CRANFIELD UNIVERSITY
LIBRARY

CRANFIELD UNIVERSITY

Spencer Woodford

The Minimisation of Combat Aircraft Life Cycle Cost through Conceptual Design Optimisation

CRANFIELD COLLEGE OF AERONAUTICS

Ph.D. THESIS

CRANFIELD UNIVERSITY
LIBRARY

CRANFIELD UNIVERSITY

CRANFIELD COLLEGE OF AERONAUTICS

Ph.D. THESIS

1999

Spencer Woodford

The Minimisation of Combat Aircraft Life Cycle Cost through Conceptual Design Optimisation

Supervisor: Dr. Howard Smith

May 1999

ABSTRACT

In an effort to increase the cost-effectiveness of military equipment, a method has been developed to perform conceptual studies on combat aircraft, resulting in designs of specified capability optimised for minimum Life Cycle Cost (LCC). Consequently, the cost design loop can be considered as being closed, allowing the automated production of a consistent set of cost and performance data for different aircraft solutions. The design engineer can thus make informed, unbiased, design decisions, leading to a more efficient use of shrinking Defence budgets.

Because of the vast scale to which the cost model could be developed, 'deep overheads' are not included, restricting the use of the tool to the comparison of similar weapons systems (combat aircraft), with a common set of design objectives and performance constraints. The aircraft conceptual design tool is based on classical design methods, recently adapted and updated, and validated with existing aircraft data. The engine performance and sizing modules have been developed from detailed thermodynamic models, whilst the LCC model is an amalgamation and update of several different methods, each written for a different phase in the system life cycle.

The aircraft synthesis models, optimisation tool and LCC algorithms are described, and validation results are presented where possible. The software cost model was used to generate a series of results, mimicking the early stages of an aircraft design selection procedure, and allowing a demonstration of the various trade-off studies that can be performed. Results from the selection process are presented and discussed, overall study conclusions are drawn, and areas for further work suggested. Published data for real aircraft and engines are included in the Appendices, together with detailed aircraft parameter and cost output data generated by the model.

This page is intentionally blank.

ACKNOWLEDGEMENTS

The Author would like to thank the staff of the Cranfield College of Aeronautics for their help during the course of this research, in particular the support staff in the Air Vehicle Design Department, and Denis Howe, John Fielding, Robert Jones, Alan Morris, Pericles Pilidis, and John Stollery, for advice in their specialist areas. Special thanks to Howard Smith, for his encouragement, inspiration, guidance, and support through the whole of this research programme.

Thanks also to:

Jack Mattingly, of Seattle University, for provision of the source code for the ONX and OFFX thermodynamic modelling routines, and advice on their improvement.

Jack Vondrell, of Wright Patterson Air Force Base, for his help in gathering and validating the engine data contained in Appendix B, from which the engine dimension and mass prediction methodology was developed.

Wendy Kunc, of the Office of the US Assistant Secretary of the Air Force (Financial Management and Comptroller), and Lisa Johnson of TASC, Inc., for confidential provision of the USAF Visibility And Management of Operation and Support Cost (VAMOSC) data and USAF personnel numbers.

Tina Colarrosi, of the US Air Force Cost Analysis Agency, for her help on engine production cost drivers and prediction methods.

Philip Pugh, formerly of MoD Directorate of Project, Time, and Cost Analysis, for his guidance on drivers, analysis, and thought-provoking discussion of Life Cycle Costing.

Tony Gill and Andy Clifford, of the Defence Evaluation and Research Agency, for their thorough review and feedback during the research study.

Finally, thanks to Vikki, and my Family and Friends, whose love and support has been unwavering through the 'tricky bits'.

This page is intentionally blank.

TABLE OF CONTENTS

ABSTRACT i

ACKNOWLEDEGMENTS iii

TABLE OF CONTENTS v

LIST OF FIGURES ix

LIST OF TABLES xi

NOTATION xiii

1 INTRODUCTION 1

1.1 BACKGROUND AND DEFINITION 2
1.2 RESEARCH JUSTIFICATION 4
1.3 AIMS AND OBJECTIVES 8

2 LITERATURE SEARCH 11

2.1 INTRODUCTION 12
2.2 PREVIOUS WORK 12
2.3 AREAS OF STUDY 15
2.3.1 General Costing Literature 15
2.3.2 Airframe Layout 17
2.3.3 Airframe Materials 18
2.3.4 Propulsion Options 22
2.3.5 Avionics 28
2.3.6 Software 33
2.3.7 Reliability and Maintainability 36
2.3.8 Other Data Sources 41
2.4 SUMMARY 44

3 OPTIMISATION PROCESSES AND TOOLS 45

3.1 INTRODUCTION 46
3.2 LIMITATIONS AND PITFALLS 47
3.2.1 Discontinuities 47
3.2.2 Local Minima 49
3.3 DESCRIPTION OF RQPMIN 50
3.3.1 Problem Formulation 50
3.3.2 Variable Categorisation 51
3.3.3 Quantities Calculated by RQPMIN 51
3.3.4 Convergence Criteria 52
3.3.5 Inner Iterations 53
3.3.6 Newton Iterations 53
3.3.7 The Minimisation Step 54
3.3.8 The Feasibility Step 55
3.3.9 Other Functions 56
3.3.10 Program Termination 56
3.4 SUMMARY 57

4 PROGRAM STRUCTURE AND AIRFRAME DEFINITION 59

4.1 DESCRIPTION AND OVERALL STRUCTURE 60
4.1.1 File Read and Variable Initialisation 61
4.1.2 Component Sizing 61
4.1.3 Mass Estimation and Volume Accounting 62

Contents

4.1.4 Aerodynamic Estimation *62*
4.1.5 Propulsion Modelling *62*
4.1.6 Performance Calculations *63*
4.1.7 Life Cycle Cost Model *64*
4.1.8 Aircraft Synthesis Output Routines *64*
4.1.9 Other Modules *65*
4.2 AIRFRAME COMPONENT SIZING 66
4.2.1 Aircraft Description Variables *67*
4.2.2 Fuselage Sizing *68*
4.2.3 Wing Sizing *69*
4.2.4 Sizing Correlations *71*
4.2.5 Empennage Sizing *72*
4.3 MASS ESTIMATION AND VOLUME ACCOUNTING 74
4.3.1 Structural Mass Estimation *74*
4.3.2 Systems Mass Estimation *76*
4.3.3 Mass Estimation Results *78*
4.3.4 Volume accounting *79*
4.4 SUMMARY 82

5 AERODYNAMICS AND PROPULSION 83

5.1 AERODYNAMIC MODELLING 84
5.1.1 Lift Model *84*
5.1.2 Angle-of-Attack Model *88*
5.1.3 Drag Model *91*
5.2 ENGINE MODELLING ROUTINES 98
5.2.1 Code Modifications *101*
5.2.2 Engine Design and Installation *103*
5.2.3 Off-Design Analysis & Engine Losses *105*
5.2.4 Engine Dimensions and Mass Estimation *111*
5.3 SUMMARY 115

