
```

FIG. J.3.1. PROGRESS REPORT FILE

```
   nfe =      1   context:  initial call to user routine
   fdatum =  0.1974196E+01 gdatum =  0.4011459E+02

   nfe =     11   context:  minimization step
   fstart =  0.1974196E+01     gstart =  0.4011459E+02
   pstart = -0.4642928E+00     lstart = -0.4642928E+00
   nu     =  0.0000000E+00     numax  =    infinite
   unrmxs =  0.2178122E-01     rtol   =  0.1000000E+00
   vnormx =  0.2156601E+02     vscale =  0.1000000E+01

   nfe =     21   context:  minimization step
   fstart =  0.1974196E+01     gstart =  0.4011459E+02
   pstart = -0.4642928E+00     lstart = -0.4642928E+00
   nu     =  0.0000000E+00     numax  =    infinite
   unrmxs =  0.2178122E-01     rtol   =  0.1000000E+00
   vnormx =  0.2156601E+02     vscale =  0.1000000E+01

   nfe =     31   context:  minimization step
   fstart =  0.1974196E+01     gstart =  0.4011459E+02
   pstart = -0.4642928E+00     lstart = -0.4642928E+00
   nu     =  0.0000000E+00     numax  =    infinite
   unrmxs =  0.2178122E-01     rtol   =  0.1000000E+00
   vnormx =  0.2156601E+02     vscale =  0.1000000E+01

nfe =     41   context:  feasibility step
fdatum =  0.1974196E+01 gdatum =  0.4011459E+02
unormx =  0.2178122E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =     51   context:  feasibility step
fdatum =  0.1979511E+01 gdatum =  0.2707928E+02
unormx =  0.2178122E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =     61   context:  feasibility step
fdatum =  0.1979511E+01 gdatum =  0.2707928E+02
unormx =  0.2741946E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =     71   context:  feasibility step
fdatum =  0.1920538E+01 gdatum =  0.1670554E+02
unormx =  0.2741946E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =     81   context:  feasibility step
fdatum =  0.1920538E+01 gdatum =  0.1670554E+02
unormx =  0.2250582E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =     91   context:  feasibility step
fdatum =  0.1863666E+01 gdatum =  0.9189264E+01
unormx =  0.2250582E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =    101   context:  feasibility step
fdatum =  0.1863666E+01 gdatum =  0.9189264E+01
unormx =  0.1727865E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =    111   context:  feasibility step
fdatum =  0.1808847E+01 gdatum =  0.4144701E+01
unormx =  0.1727865E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =    121   context:  feasibility step
fdatum =  0.1808847E+01 gdatum =  0.4144701E+01
unormx =  0.1202228E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =    131   context:  feasibility step
fdatum =  0.1792240E+01 gdatum =  0.3808478E+01
```

```
unormx =  0.1202228E-01 uscale =  0.1627077E+00 rtol   =  0.1000000E-01

nfe =   141   context:  feasibility step
fdatum =  0.1792240E+01 gdatum =  0.3808478E+01
unormx =  0.7232637E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   151   context:  feasibility step
fdatum =  0.1792240E+01 gdatum =  0.3808478E+01
unormx =  0.7232637E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   161   context:  feasibility step
fdatum =  0.1789221E+01 gdatum =  0.3708393E+01
unormx =  0.4219544E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   171   context:  feasibility step
fdatum =  0.1789221E+01 gdatum =  0.3708393E+01
unormx =  0.4219544E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   181   context:  feasibility step
fdatum =  0.1789221E+01 gdatum =  0.3708393E+01
unormx =  0.4219544E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   191   context:  feasibility step
fdatum =  0.1757125E+01 gdatum =  0.1042505E+00
unormx =  0.5331941E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   201   context:  feasibility step
fdatum =  0.1747497E+01 gdatum =  0.1034237E+00
unormx =  0.1710703E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   211   context:  feasibility step
fdatum =  0.1774597E+01 gdatum =  0.5908370E-01
unormx =  0.2727981E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   221   context:  feasibility step
fdatum =  0.1774597E+01 gdatum =  0.5908370E-01
unormx =  0.1356967E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   231   context:  feasibility step
fdatum =  0.1780410E+01 gdatum =  0.5019030E-01
unormx =  0.1481499E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   241   context:  feasibility step
fdatum =  0.1783622E+01 gdatum =  0.2248653E-02
unormx =  0.1473954E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   251   context:  feasibility step
fdatum =  0.1783622E+01 gdatum =  0.2545705E+00
unormx =  0.4741210E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   261   context:  feasibility step
fdatum =  0.1785777E+01 gdatum =  0.2502672E+00
unormx =  0.4741210E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   271   context:  feasibility step
fdatum =  0.1773504E+01 gdatum =  0.7996560E-02
unormx =  0.1085957E+00 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   281   context:  feasibility step
fdatum =  0.1775730E+01 gdatum =  0.7825792E-02
unormx =  0.8665220E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   291   context:  feasibility step
fdatum =  0.1774667E+01 gdatum =  0.7403573E-02
unormx =  0.8407897E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01
```