6 AIRCRAFT POINT AND MISSION PERFORMANCE 117

6.1 POINT PERFORMANCE CALCULATIONS 118
6.1.1 Take-Off Performance *119*
6.1.2 Landing Performance *120*
6.1.3 Attained Turn Performance *121*
6.1.4 Sustained Turn Performance *122*
6.1.5 Mach/Speed Performance *123*
6.1.6 Climb and Acceleration Performance *124*
6.1.7 Specific Excess Power Constraint *127*
6.1.8 Calculation of Maximum Altitude *127*
6.2 MISSION PERFORMANCE CALCULATIONS 128
6.2.1 Engine Start, Warm-up, Taxi, etc. *129*
6.2.2 Take-Off *129*
6.2.3 Change in Energy-Height *130*
6.2.4 Cruise *130*
6.2.5 Manoeuvres *131*
6.2.6 Weapons Drop *132*
6.2.7 Loiter / Combat Air Patrol *132*
6.2.8 Landing *133*
6.2.9 Mission Termination *133*
6.2.10 Attained & Sustained Turn Data *133*
6.2.11 Post-Mission Calculations and Constraint Evaluation *135*
6.3 SUMMARY 137

Contents

7 LIFE CYCLE COST MODEL 139

7.1 DESCRIPTION AND MODEL STRUCTURE 140
7.1.1 Research, Development, Test and Evaluation (RDT&E) 140
7.1.2 Production 140
7.1.3 Operation and Support (O&S) 140
7.1.4 Disposal 141
7.1.5 Structure Definition 141
7.2 PARAMETER INITIALISATION 142
7.2.1 Advanced Technology Factor 142
7.2.2 Number of Partners 142
7.2.3 AMPR Mass 142
7.2.4 Consumer Price Index 143
7.2.5 Labour Rates 143
7.2.6 Advanced Materials Factor 144
7.2.7 Collaborative Factor 145
7.3 RESEARCH, DEVELOPMENT, TEST, & EVALUATION 146
7.3.1 Airframe Development 146
7.3.2 Engine Development 148
7.3.3 Avionics Development 149
7.3.4 Total Development 149
7.4 PRODUCTION 150
7.4.1 Airframe Production 150
7.4.2 Engine Production 151
7.4.3 Avionics Production 152
7.4.4 Total Aircraft Production 152
7.4.5 Acquisition Unit Cost and Recurring Flyaway Cost 152
7.5 GROUND SUPPORT EQUIPMENT AND INITIAL SPARES 153
7.6 RELIABILITY AND MAINTAINABILITY 154
7.6.1 Arising Maintenance 155
7.6.2 Scheduled Maintenance 158
7.6.3 Total Maintenance Effort 159
7.7 OPERATION AND SUPPORT (O&S) 160
7.7.1 Operation Personnel 160
7.7.2 Support Personnel 161
7.7.3 Total Personnel Numbers and Cost 162
7.7.4 Service Allowances, Personnel Support and Training 162
7.7.5 Unit Level Consumption 163
7.7.6 Contracts 165
7.7.7 Sustaining Support 166
7.7.8 Installation Support Funds 167
7.7.9 Total Operation and Support Cost 167
7.8 DISPOSAL 168
7.9 LIFE CYCLE COST 169
7.10 SUMMARY 170

8 GENERATION OF RESULTS 171

8.1 INTRODUCTION 172
8.1.1 Optimiser Convergence 172
8.1.2 Combat Validation 174
8.2 PARAMETER DEFINITION 176
8.2.1 Point and Mission Performance Parameters 176
8.2.2 Engine Parameters 177
8.2.3 Procurement and Economic Parameters 179
8.3 CONFIGURATION OPTIONS 180
8.3.1 Single Engine Configurations 180
8.3.2 Twin Engine Configurations 182

Contents

8.3.3 Single Engine vs. Twin Engine Comparison 184
8.3.4 Initial Configuration Selection 187
8.3.5 Internal Weapons Carriage 189
8.3.6 Effect of Advanced Materials 191
8.3.7 Influence of the Objective Function 194
8.3.8 Final Configuration Selection 197
8.3.9 More Electric Aircraft Study 197
8.4 SUMMARY 202

9 DISCUSSION AND CONCLUSIONS 203

9.1 INTRODUCTION 204
9.2 FULFILMENT OF PROJECT AIMS 205
9.2.1 Literature Review 205
9.2.2 Optimisation 205
9.2.3 Program Structure and Airframe Definition 206
9.2.4 Aerodynamics and Propulsion 206
9.2.5 Aircraft Point and Mission Performance 208
9.2.6 Life Cycle Cost Model 209
9.2.7 Validation 211
9.2.8 Generation of Results 211
9.3 AREAS FOR FURTHER WORK 211
9.3.1 Reliability and Maintainability Models 211
9.3.2 Operation and Support Contract Costs 212
9.3.3 Improvement of LCC Model 212
9.3.4 Investigation of Avionics and Software LCC 213
9.3.5 Upgrade of the Engine Modelling Routines 213
9.3.6 Aircraft 'Advanced Technologies' 213
9.3.7 Aircraft Synthesis Modifications 213
9.4 CONCLUSIONS 214

REFERENCES 215

APPENDIX A - AIRCRAFT DATA 223

APPENDIX B - ENGINE DATA 225

APPENDIX C - COMBAT OUTPUT DATA 227

C.1 EXAMPLE AIRCRAFT CONFIGURATIONS 227
C.2 SINGLE ENGINE CONFIGURATIONS 231
C.3 TWIN ENGINE CONFIGURATIONS 247
C.4 ADVANCED MATERIALS STUDY 263
C.5 MORE ELECTRIC AIRCRAFT STUDY 277

Contents

LIST OF FIGURES

FIGURE 1-1. GENERAL WEIGHT, COST, TECHNOLOGY, AND CAPABILITY TRENDS 2
FIGURE 1-2. LIFE CYCLE COST "WINDOW OF OPPORTUNITY" 3
FIGURE 1-3. TREND OF INCREASING COST OF TACTICAL AIRCRAFT[13] 5
FIGURE 2-1. GENERAL ELECTRIC AVEN™ NOZZLE 24
FIGURE 2-2. LCC FOR DIGITAL AVIONICS[95] 30
FIGURE 2-3. % UNSCHEDULED MMH/FH, BY SYSTEM - BRITISH COMBAT AIRCRAFT 39
FIGURE 3-1. EXAMPLES OF (A) WELL-BEHAVED FUNCTION (B) ILL-CONDITIONED FUNCTION[69] 48
FIGURE 3-2. EXAMPLE OF FUNCTION WITH LOCAL AND GLOBAL MINIMUM VALUES[69] 49
FIGURE 4-1. OVERALL PROGRAM FLOWCHART 60
FIGURE 4-2. AIRCRAFT VARIABLE DEFINITIONS 66
FIGURE 4-3. DEFINITION OF FUSELAGE WIDTH AND HEIGHT DIMENSIONS 68
FIGURE 4-4. DEFINITION OF WING DIMENSIONS 69
FIGURE 4-5. GRAPHICAL REPRESENTATION OF COEFFICIENT OF DETERMINATION 71
FIGURE 4-6. TAILPLANE SIZING CORRELATION RESULTS 72
FIGURE 4-7. FIN SIZING CORRELATION RESULTS 73
FIGURE 4-8. MASS ESTIMATION CORRELATION 78
FIGURE 4-9. AIRCRAFT INTERNAL FUEL STORAGE VOLUMES 79
FIGURE 5-1. EXAMPLE LIFT CORRECTION WITH MACH NUMBER 87
FIGURE 5-2. TYPICAL LIFT-CURVE SLOPES 89
FIGURE 5-3. COMPRESSIBILITY DRAG FUSELAGE GEOMETRY REPRESENTATION 94
FIGURE 5-4. EXAMPLE COMPRESSIBILITY DRAG BUILD-UP 95
FIGURE 5-5. EXTERNAL STORE ZERO-LIFT DRAG COEFFICIENTS 96
FIGURE 5-6. EXAMPLE ZERO-LIFT DRAG BUILD-UP 96
FIGURE 5-7. EXAMPLE TOTAL DRAG COEFFICIENT 97
FIGURE 5-8. REFERENCE STATIONS - MIXED-STREAM TURBOFAN ENGINE[65] 98
FIGURE 5-9. ENGINE MISSION ANALYSIS PROCEDURE FLOW CHART 106
FIGURE 5-10. EXAMPLE INSTALLED MAXIMUM THRUST 109
FIGURE 5-11. EXAMPLE INSTALLED MILITARY THRUST 109
FIGURE 5-12. EXAMPLE INSTALLED TSFC - MAXIMUM THRUST 110
FIGURE 5-13. ENGINE INLET DIAMETER CORRELATION RESULTS 111
FIGURE 5-14. ENGINE MAXIMUM DIAMETER CORRELATION RESULTS 112
FIGURE 5-15. ENGINE LENGTH CORRELATION RESULTS 113
FIGURE 5-16. ENGINE MASS CORRELATION RESULTS 114
FIGURE 5-17. PERCENTAGE ERROR BANDS FOR ENGINE PARAMETER PREDICTION EQUATIONS 114
FIGURE 6-1. TAKE-OFF DISTANCE CORRELATION RESULTS 119
FIGURE 6-2. LANDING DISTANCE CORRELATION RESULTS 120
FIGURE 6-3. F-16C ATTAINED TURN LOAD FACTOR COMPARISON RESULTS 134
FIGURE 6-4. F-16C SUSTAINED TURN LOAD FACTOR COMPARISON RESULTS 135
FIGURE 7-1. CONSUMER PRICE INDEX, 1983 = 100.0 143
FIGURE 7-2. RDT&E AND PRODUCTION LABOUR RATES, US$/HOUR 144
FIGURE 7-3. MATERIALS COST FACTOR VS. % BY MASS OF ADVANCED MATERIALS 145
FIGURE 7-4. RECURRING FLYAWAY COST COMPARISON - 1996M$ 152
FIGURE 8-1. EXAMPLE RQPMIN/COMBAT CONVERGENCE PLOT 173
FIGURE 8-2. AIRCRAFT MISSION PERFORMANCE REQUIREMENTS 176
FIGURE 8-3. MAXIMUM AND MILITARY SPECIFIC THRUST VS. IOC YEAR 178
FIGURE 8-4. MAXIMUM & MILITARY TSFC VS. IOC YEAR 179
FIGURE 8-5. SINGLE ENGINE CONFIGURATIONS 180
FIGURE 8-6. SINGLE ENGINE CONFIGURATION MASSES 181
FIGURE 8-7. SINGLE ENGINE CONFIGURATION COST BREAKDOWN 182
FIGURE 8-8. TWIN ENGINE CONFIGURATIONS 182
FIGURE 8-9. TWIN ENGINE CONFIGURATION MASSES 183
FIGURE 8-10. TWIN ENGINE CONFIGURATION COST BREAKDOWN 184
FIGURE 8-11. RELATIVE MASS AND LCC – SINGLE VS. TWIN ENGINE 185
FIGURE 8-12. WEAPON DIMENSIONS 189

Contents

FIGURE 8-13. MASS TOTALS – INTERNAL VS. EXTERNAL WEAPONS CARRIAGE. 190
FIGURE 8-14. COST BREAKDOWN – INTERNAL VS. EXTERNAL WEAPONS CARRIAGE. 191
FIGURE 8-15. MASS TOTALS VS. ADVANCED MATERIALS FRACTION..... 192
FIGURE 8-16. LCC BREAKDOWN VS. ADVANCED MATERIALS FRACTION..... 193
FIGURE 8-17. MASS TOTALS – MASS AND LCC-OPTIMISED SOLUTIONS..... 194
FIGURE 8-18. COST BREAKDOWN – MASS AND LCC-OPTIMISED SOLUTIONS..... 196
FIGURE 8-19. MASS TOTALS – MORE ELECTRIC AIRCRAFT OPTIONS..... 199
FIGURE 8-20. COST BREAKDOWN - MORE ELECTRIC AIRCRAFT OPTIONS. 200

List of Tables

Table 2-1. COCOMO Embedded Mode software features. 34
Table 2-2. Benefits of Improved Reliability 38
Table 4-1. Empirical Wing Mass Factors. 74
Table 4-2. Aircraft System Densities 81
Table 5-1. High-Lift Device and Configuration Lift Coefficient Increments. 85
Table 5-2. Engine Reference Station Definitions. 99
Table 5-3. Component Figures of Merit for Different Technological Levels. 101
Table 6-1. Combat Point Performance Constraint Codes 118
Table 6-2. Point Performance Constraint Variable Definitions. 118
Table 6-3. Mission Performance Codes 128
Table 6-4. Mission Variable Explanations 128
Table 8-1. Example Optimisation Start and Finish Values. 173
Table 8-2. Quoted and Calculated Aircraft Parameters. 175
Table 8-3. Single vs. Twin Engine Mass and Volume Comparison. 185
Table 8-4. Aircraft Parameters - Mass and LCC-Optimised Solutions. 195
Table 8-5. Mass, Maintenance, and Cost Factors for MEA Systems. 198
Table 8-6. Mass Changes – More Electric Aircraft Options. 199
Table 8-7. Cost Changes – More Electric Aircraft Options. 201

Contents

This page is intentionally blank.

NOTATION

Abbreviations

AFB	Air Force Base
AFFDL	Air Force Flight Dynamics Laboratory (US)
AIAA	American Institute of Aeronautics and Astronautics
AMPR	Airframe-only mass (defined by MIL-STD-1374)
APU	Auxiliary Power Unit
ATF	Advanced Tactical Fighter
AVEN	Axisymmetric Vectoring Exhaust Nozzle
BIT	Built-In Test
CAIG	Cost Analysis and Improvement Group
CER	Cost Estimating Relationship
CFD	Computational Fluid Dynamics
CIP	Component Improvement Programme
COCOMO	COnstructive COst MOdel for software cost prediction
COSS	Cost Of Support Spreadsheet
CPI	Consumer Price Index
DAeT	Department of Aerospace Technology
DB	Diffusion Bonding
DERA	Defence Evaluation and Research Agency
DLR	Depot Level Reparable
DMHR	Defect Man-Hour Rate
DOC	Direct Operating Cost
DoD	Department of Defense (USA)
DPTCAn	Directorate of Project, Time, and Cost Analysis
DTLCC	Design to Life Cycle Cost
ECS	Environmental Control System
EHA	Electrohydrostatic Actuator
EMA	Electromechanical Actuator
FBW	Fly By Wire
FCS	Flight Control System
FH	Flying Hours
GAO	General Accounting Office (USA)
GE	General Electric
GEAE	General Electric Aircraft Engines
GSE	Ground Support Equipment
HARV	High Angle-of-attack Research Vehicle
HPT	High-Pressure Turbine
IMA	Integrated Modular Avionics
IOC	Initial Operational Capability
IR	Infra-Red
IS	Initial Spares
ISA	International Standard Atmosphere
JIT	Just In Time
LCC	Life Cycle Cost