```
nfe =    301   context:  feasibility step
fdatum =  0.1782394E+01 gdatum =  0.1958731E-03
unormx =  0.8599017E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    311   context:  feasibility step
fdatum =  0.1784593E+01 gdatum =  0.2424989E+00
unormx =  0.1823670E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    321   context:  feasibility step
fdatum =  0.1777014E+01 gdatum =  0.1804448E-04
unormx =  0.7835994E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    331   context:  minimization step
fstart =  0.1777014E+01     gstart =  0.1804448E-04
pstart =  0.1777925E+01     lstart =  0.1777925E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.2609081E-03     rtol   =  0.1000000E-01
vnormx =  0.4427241E+01     vscale =  0.1000000E+01

nfe =    341   context:  feasibility step
fdatum =  0.1777014E+01 gdatum =  0.6840676E-02
unormx =  0.1414724E+00 uscale =  0.9352694E+00 rtol    =  0.1000000E-01

nfe =    351   context:  feasibility step
fdatum =  0.1777014E+01 gdatum =  0.6840676E-02
unormx =  0.1414724E+00 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    361   context:  feasibility step
fdatum =  0.1779535E+01 gdatum =  0.3515339E-03
unormx =  0.8123719E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    371   context:  feasibility step
fdatum =  0.1779535E+01 gdatum =  0.2143618E+00
unormx =  0.1714180E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    381   context:  feasibility step
fdatum =  0.1781732E+01 gdatum =  0.2135865E+00
unormx =  0.1714180E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    391   context:  minimization step
fstart =  0.1774187E+01     gstart =  0.8941128E-03
pstart =  0.1771384E+01     lstart =  0.1771384E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.8049287E-03     rtol   =  0.1000000E-01
vnormx =  0.2708797E+01     vscale =  0.1000000E+01

nfe =    401   context:  minimization step
fstart =  0.1774187E+01     gstart =  0.8941128E-03
pstart =  0.1771384E+01     lstart =  0.1771384E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.8049287E-03     rtol   =  0.1000000E-01
vnormx =  0.2708797E+01     vscale =  0.1000000E+01

nfe =    411   context:  feasibility step
fdatum =  0.1779677E+01 gdatum =  0.4807698E-02
unormx =  0.9399574E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    421   context:  feasibility step
fdatum =  0.1779677E+01 gdatum =  0.4807698E-02
unormx =  0.8363417E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    431   context:  feasibility step
fdatum =  0.1779677E+01 gdatum =  0.4807698E-02
unormx =  0.8363417E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =    441   context:  feasibility step
```

```
fdatum =  0.1775684E+01 gdatum =  0.4069999E-02
unormx =  0.8363417E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   451   context:  feasibility step
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02
unormx =  0.6378009E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   461   context:  feasibility step
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02
unormx =  0.6534993E-01 uscale =  0.9700275E+00 rtol    =  0.1000000E-01

nfe =   471   context:  feasibility step
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02
unormx =  0.6534993E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   475   context:  changing to central differences
                        (after failed feasibility step)
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02 rtol =  0.1000000E-01

nfe =   481   context:  changing to central differences
                        (after failed feasibility step)
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02 rtol =  0.1000000E-01

nfe =   491   context:  changing to central differences
                        (after failed feasibility step)
fdatum =  0.1779419E+01 gdatum =  0.3389390E-02 rtol =  0.1000000E-01

nfe =   501   context:  feasibility step
fdatum =  0.1776365E+01 gdatum =  0.3128727E-02
unormx =  0.5379762E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   511   context:  feasibility step
fdatum =  0.1776365E+01 gdatum =  0.3128727E-02
unormx =  0.5574904E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   521   context:  feasibility step
fdatum =  0.1775610E+01 gdatum =  0.1581158E-02
unormx =  0.5574904E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   531   context:  feasibility step
fdatum =  0.1775610E+01 gdatum =  0.1581158E-02
unormx =  0.3428286E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   541   context:  feasibility step
fdatum =  0.1776763E+01 gdatum =  0.1119651E-02
unormx =  0.3428286E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   551   context:  feasibility step
fdatum =  0.1775757E+01 gdatum =  0.1011031E-02
unormx =  0.3075380E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   561   context:  feasibility step
fdatum =  0.1775757E+01 gdatum =  0.1011031E-02
unormx =  0.3075380E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   571   context:  feasibility step
fdatum =  0.1776438E+01 gdatum =  0.9364336E-03
unormx =  0.3179196E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   581   context:  feasibility step
fdatum =  0.1776438E+01 gdatum =  0.9364336E-03
unormx =  0.3179196E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01

nfe =   591   context:  feasibility step
fdatum =  0.1775728E+01 gdatum =  0.8742173E-03
unormx =  0.2891267E-01 uscale =  0.1000000E+01 rtol    =  0.1000000E-01
```