Notation

LEX	Leading-Edge eXtension
LO	Low Observable
MASA	Modular Avionics Systems Architecture
MATV	Multi-Axis Thrust Vectoring
MEA	More Electric Aircraft
MFHBF	Mean Flight Hours Between Failures
MFN	Multi-Function Nozzles
MLCCM	Modular Life Cycle Cost Model
MMH	Maintenance Man-Hours
MPT	Manpower, Personnel, and Training
MTBF	Mean Time Between Failures
MTBM	Mean Time Between Maintenance
MVO	Multi-Variate Optimisation
NASA	National Aeronautics and Space Administration
O&S	Operation and Support
OBOGS	On-Board Oxygen Generation System
OFFX	Off-design thermodynamic engine model used by COMBAT
ONX	On-design thermodynamic engine model used by COMBAT
OSD	Office of the Secretary of Defense (US)
PCS	Permanent Change of Station
POL	Petroleum, Oil, and Lubricants
PW	Pratt and Whitney
RDT&E	Research, Development, Test & Evaluation
RQPMIN	Optimisation Tool used by COMBAT
SAWE	Society of Allied Weight Engineers
SFC	Specific Fuel Consumption
SGR	Sortie Generation Rate
SLOC	Source Lines of Code
SLS	Sea-Level Static
SPF	Super-Plastic Forming (normally of titanium)
STOL	Short Take-Off and Landing
TAS	True Air Speed
TET	Turbine Entry Temperature (also called T_{T4})
TSFC	Thrust Specific Fuel Consumption
USAF	United States Air Force
USN	United States Navy
VAMOSC	Visibility And Management of Operation and Support Costs
VHSIC	Very High Speed Integrated Chips
VLO	Very Low Observable
VTOL	Vertical Take-Off and Landing
WAT	Wrap-Around Test

Notation

COMBAT Variable Names

AEFN	aspect ratio of vertical fin
AETN	aspect ratio of horizontal tail
AW	aspect ratio of wing
BFH	width (maximum) of fuselage (m) (see Figure 4-2)
BW	wing span (m)
CACQ	acquisition cost ($)
CDADDF	coefficient of zero-lift flap drag
CDADDL	coefficient of zero-lift landing gear drag
CDADDS	coefficient of zero-lift stores drag (see Figure 5-5)
CDC	compressibility drag coefficient
CDCASE	coefficient of aircraft drag at the flight conditions
CDCF	coefficient of fuselage compressibility drag
CDCINT	coefficient of interference compressibility drag
CDCW	coefficient of wing compressibility drag
CDDS	cost of development support
CDF	coefficient zero-lift friction drag
CDFT	cost of development flight testing
CDMME	cost of development manufacturing materials ($)
CDP	coefficient of wing pressure drag
CDPF	cost of development aircraft engines ($)
CDQL	cost of development quality control ($)
CFCOR	ratio of compressible to incompressible friction drag at flight Mach number
CLAB	wing-alone lift-curve slope with no thickness effects (/°)
CLCASE	lift coefficient at current flight condition
CLDES	wing design lift coefficient
CLE	labour rate - engineering ($/hr)
CLHS	high-speed (manoeuvre) lift coefficient
CLLA	landing lift coefficient
CLM	labour rate - manufacturing ($/hr)
CLT	labour rate - tooling & quality control ($/hr)
CLTO	take-off lift coefficient
COSZ	cosine of ½ chord sweep
CPP	cost of engines for production aircraft - average cumulative ($)
CPR	civilian pay rate ($)
CRF	recurring flyaway cost ($)
CWCB	chord of wing root (m)
CWCC	chord of wing centreline (m)
CWCT	chord of wing tip (m)
DAR	engine depot arrival rate
DELTCL	factor for manoeuvre lift coefficient
DELTEL	landing trailing-edge device lift contribution - conventional configurations
DELTET	take-off trailing-edge device lift contribution - conventional configurations
DELTH	high-speed configuration lift increment
DELTL	low-speed configuration lift increment
DELTLE	leading-edge device lift contribution - conventional configurations

Notation

DFH	diameter (effective) of fuselage (m)
DPH	diameter (maximum) of engine (m)
DTT	engine relative technology metric (see section 7.4.2)
DZE	change in energy height (m)
EDAE	effort - development airframe engineering (hr)
EDML	effort for development manufacturing (hr)
EDQL	effort for development quality control (hr)
EDTP	effort for development tooling design (hr)
EMST	total scheduled maintenance effort (man-hours/flight-hour)
EMT	total maintenance effort (man-hours/flight-hour)
EPR	enlisted pay rate ($)
ERM	engine repair ratio
ERO	engine overhaul ratio
ET	total arising maintenance effort (man-hours/flight-hour)
FADT	cost factor for advanced design tools
FAMA	cost factor for advanced materials
FATF	cost factor for advanced technology
FATT	cost factor for advanced technology testing
FBWFI	ratio of inner flap position as fraction of wing chord (see Figure 4-4)
FBWFO	ratio of outer flap position as fraction of wing chord (see Figure 4-4)
FDRAG	drag force (N)
FHY	average flying rate (hours/year)
FLO	cost factor for low-observable technology
FMF	fuel mass fraction – finish of energy-height manoeuvre
FMFG	factor to allow for advanced materials in fuselage volume calculations
FMFMT	fuel mass fraction of gross mass
FMI	fuel mass fraction – start of energy-height manoeuvre
FMOF	crew ratio
FPART	cost factor for number of partners
FPROR	engine overhaul/repair ratio
FUNC()	array containing RQPMIN constraint calculation values
GITR	instantaneous turn rate load factor
GITRC	calculated instantaneous turn load factor
GND	aircraft design limit load. (g's)
GSTR	sustained turn rate load factor
HFH	height (maximum) of fuselage (m) (see Figure 4-3)
HFM	height of fuselage centreline from ground (m)
HMAX	maximum aircraft altitude (m)
IAPU	auxiliary power unit (yes/no)
IDRAG	write drag data output file (yes/no)
IFLAP	flap type (single-slotted/ double-slotted)
IMS	integral mean slope for nozzle loss calculations
IOPT	objective function selection
IPRF	write performance data output file (yes/no)
ISCRN	write COMBAT output to screen (yes/no)
KV	factor for vortex lift calculation
MACHLE	Mach number leading-edge

Notation

MAXI	installed avionics mass (kg). If 0.0, will be estimated by COMBAT
MBI	mass of design mission internal payload - no drag penalty (kg)
MBMAX	mass of maximum allowed aircraft payload (kg)
MBX	mass of external payload to be carried on design mission (kg)
MCP	mass of crew (kg)
MDD	Mach number difference from wing design Mach number
ME	mass of empennage (kg)
MGCA	mass of gun and ammunition (kg)
MLOW	Mach number for structure mass calculations (design diving speed)
MMAX	Mach number (maximum)@ high-altitude (> 11000m)
MPA	mass of APU (kg)
MPAD	engine air mass flow rate (kg/s)
MTE	mass of the aircraft – empty (kg)
MTG	mass of the aircraft – gross (kg)
MTGF	mass of aircraft fuel – total (kg)
MTGFI	mass of internal fuel (kg)
MTGFX	mass of external fuel (kg)
MTGFXI	mass of external fuel allowed for design mission (kg)
MTT	mass of the aircraft – mission (kg)
MWC	mass of complete wing (kg)
NAC	number of aircraft on main operating base
NCP	number of crew
NEF	number of fins
NENG	number of engines
NHARD	number of pylons available for weapons/fuel carriage.
NPROD	number of production aircraft
NRDTE	number of development airframes
NSHAFT	number of main engine shafts
NSTAT	number of static test development airframes
NULT	load factor - ultimate (g)
OFFDAT	engine output array
OFGB	area for fuselage base drag calculations (m^2)
OII	area of engine inlet (m^2)
OPI	fuselage cross-sectional area excluding the intakes
OPR	officer pay rate ($)
QFLA	landing flap setting angle for flap pressure drag calculations (°)
QWL	leading-edge sweep of wing (°)
RCDEF	fin form factor (for drag calculations)
RCDET	tail form factor (for drag calculations)
RCDF	fuselage form factor (for drag calculations)
RCDW	wing form factor (for drag calculations)
RCOMB	combat radius (km)
RCWF	ratio of flap chord/wing chord (see Figure 4-4)
RDFL	fuselage fineness ratio (for drag calculations)
RDN	degree-to-radian conversion ($\pi/180$)
RMATAD	unused fraction of advanced materials, by empty mass
ROAC	airframe packing density (kg/m^3)

Notation

ROLE	aircraft role (fighter, ground attack, trainer)
RPBY	engine bypass ratio at sea-level static maximum power
RPCH	compressor pressure ratio at sea-level static maximum power
RTEF	thickness/chord of vertical fin
RTET	thickness/chord of horizontal tail
RTPG	ratio of available to required thrust (optimiser constraint function)
RTW	thickness/chord ratio of wing
SEFN	area of vertical tail surface (m^2)
SETN	area of horizontal tail surface (m^2)
SLAND	landing ground roll (m)
SLAT	leading-edge device (flap or slat) available (yes/no)
STOG	take-off ground roll (m)
TAIL	tail placement (aft-tail, delta, canard)
TLIFE	life of aircraft
TPGD	uninstalled SLS engine military thrust (max. dry thrust) (N)
TPGH	uninstalled SLS engine maximum thrust (max. wet thrust) (N)
TWOD	two-dimensional lift curve slope (/°)
TYPE	aerofoil type for subsonic wings (conventional/supercritical)
UMATAD	advanced materials fraction, by empty mass
UW	taper ratio of wing
VBI	volume required for internal weapons bays (m^3)
VCI	volume of cockpit (m^3)
VFG	fuselage volume (total) (m^3)
VFI	volume of fuselage nose section (m^3)
VFW	volume of fuselage structure (m^3)
VIDG	volume of the engine intake ducts (m^3)
VM	true air speed (m/s)
VMAX	maximum speed (Kt)
VPB	volume of engine bay (m^3)
VRO	speed for constant dynamic pressure during climb (m/s)
VSTALL	stall speed (m/s)
VTGF	volume required for mission fuel (m^3)
VTGFI	volume available for internal fuel storage (m^3)
VTGFI	volume of internal fuel (m^3)
VTO	take-off velocity (m/s)
VWFG	volume available for wing fuel storage (m^3)
VZ	vertical component of climb speed (m/s)
WAMPR	airframe-only mass (lb.)
WV	weight of uninstalled avionics (lb.)
XFN	length of fuselage (m)
YRACC	accounting year
YRIOC	year of initial operational capability
ΔDIST	distance increment for cruise calculations (km)

1 Introduction

This chapter gives an introduction to the research study, describing the background behind the project and the definition of the project boundaries. The project justification is detailed, which, together with the project definition, enables the research aims and objectives to be decided. These have been written in a single objective statement, with explanations of the key words and phrases, enabling a full description of the scope of the programme of work to be made simply and efficiently.

1.1 Background and Definition

Life cycle costing (LCC) was first applied to military projects by the US Department of Defence in the early 1960's. It has become more popular and important in the procurement of military equipment, as the budgets for the World's fighting forces are ever-increasingly tightened. The reasons for this are numerous and highly involved, needless to say that the end of the Cold War, the global recession of the early nineteen-nineties, and the flood of cheap military equipment from the former Soviet Union have all played contributing roles.

LCC is a complex and wide ranging subject that is concerned with quantifying options to ascertain the optimum choice of assets and asset configuration. When related to a combat aircraft, this leads to the type of aircraft, its specification, and configuration. In order to provide defensive and strike roles effectively in the face of improvements in the potential enemies' forces, it has been necessary to continually advance the performance, capability, survivability, operational role, and support characteristics of the aircraft and its associated weapon systems and countermeasures. This in turn has led to increasing complexity of aircraft and systems and, in most instances, increasing costs, not only in absolute terms, but also in real terms (accounting for inflation).