```
nfe =   601   context:  feasibility step
fdatum =  0.1775728E+01 gdatum =  0.8742173E-03
unormx =  0.2891267E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   611   context:  feasibility step
fdatum =  0.1776383E+01 gdatum =  0.8102162E-03
unormx =  0.2956483E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   621   context:  feasibility step
fdatum =  0.1776383E+01 gdatum =  0.8102162E-03
unormx =  0.2956483E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   631   context:  feasibility step
fdatum =  0.1775691E+01 gdatum =  0.7506196E-03
unormx =  0.2671774E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   641   context:  feasibility step
fdatum =  0.1775691E+01 gdatum =  0.7506196E-03
unormx =  0.2671774E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   651   context:  feasibility step
fdatum =  0.1776330E+01 gdatum =  0.6926201E-03
unormx =  0.2738617E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   661   context:  feasibility step
fdatum =  0.1776330E+01 gdatum =  0.6926201E-03
unormx =  0.2738617E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   671   context:  feasibility step
fdatum =  0.1775664E+01 gdatum =  0.6419894E-03
unormx =  0.2471505E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   681   context:  feasibility step
fdatum =  0.1775664E+01 gdatum =  0.6419894E-03
unormx =  0.2533860E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   691   context:  feasibility step
fdatum =  0.1777400E+01 gdatum =  0.6208724E-03
unormx =  0.2533860E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   701   context:  feasibility step
fdatum =  0.1777400E+01 gdatum =  0.6208724E-03
unormx =  0.1992968E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   711   context:  feasibility step
fdatum =  0.1775634E+01 gdatum =  0.4679601E-03
unormx =  0.1992968E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   721   context:  feasibility step
fdatum =  0.1776508E+01 gdatum =  0.3427783E-03
unormx =  0.2161383E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   731   context:  feasibility step
fdatum =  0.1776508E+01 gdatum =  0.3427783E-03
unormx =  0.2161383E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   741   context:  feasibility step
fdatum =  0.1775797E+01 gdatum =  0.3299022E-03
unormx =  0.1808701E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   751   context:  feasibility step
fdatum =  0.1775797E+01 gdatum =  0.3299022E-03
unormx =  0.1808701E-01 uscale =  0.1000000E+01 rtol    =   0.1000000E-01

nfe =   761   context:  feasibility step
```

```
fdatum =  0.1776790E+01 gdatum =  0.3195815E-03
unormx =  0.1815852E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   771   context:  feasibility step
fdatum =  0.1776790E+01 gdatum =  0.3195815E-03
unormx =  0.1815852E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   781   context:  feasibility step
fdatum =  0.1775540E+01 gdatum =  0.2453064E-03
unormx =  0.1454128E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   791   context:  feasibility step
fdatum =  0.1775540E+01 gdatum =  0.2453064E-03
unormx =  0.1454128E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   801   context:  feasibility step
fdatum =  0.1776081E+01 gdatum =  0.1484316E-03
unormx =  0.1570696E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   811   context:  feasibility step
fdatum =  0.1776081E+01 gdatum =  0.1484316E-03
unormx =  0.1208820E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   821   context:  feasibility step
fdatum =  0.1775508E+01 gdatum =  0.1264375E-03
unormx =  0.1208820E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   831   context:  feasibility step
fdatum =  0.1775971E+01 gdatum =  0.1056686E-03
unormx =  0.1111863E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   841   context:  feasibility step
fdatum =  0.1775971E+01 gdatum =  0.1056686E-03
unormx =  0.1111863E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   851   context:  feasibility step
fdatum =  0.1775444E+01 gdatum =  0.6010631E-04
unormx =  0.1024173E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   861   context:  feasibility step
fdatum =  0.1775444E+01 gdatum =  0.6010631E-04
unormx =  0.1024173E-01 uscale =  0.1000000E+01 rtol   =  0.1000000E-01

nfe =   871   context:  minimization step
fstart =  0.1775444E+01     gstart =  0.6010631E-04
pstart =  0.1775588E+01     lstart =  0.1775588E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.2251606E-02     rtol   =  0.1000000E-01
vnormx =  0.2532992E+00     vscale =  0.8568215E+00

nfe =   881   context:  minimization step
fstart =  0.1750189E+01     gstart =  0.4108106E-03
pstart =  0.1749874E+01     lstart =  0.1749874E+01
nu     =  0.0000000E+00     numax  =  0.7331395E+02
unrmxs =  0.2992225E-02     rtol   =  0.1000000E-01
vnormx =  0.4976464E-01     vscale =  0.1000000E+01

nfe =   891   context:  minimization step
fstart =  0.1750189E+01     gstart =  0.4108106E-03
pstart =  0.1749874E+01     lstart =  0.1749874E+01
nu     =  0.0000000E+00     numax  =  0.7331395E+02
unrmxs =  0.2992225E-02     rtol   =  0.1000000E-01
vnormx =  0.4976464E-01     vscale =  0.1000000E+01

nfe =   901   context:  minimization step
fstart =  0.1744845E+01     gstart =  0.6632095E-03
```