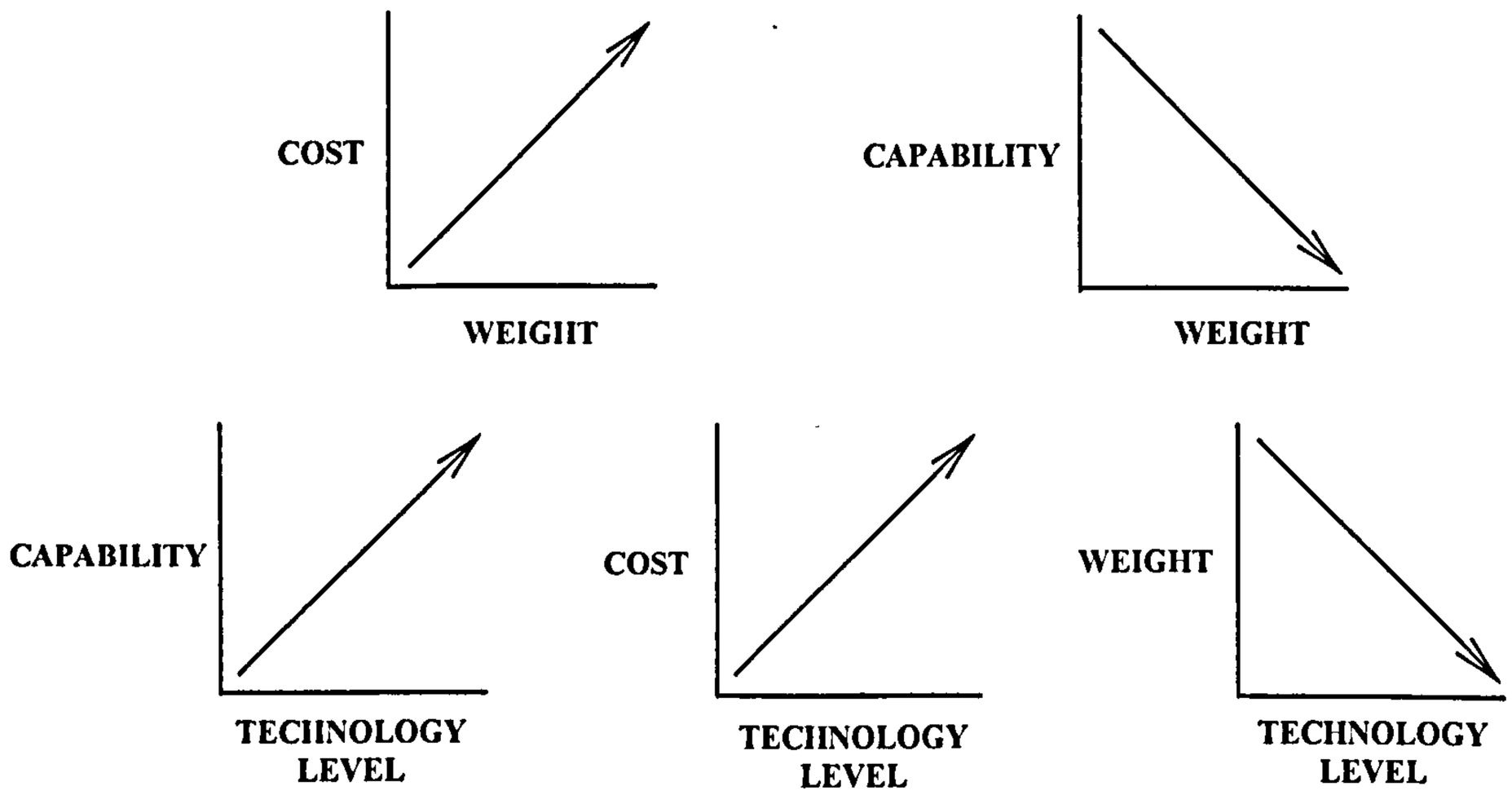


Figure 1-1. General Weight, Cost, Technology, and Capability Trends.

Figure 1-1, presented by Andrews[3], shows that weight plays a major part in the determination of the life cycle costs of an aircraft, especially in the development and acquisition phase of its life, where most of the cost trends have historically been related to component masses. It is also apparent that weight can reduce with increasing levels of technology, such as the use of advanced materials, although these new materials cost more per unit mass, in terms of both design and manufacturing. It follows that there may be a point where the production cost benefit derived from the reduced mass will be outweighed by the cost of the advanced materials and their more expensive production processes. However, lighter aircraft will have cost benefits down-stream, and it is these trade-offs that must be investigated before a design philosophy is chosen.

It is becoming increasingly popular to try to include LCC analysis at the conceptual design stage, and to incorporate the idea of Design To Life Cycle Cost (DTLCC) right from the start of a programme. This is because of the diminishing opportunity to influence LCC the further the design progresses. This is dramatically represented by the "window of opportunity", discussed by Apgar[12] and others, shown in Figure 1-2.

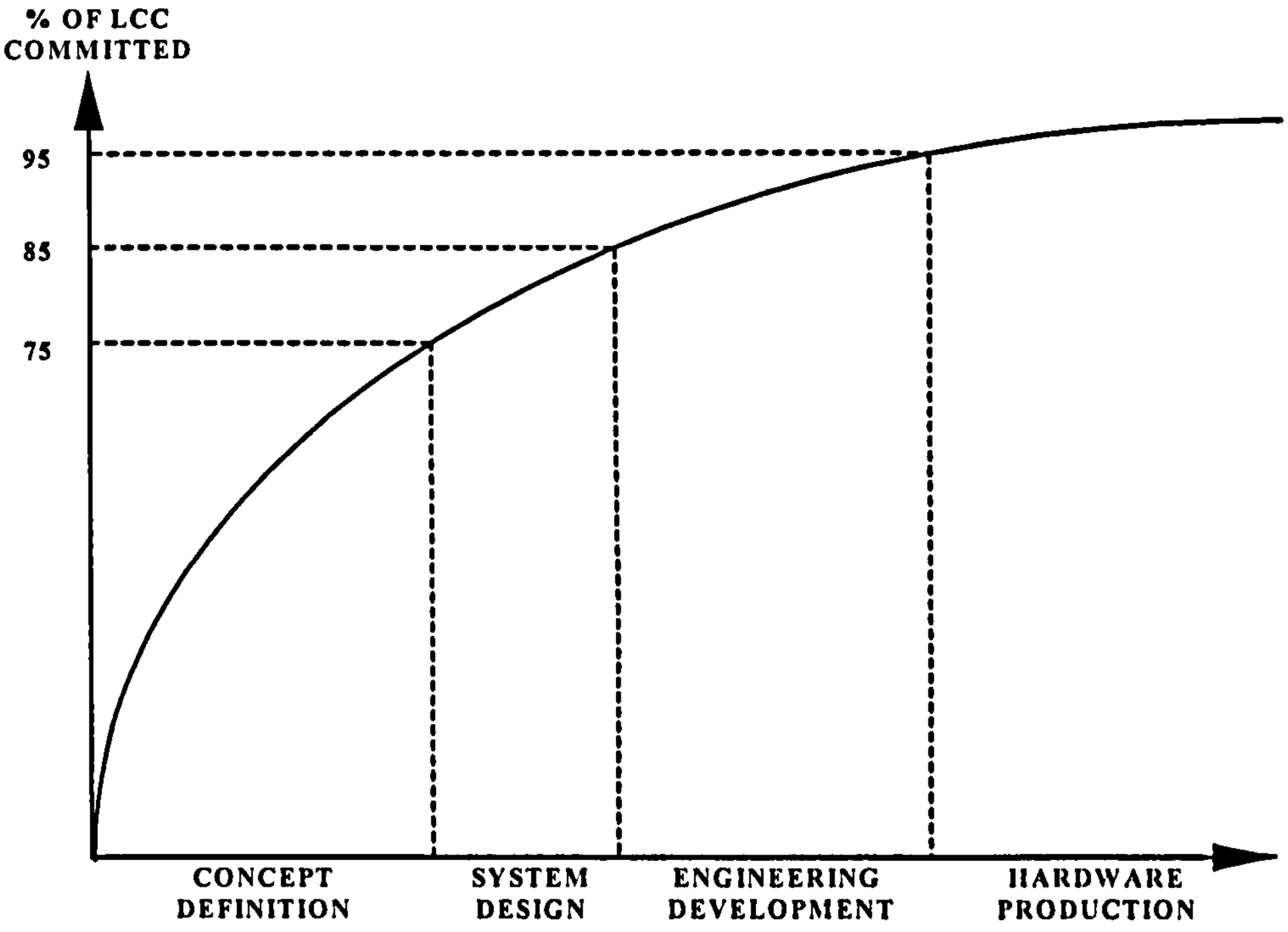


Figure 1-2. Life Cycle Cost "Window of Opportunity".

From this diagram, it can be clearly seen that the earlier on in the design process that the crucial decisions concerning the overall aircraft concept are made, the more the LCC can be affected and hence the more money that can be saved. It should be noted that in-service changes from the intended operating regime of the aircraft (Avro Vulcan), reliability improvement programmes (F-16), and extreme life extension programmes (B-52) may change the relative values in Figure 1-2 by altering the in-service cost fraction. However, if such changes have not been foreseen, they cannot be accounted for in the design process, or built into the LCC modelling tool, and it is debatable whether the design should be compromised to account for such unpredictable events. It is vital that LCC should be considered along with the initial design and that a stringent 'design-for-life-cycle-cost' programme be implemented and followed, not only by the main contractors, but also by their sub-contractors. This will lead to a more cost-effective aircraft, which will in turn lead to lower costs or larger fleets, from which further economies of scale can be drawn. Either way, the procurement agency, be it the MoD or foreign buyers, will be getting more "bang for their bucks".

What has become apparent, particularly with the collapse of the former Soviet Union, is that the previous massive defence spending can no longer be sustained, both financially, and politically. With waning support from the public, whose perception of a serious military threat has been all but removed, governments must be seen to be cutting defence spending in order to facilitate increases in spending on welfare and other

domestic programmes. Thus, military equipment must now be shown to present 'value for money' in both the long and the short term. As 'value' is difficult to quantify in the military sense, this project has chosen to avoid seeking 'value for money', but rather to try and minimise the Life Cycle Costs of an aircraft designed for a particular level of capability, mission performance and operational requirement.

1.2 Research Justification

As with any doctoral research, the aim is to provide an 'original contribution to knowledge'. With time this becomes more difficult, as many areas of important research have been investigated previously. The research study described in this thesis is both similar and different to many previous research studies at Cranfield. It is similar in that it uses Multi-Variate Optimisation (MVO) to perform the synthesis and analysis of an aircraft. It differs from previous studies in the scale and complexity of the objective functions available for the optimisation, this research being the first to examine military aircraft with an emphasis on life cycle cost.

In previous studies of military aircraft, the objective function (i.e. the variable subject of the optimisation) was nearly always mass; either empty, mission or gross mass. One of few recent studies undertaken in the United Kingdom where an aircraft cost function was to be minimised using MVO was that performed by Wilson[106], where the Direct Operating Cost (DOC) for civil airliners was the objective function. This work built on a series of MVO programmes, started at the Royal Aircraft Establishment (RAE) in the early 1970's with Perry[74], and continuing with developments by Peckham[73], Kirkpatrick[57], and Collingbourne[26]. In the civil world, DOC is normally the figure of most interest to airlines, as it is the figure that allows them to decide flight charges, and ultimately calculate profit. In the military environment, 'profit' is not shown, although the peacetime costs of operation are still just as important. This is increasingly the case in an environment of upwardly spiralling costs and reducing defence budgets, where performance and flexibility are still to be maximised, but where costs must be reduced.

Augustine[13] and Pugh[77] point to the exponential increase in costs of military equipment with time. This is surely connected with the dramatic increase in performance that has been witnessed in recent years, but there must also be a case against the procurers, and designers of those systems, for in the past there was almost no limit on the cost that would be absorbed for military programmes. This led to an industry where there was little regard for cost, and much emphasis on outright performance; engineers were not previously required to restrict costs in the same way as today. For example, when the F-111 entered service in the late 1960's, it required more than 45 maintenance man-hours per flight hour (MMH/FH), imposing a huge burden on the operating units. With more modern design practices and the emphasis shifted more towards a life-cycle approach, the F-18 entered service in the early 1980's with a requirement of approximately 25 MMH/FH, significantly cutting costs by reducing personnel and spares provisioning, albeit for smaller aircraft and higher initial cost. This has now been taken a step further with the Eurofighter Typhoon and the F-22, which have quoted design MMH/FH values of 9 and 12 respectively.

This shift in cost and the savings possible is one of the most interesting aspects of LCC analysis; how do we achieve the correct balance between spending at the start of the programme and during operations in order to minimise the LCC? It is clear that an increase in development costs to improve reliability and reduce maintenance should reduce operation and support (O&S) costs. Provided that the increases in development costs are not as large as the savings made from the reduced maintenance costs (in real terms), then the LCC has been reduced. It is the intention of this research to enable the designer to quantify the LCC difference between competing designs, including twin or single engines, aircraft configuration and geometry, numbers of crew, use of advanced materials, engine design parameters, and other conceptual design variables. In this way, a solution can be selected early in the in the design process that will meet all of the performance and mission requirements, whilst reducing the LCC. It will also highlight specific performance requirements that may be driving the design, perhaps producing a more balanced set of performance requirements for the initial specification. All of the above effects should help to reduce the cost of future aircraft weapon systems.

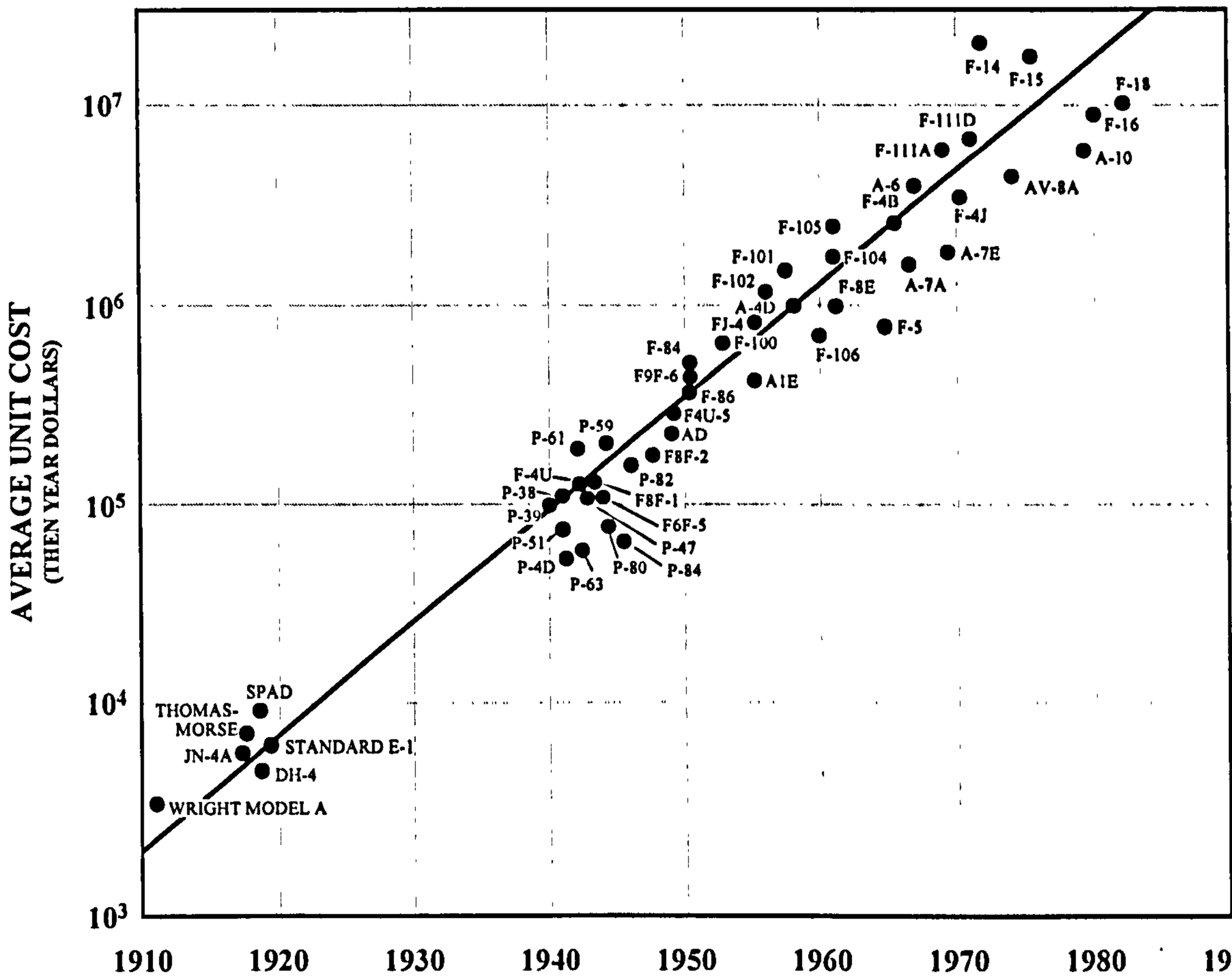


Figure 1-3. Trend of Increasing Cost of Tactical Aircraft[13].