```
pstart =  0.1744591E+01      lstart =  0.1744591E+01
nu     =  0.0000000E+00      numax  =  0.2151468E+02
unrmxs =  0.5363435E-02      rtol   =  0.1000000E-01
vnormx =  0.5601362E-01      vscale =  0.1000000E+01

nfe =    911   context:  minimization step
fstart =  0.1744845E+01      gstart =  0.6632095E-03
pstart =  0.1744591E+01      lstart =  0.1744591E+01
nu     =  0.0000000E+00      numax  =  0.2151468E+02
unrmxs =  0.5363435E-02      rtol   =  0.1000000E-01
vnormx =  0.5601362E-01      vscale =  0.1000000E+01

nfe =    921   context:  minimization step
fstart =  0.1743299E+01      gstart =  0.5532315E-03
pstart =  0.1743527E+01      lstart =  0.1743527E+01
nu     =  0.0000000E+00      numax  =  0.2151468E+02
unrmxs =  0.5131797E-02      rtol   =  0.1000000E-01
vnormx =  0.6095141E-01      vscale =  0.1000000E+01

nfe =    931   context:  minimization step
fstart =  0.1743299E+01      gstart =  0.5532315E-03
pstart =  0.1743527E+01      lstart =  0.1743527E+01
nu     =  0.0000000E+00      numax  =  0.2151468E+02
unrmxs =  0.5131797E-02      rtol   =  0.1000000E-01
vnormx =  0.6095141E-01      vscale =  0.1000000E+01

nfe =    941   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
vnormx =  0.7518478E-01      vscale =  0.1000000E+01

nfe =    951   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
vnormx =  0.7518478E-01      vscale =  0.1000000E+01

nfe =    961   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
vnormx =  0.7518478E-01      vscale =  0.1000000E+01

nfe =    971   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
vnormx =  0.7518478E-01      vscale =  0.1000000E+01

nfe =    981   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
vnormx =  0.7518478E-01      vscale =  0.1000000E+01

nfe =    991   context:  minimization step
fstart =  0.1742876E+01      gstart =  0.1089354E-02
pstart =  0.1743169E+01      lstart =  0.1743169E+01
nu     =  0.0000000E+00      numax  =  0.6660532E+00
unrmxs =  0.7604960E-02      rtol   =  0.1000000E-01
```

```
vnormx =  0.7518478E-01     vscale =  0.1000000E+01

nfe =  1001   context:  feasibility step
fdatum =  0.1742876E+01 gdatum =  0.1089354E-02
unormx =  0.7604960E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1011   context:  feasibility step
fdatum =  0.1746071E+01 gdatum =  0.2238109E-03
unormx =  0.7604960E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1021   context:  feasibility step
fdatum =  0.1746071E+01 gdatum =  0.2238109E-03
unormx =  0.3820512E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1031   context:  feasibility step
fdatum =  0.1745039E+01 gdatum =  0.1338516E-03
unormx =  0.3820512E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1041   context:  feasibility step
fdatum =  0.1742646E+01 gdatum =  0.9604336E-04
unormx =  0.2791401E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1051   context:  feasibility step
fdatum =  0.1742646E+01 gdatum =  0.9604336E-04
unormx =  0.2791401E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1061   context:  feasibility step
fdatum =  0.1746136E+01 gdatum =  0.5812387E-06
unormx =  0.2211462E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1071   context:  feasibility step
fdatum =  0.1746136E+01 gdatum =  0.5812387E-06
unormx =  0.2211462E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1081   context:  minimization step
fstart =  0.1746136E+01     gstart =  0.5812387E-06
pstart =  0.1746120E+01     lstart =  0.1746120E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.1933031E-03     rtol   =  0.1000000E-02
vnormx =  0.7526059E-01     vscale =  0.1000000E+01

nfe =  1091   context:  minimization step
fstart =  0.1746136E+01     gstart =  0.5812387E-06
pstart =  0.1746120E+01     lstart =  0.1746120E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.1933031E-03     rtol   =  0.1000000E-02
vnormx =  0.7526059E-01     vscale =  0.1000000E+01

nfe =  1101   context:  minimization step
fstart =  0.1746136E+01     gstart =  0.5812387E-06
pstart =  0.1746120E+01     lstart =  0.1746120E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.1933031E-03     rtol   =  0.1000000E-02
vnormx =  0.7526059E-01     vscale =  0.1000000E+01

nfe =  1111   context:  minimization step
fstart =  0.1746136E+01     gstart =  0.5812387E-06
pstart =  0.1746120E+01     lstart =  0.1746120E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.1933031E-03     rtol   =  0.1000000E-02
vnormx =  0.7526059E-01     vscale =  0.1000000E+01

nfe =  1121   context:  minimization step
fstart =  0.1746136E+01     gstart =  0.5812387E-06
pstart =  0.1746120E+01     lstart =  0.1746120E+01
nu     =  0.0000000E+00     numax  =    infinite
```

```
unrmxs =  0.1933031E-03      rtol   =  0.1000000E-02
vnormx =  0.7526059E-01      vscale =  0.1000000E+01

nfe =  1131   context:  minimization step
fstart =  0.1744662E+01      gstart =  0.3630165E-06
pstart =  0.1744697E+01      lstart =  0.1744697E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.7013500E-03      rtol   =  0.1000000E-02
vnormx =  0.6266280E-01      vscale =  0.1000000E+01

nfe =  1141   context:  minimization step
fstart =  0.1744662E+01      gstart =  0.3630165E-06
pstart =  0.1744697E+01      lstart =  0.1744697E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.7013500E-03      rtol   =  0.1000000E-02
vnormx =  0.6266280E-01      vscale =  0.1000000E+01

nfe =  1151   context:  minimization step
fstart =  0.1744662E+01      gstart =  0.3630165E-06
pstart =  0.1744697E+01      lstart =  0.1744697E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.7013500E-03      rtol   =  0.1000000E-02
vnormx =  0.6266280E-01      vscale =  0.1000000E+01

nfe =  1161   context:  minimization step
fstart =  0.1744662E+01      gstart =  0.0000000E+00
pstart =  0.1744662E+01      lstart =  0.1744662E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1094341E+00      vscale =  0.1000000E+01

nfe =  1171   context:  minimization step
fstart =  0.1744662E+01      gstart =  0.0000000E+00
pstart =  0.1744662E+01      lstart =  0.1744662E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1094341E+00      vscale =  0.1000000E+01

nfe =  1181   context:  minimization step
fstart =  0.1744251E+01      gstart =  0.0000000E+00
pstart =  0.1744251E+01      lstart =  0.1744251E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1095533E+00      vscale =  0.1000000E+01

nfe =  1191   context:  minimization step
fstart =  0.1744251E+01      gstart =  0.0000000E+00
pstart =  0.1744251E+01      lstart =  0.1744251E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1095533E+00      vscale =  0.1000000E+01

nfe =  1201   context:  minimization step
fstart =  0.1744135E+01      gstart =  0.0000000E+00
pstart =  0.1744135E+01      lstart =  0.1744135E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1095533E+00      vscale =  0.1000000E+01

nfe =  1211   context:  minimization step
fstart =  0.1744135E+01      gstart =  0.0000000E+00
pstart =  0.1744135E+01      lstart =  0.1744135E+01
nu     =  0.0000000E+00      numax  =    infinite
unrmxs =  0.0000000E+00      rtol   =  0.1000000E-02
vnormx =  0.1095533E+00      vscale =  0.1000000E+01
```

```
nfe =  1221   context:  feasibility step
fdatum =  0.1744493E+01 gdatum =  0.6238485E-05
unormx =  0.1894773E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1231   context:  feasibility step
fdatum =  0.1744493E+01 gdatum =  0.6238485E-05
unormx =  0.1894773E-02 uscale =  0.1000000E+01 rtol   =  0.1000000E-02

nfe =  1241   context:  minimization step
fstart =  0.1744493E+01     gstart =  0.6238485E-05
pstart =  0.1744524E+01     lstart =  0.1744524E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.5947461E-03     rtol   =  0.1000000E-02
vnormx =  0.2788448E-01     vscale =  0.1000000E+01

nfe =  1251   context:  minimization step
fstart =  0.1744493E+01     gstart =  0.6238485E-05
pstart =  0.1744524E+01     lstart =  0.1744524E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.5947461E-03     rtol   =  0.1000000E-02
vnormx =  0.2788448E-01     vscale =  0.1000000E+01

nfe =  1261   context:  minimization step
fstart =  0.1744493E+01     gstart =  0.6238485E-05
pstart =  0.1744524E+01     lstart =  0.1744524E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.5947461E-03     rtol   =  0.1000000E-02
vnormx =  0.2788448E-01     vscale =  0.1000000E+01

nfe =  1271   context:  minimization step
fstart =  0.1744493E+01     gstart =  0.6238485E-05
pstart =  0.1744524E+01     lstart =  0.1744524E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.5947461E-03     rtol   =  0.1000000E-02
vnormx =  0.2788448E-01     vscale =  0.1000000E+01

nfe =  1281   context:  minimization step
fstart =  0.1744493E+01     gstart =  0.6238485E-05
pstart =  0.1744524E+01     lstart =  0.1744524E+01
nu     =  0.0000000E+00     numax  =    infinite
unrmxs =  0.5947461E-03     rtol   =  0.1000000E-02
vnormx =  0.2788448E-01     vscale =  0.1000000E+01

nfe =  1291   context:  minimization step
fstart =  0.1744810E+01     gstart =  0.9750373E-07
pstart =  0.1744820E+01     lstart =  0.1744815E+01
nu     =  0.4742487E+02     numax  =    infinite
unrmxs =  0.9208686E-04     rtol   =  0.1000000E-02
vnormx =  0.5908068E-01     vscale =  0.1000000E+01

nfe =  1301   context:  minimization step
fstart =  0.1744810E+01     gstart =  0.9750373E-07
pstart =  0.1744820E+01     lstart =  0.1744815E+01
nu     =  0.4742487E+02     numax  =    infinite
unrmxs =  0.9208686E-04     rtol   =  0.1000000E-02
vnormx =  0.5908068E-01     vscale =  0.1000000E+01

nfe =  1311   context:  minimization step
fstart =  0.1744810E+01     gstart =  0.9750373E-07
pstart =  0.1744820E+01     lstart =  0.1744815E+01
nu     =  0.4742487E+02     numax  =    infinite
unrmxs =  0.9208686E-04     rtol   =  0.1000000E-02
vnormx =  0.5908068E-01     vscale =  0.1000000E+01
```

FIG. J.4.1. OUTPUT FILE

Program RQPMIN Version 1.0 for Multics

ANALYSIS OF INPUT FILE DATA AND CONTROL PARAMETERS

Input file: test.dat
Output file: test.out

Variable data

Number of variables = 8

Index	Status	Scale	Starting value	Lower bound	Upper bound
1	1	0.1000000E+01	0.1200000E+04	0.1000000E+04	0.2000000E+04
2	1	0.1000000E+01	0.1400000E+01	0.1000000E+01	0.3000000E+01
3	1	0.1000000E+01	0.6400000E+01	0.3000000E+01	0.9000000E+01
4	1	0.1000000E+01	0.3750000E+01	0.2000000E+01	0.6000000E+01
5	1	0.1000000E+00	0.3900000E+01	0.2000000E+01	0.5000000E+01
6	1	0.1000000E+01	0.5000000E+01	0.0000000E+00	0.4500000E+02
7	1	0.1000000E+00	0.1000000E+01	0.6000000E+00	0.2000000E+01
8	1	0.1000000E+00	0.5500000E+01	0.0000000E+00	0.1000000E+02

Problem function data

Number of constraints = 4

Index	Status	Type	Scale	Index	Status	Type	Scale
1	0	0	0.1000000E+03	2	-1	0	0.1000000E+02
5	-1	0	0.1000000E+00	6	-1	0	0.1000000E+01
8	-1	0	0.1000000E+01				

objective is function 1 problem number = 0

```
Control parameters
------------------

nfemax =  4000     nimax  =  1000     nsmax  =    20
nsetc  =     4     nsetcf =     8     nsetv  =     4     nsetvf =     8
ofreq  =     0     ofrom  =     0     rfreq  =    10     rfrom  =     1

centrl = F     fdset  = F     norep  = F     cheats = T     fast   = T     quasi  = T
fixrp  = F     timid  = F     monitr = F     nofrec = F     yesbc  = F     shrnk  = T     projct = T

xtol   =  0.1000000E-05    xtolu  =  0.1000000E-02    xtolv  =  0.1000000E-02    gtol   =  0.1000000E-02
rtol   =  0.1000000E+00    omegar =  0.1000000E+00    rpmax  =  0.2000000E+00
umin   =  0.1000000E-05    vminf  =  0.1000000E-02    vminc  =  0.1000000E-05    ctol   =  0.1000000E-02
umax   =  0.1000000E+00    vmax   =  0.1000000E+00    omega  =  0.1000000E+00    mu     =  0.1000000E-03
bdtol  =  0.1000000E+00    lmtol  =  0.1000000E+00    mtol   =  0.5000000E+00    cmax   =  0.1000000E-01
subtol =  0.1000000E-02    qrtol  =  0.1000000E+02    bfstol =  0.1000000E+02    diftol =  0.5000000E-03
ztol   =  0.1000000E-29

End of input data
-----------------
```

```
Program  RQPMIN  Version 1.0  for Multics

STARTING POINT

free variables
--------------

 1  0.1200000E+04   2  0.1400000E+01   3  0.6400000E+01   4  0.3750000E+01   5  0.3900000E+01   6  0.5000000E+01   7  0.10000(
 8  0.5500000E+01

objective function
------------------

f(x) =  0.1974196E+01

active inequality constraints
-----------------------------

 2  0.6333608E+01

inactive inequality constraints
-------------------------------

 5 -0.4665589E+00  6 -0.6159657E+01  8 -0.2288532E+01

initial vehicle configuration data
----------------------------------

gross wing geometry.
--------------------
                    wing span        = 2.993326 m
                    wing area        = 1.400000 m^2
                    aspect ratio     = 6.400000
                    c.l. chord       = 0.603493 m
                    geo. mean chord  = 0.467707 m
```

```
aero. mean chord = 0.480848 m
tip chord        = 0.331921 m
taper ratio      = 0.550000 m
thick/ch ratio   = 0.100000 m
1/4 chord sweep  = 5.000000 (degs)
l.e. sweep       = 7.567520 (degs)
1/2 chord sweep  = 2.412203 (degs)
t.e. sweep (fwd) = 2.782387 (degs)
```

nett wing geometry.

```
wing span        = 2.603326 m
wing area        = 1.171537 m^2
root chord       = 0.568110 m
taper ratio      = 0.584255 m
mean chord       = 0.450016 m
aspect ratio     = 5.784968 m
```

fuselage geometry.

```
length          = 3.750000 m
width           = 0.390000 m
height          = 0.390000 m
surface area    = 3.744000 m^2
x-section area  = 0.119459 m^2
volume          = 0.518724 m^3
skin thickness  = 1.179757 mm
eng. diam + tol. = 0.343344 m
```

```
engine thrust (s/l)  =  1200.00 n
design div.speed     =   200.00 (m/S)
```

---

a/c section	mass (kg)	zero lift drag
wing	10.98	0.00981
fuselage	15.29	0.00907
tail	2.75	-------
engine	31.86	0.00282
eng. inst.	41.41	-------

fly.cont.	8.82	-------
fuel sys.	4.89	-------
electrics	8.38	-------
avionics	10.00	-------
rec. sys.	10.12	-------
payload	25.00	-------

total mass = 130.9977 kg

---

packaging:-

From independent variables and geometry:

fuselage length	= 3.750000 m
fuselage diameter	= 0.390000 m
fuselage volume	= 0.518724 m^3
fuel that can be carried	=129.768066 kg

From sortie analysis: using minimum fuselage diameter

	density	volume	length
payload	640.80	0.03901	0.42137
rec. sys.	320.00	0.03162	0.34148
electrics	480.60	0.02762	0.29828
avionics	640.80	0.01561	0.16855
engine		0.05433	0.58679
fuel	800.00	0.08303	0.89675
nose cone		0.09433	0.50942
tail cone		0.01060	0.17167

fuselage length	= 3.394319 m
fuselage diameter	= 0.343344 m
fuselage volume	= 0.356136 m^3
fuel required for sortie	= 66.421989 kg
(includes 10.% reserves)	

c.g. position (x-dir.) = 1.769655

```
    c.g. position of fuel tank = 1.547719

------------------------------------------------------------------

sortie analysis:
----------------

climb performance:
------------------
                     initial altitude =  100.000 m
                     final altitude   = 3000.000 m
                     Mach Number      =    0.500
                     time to height   =   61.439 secs
                     fuel used        =    0.905 kg

cruise performance:
-------------------
                     altitude    = 3000.000 m
                     Mach Number =    0.600
                     distance    =   92.600 km
                     fuel used   =   10.525 kg

loiter performance:
-------------------
                     altitude    = 3000.000 m
                     Mach Number =    0.500
                     loiter time =   30.000 mins
                     fuel used   =   25.864 kg

acceleration performance:
-------------------------
                     altitude          = 3000.000 m
                     initial Mach No.  =    0.500
                     final Mach No.    =    0.800
                     acceleration time =   26.134 secs
                     fuel used         =    0.386 kg

dash performance:
-----------------
                     altitude    = 3000.000 m
                     Mach Number =    0.800
                     dash time   =    5.000 mins
                     fuel used   =   11.628 kg
```

```
manoeuvre phase:
----------------
                    altitude                  = 2000.000 m
                    number of turns           =    2.000
                    Mach Number               =    0.500
                    rate of turn              =    9.562 m/s
                    time taken                =   75.301 secs
                    fuel used                 =    1.097 kg
                    load factor (sus.)        =    3.000
                    load factor (sus. ach.)   =    9.160

return preformance:
-------------------
                    altitude         = 2000.000 m
                    Mach Number      =    0.600
                    distance         =   92.600 km
                    fuel used        =    9.373 kg

         fuel reserves      =     6.642 kg

-----------------------------------------------------------------
         total fuel         =    66.422 kg
         total vehicle mass =   197.420 kg

-----------------------------------------------------------------
```

```
Program  RQPMIN  Version 1.0  for Multics

end of iteration number    41
------------------------------

CONVERGENCE DETECTED  [ Code B ]     after  1318 calls to user routine


free variables
--------------

 5  0.3568697E+01  4  0.2998813E+01  7  0.7720977E+00  1  0.1199735E+04  8  0.5029928E+01  6  0.4543227E+01  3  0.59292

variables fixed at or near their lower bounds
---------------------------------------------

 2  0.1000000E+01

objective function
------------------

f(x) =  0.1744835E+01

active inequality constraints
-----------------------------

 2  0.6307831E-03

inactive inequality constraints
-------------------------------

 5 -0.1353431E+00  6 -0.6604000E+01  8 -0.2418158E+01

Lagrange multiplier estimates
-----------------------------
```

```
 2  0.1529278E-01

Partial derivatives of Lagrangian function
------------------------------------------

 5  0.0000000E+00   4   0.5871996E-01   7 -0.2501234E-01   1 -0.1801203E-01   8   0.3648818E-02   6   0.2150840E-03   3   0.48881(
 2  0.2522860E+00

Norms of active constraint gradients
------------------------------------
 2  0.4769991E+01

convergence criteria
--------------------

pdatum =  0.1744820E+01     ldatum =  0.1744815E+01     gdatum =  0.9750373E-07
nu     =  0.4742487E+02     numax  =    infinite
unormx =  0.9208686E-04     unormd =  0.9208686E-04     rtol    =  0.1000000E-02     xtolu  =  0.1000000E-02
vnormx =  0.5908068E-01     vnorm  =  0.5908068E-01     xtolv   =  0.1000000E-02     grdldn =  0.5871996E-01
nde    =      0      ndef =     33      ndec =     47     ncalls =    382     nfuncs =    302

final vehicle configuration data
--------------------------------

gross wing geometry.
--------------------
                    wing span          = 2.434995 m
                    wing area          = 1.000000 m^2
                    aspect ratio       = 5.929202
                    c.l. chord         = 0.546481 m
                    geo. mean chord    = 0.410678 m
                    aero. mean chord   = 0.425647 m
                    tip chord          = 0.274876 m
                    taper ratio        = 0.502993 m
                    thick/ch ratio     = 0.077210 m
                    1/4 chord sweep    = 4.543227 (degs)
                    l.e. sweep         = 7.701502 (degs)
                    1/2 chord sweep    = 1.357069 (degs)
                    t.e. sweep (fwd)   = 5.020688 (degs)
```

```
nett wing geometry.
-------------------
                    wing span        = 2.078126 m
                    wing area        = 0.812080 m^2
                    root chord       = 0.506675 m
                    taper ratio      = 0.542510 m
                    mean chord       = 0.390775 m
                    aspect ratio     = 5.317955 m

fuselage geometry.
------------------
                    length         = 2.998813 m
                    width          = 0.356870 m
                    height         = 0.356870 m
                    surface area   = 2.739676 m^2
                    x-section area = 0.100025 m^2
                    volume         = 0.349056 m^3
                    skin thickness = 1.068836 mm
                    eng. diam + tol. = 0.343335 m

----------------------------------------------------------------

        engine thrust (s/l)   =  1199.74 n
        design div.speed      =   200.00 (m/S)

----------------------------------------------------------------

    a/c          mass       zero lift
  section        (kg)          drag

wing             7.78        0.00664
fuselage         9.98        0.00664
tail             1.84        -------
engine          31.85        0.00282
eng. inst.      41.40        -------
fly.cont.        7.68        -------
fuel sys.        4.23        -------
electrics        7.18        -------
avionics        10.00        -------
rec. sys.        8.42        -------
payload         25.00        -------

total mass  = 123.5061 kg
```

---

packaging:-

From independent variables and geometry:

```
fuselage length            = 2.998813 m
fuselage diameter          = 0.356870 m
fuselage volume            = 0.349056 m^3
fuel that can be carried   = 50.961113 kg
```

From sortie analysis: using minimum fuselage diameter

	density	volume	length
payload	640.80	0.03901	0.42140
rec. sys.	320.00	0.02631	0.28423
electrics	480.60	0.02374	0.25637
avionics	640.80	0.01561	0.16856
engine		0.05432	0.58675
fuel	800.00	0.06372	0.68827
nose cone		0.09432	0.50939
tail cone		0.01060	0.17167

```
fuselage length            = 3.086640 m
fuselage diameter          = 0.343335 m
fuselage volume            = 0.327631 m^3
fuel required for sortie   = 50.977421 kg
(includes 10.% reserves)
```

```
c.g. position (x-dir.)      = 1.591591
c.g. position of fuel tank  = 1.443484
```

---

sortie analysis:

climb performance:

```
------------------
                    initial altitude =  100.000 m
                    final altitude   = 3000.000 m
                    Mach Number      =    0.500
                    time to height   =   37.428 secs
                    fuel used        =    0.551 kg

cruise performance:
-------------------
                    altitude    = 3000.000 m
                    Mach Number =    0.600
                    distance    =   92.600 km
                    fuel used   =    8.010 kg

loiter performance:
-------------------
                    altitude    = 3000.000 m
                    Mach Number =    0.500
                    loiter time =   30.000 mins
                    fuel used   =   19.866 kg

acceleration performance:
-------------------------
                    altitude          = 3000.000 m
                    initial Mach No.  =    0.500
                    final Mach No.    =    0.800
                    acceleration time =   22.592 secs
                    fuel used         =    0.334 kg

dash performance:
-----------------
                    altitude    = 3000.000 m
                    Mach Number =    0.800
                    dash time   =    5.000 mins
                    fuel used   =    8.850 kg

manoeuvre phase:
----------------
                    alitude                  = 2000.000 m
                    number of turns          =    2.000
                    Mach Number              =    0.500
                    rate of turn             =    9.562 m/s
                    time taken               =   75.301 secs
                    fuel used                =    1.097 kg
```

```
                    load factor (sus.)        =    3.000
                    load factor (sus. ach.)   =    9.604

return preformance:
------------------
                    altitude         = 2000.000 m
                    Mach Number      =    0.600
                    distance         =   92.600 km
                    fuel used        =    7.172 kg

        fuel reserves       =     5.098 kg

----------------------------------------------------------------
        total fuel          =    50.977 kg
        total vehicle mass =   174.484 kg

----------------------------------------------------------------
```