Recently, concepts such as 'Concurrent Engineering' have been making headlines, yet this is nothing more than common sense re-written in modern management jargon. There is now, quite correctly, a greater emphasis placed on the total cost of weapons systems, from 'cradle to grave'. This, in time, will lead to the changing of past cost

trends and will eventually produce a more efficient, and probably effective, fighting force. One need look only as far as the automotive industry and compare a modern vehicle with one 30 years old to see the changes that are possible when there is an unwavering demand for change. A similar change must take place in the military aerospace industry, albeit by a smaller amount due to the very demanding performance requirements and operating environment of military aircraft. It is hoped that this research will prove to be a useful study, and will result in a tool that forms a basis for a change in the aircraft design process in the United Kingdom.

Whilst the concept of costing is the same across all engineering disciplines (aerospace, automotive, civil, electronics, etc.), the individual costing processes and estimating relationships are peculiar for each one, as are the main cost drivers. It is not realistic to try to apply a civil aircraft costing approach to a military combat operating regime for a number of reasons, as outlined below. It should be noted that the following points apply to a comparison of turbo-engine combat aircraft and civil airliners.

- The design missions and operating environments are completely different, with military aircraft spending large amounts of time at low-level, full thrust, and high load case flight phases. Civil airliners spend the majority of their lives at high altitude, with moderate to low thrust levels, depending on the number of engines on the aircraft. The loads endured by military aircraft are therefore much more severe and consequently the amount of maintenance required is an order of magnitude higher than for civil aircraft.
- The operating procedures are not similar, with civil aircraft being maintained by a specialist company, which is often not connected with the actual operator of the aircraft. This is in direct contrast to military aircraft, which are maintained (for the most part) by the operating Air Force, which has full responsibility for the cost of the maintainers and their dependants.
- The staffing levels for civil maintenance contractors are determined by the maintenance requirements, and from the numbers of aircraft they service. The number of personnel required by military maintenance facilities is mainly dictated by that Air Force's readiness requirement. Readiness is a measure of the military requirement that they should be able to respond to a threat by achieving a specified sortie generation rate (SGR) within a threshold time limit. This drives the numbers of maintenance personnel up, as wartime operating tempos are likely to be higher than those encountered in peacetime operations.

Once it was ascertained that a civil cost model could not simply be applied to military aircraft, a literature search was performed to ascertain what costing models were available, and whether they alone were enough to build a comprehensive military aircraft Life Cycle Cost model. The literature review details are given in Chapter 2, but the main finding was that there was very little aircraft costing literature available, and no literature at all in the public domain describing realistic models for LCC estimation of combat aircraft. Of course, there are many papers and journal articles available discussing the through-life costing process in general, and many for other engineering disciplines, but there was found to be a severe shortage of good information for military aircraft LCC evaluation.

Several models dealing with individual cost phases were discovered and utilised in an attempt to generate a realistic cost model covering all of the major life phases. It was soon apparent that many of these models would need to be upgraded and calibrated with what little data is available. During the course of the literature review, data was also sought, in an attempt to validate existing algorithms, and help with the development of new models or upgrading of those found to be deficient. This was an enlightening process, as it soon became apparent that costing data is possibly the most 'variable' of all 'official' data. There are several reasons for this, some of which are outlined below.

- It is often not specified whether the quoted values are costs to the manufacturer, or to the purchaser, i.e. a price. It is unlikely that manufacturers would publish true costs, as this could be used by procurement agencies to reduce selling prices and profits.
- Because of the nature of the market, selling price will vary significantly from one buyer to another, depending on mainly political factors, and whether price cuts to obtain an order will be later reclaimed through sales of spares and support. It is often said that this is particularly true for engine manufacturers, although it is not known why this should be the case.
- The number of aircraft purchased will have a significant effect on the selling price, as aircraft are very rarely sold 'off-the-shelf'. Most military customers will request specialist equipment fits (particularly avionics), which can have a dramatic impact on the cost of the aircraft. There are additional considerations, such as discounting for larger orders, and long lead-times, which complicate the accounting process.
- Because of the variation in inflation and foreign currency values, prices can fluctuate according to local economic conditions. Whilst this does not normally account for very large discrepancies, an error of 5-10% would not be impossible, depending on when the values were calculated, and the assumptions made for the exchange rate values. In addition, values quoted in January can be as much as 5 or 6% different to those quoted in December of the same year, yet the data will often simply be presented as a 'then-year' value.
- It is commonly accepted that costs to the home nation will be less than to foreign buyers, as the government will undoubtedly have sponsored the aircraft development programme to some degree, and will purchase spares from the manufacturer, as mentioned above. The exception to this may be from the former Soviet Union, where foreign currency is so important that sales may take place at (or below) cost price to ensure an influx of foreign currency. This is one of the problems facing Western manufacturers, as alluded to in previous sections.
- Some cost values are labelled incorrectly and differences in terminology prevent firm conclusion being drawn as to the true cost of the item. Quoted values often include costs for support equipment, logistics, operator training and other services. Whilst these form part of the total package, they should be defined as such, and not included under headings such as 'Recurring Flyaway Cost', which has a strict definition.
- Cost values quoted by official agencies are seldom the cost of the original item, but a depreciated insurance cost. As this is calculated using complex, sensitive, formulae to account for the purchase price, remaining life, perceived value, and other factors, it is almost impossible to calculate the original cost of the item without access to the original algorithms.

All of the above components make it very difficult to define a cost for a particular piece of equipment. In some cases, this is probably deliberate, in an attempt to deceive competitors and create interest in the product. In many cases, it is naiveté on the part of those quoting cost values, and their lack of understanding of conventions and common practices. Whatever the reasons for the inconsistencies, it produces a very complex tangle of figures for the estimator to unravel before the analysis procedure can begin.

1.3 Aims and Objectives

From the previous sections, several parts of the main research objective will have become apparent, but before these are formalised, the main thrust of the work should be explained. As with many previous Doctoral and Masters studies at Cranfield, this research was sponsored by the Defence Evaluation and Research Agency (DERA), and it was with them that a formal requirement was established. That requirement can be summed up as follows:

"To produce a useable, working model to optimise the design of combat aircraft for minimum peacetime life cycle cost."

The objective statement above contains several key words and phrases, which are expanded on below. By implication, the word 'model' indicates a software tool that can be used to perform trade studies and sensitivity analyses in the decision leading up to the final selection of the most appropriate aircraft design. It should be stressed that the aim of the research is not to predict to high levels of accuracy the total cost of ownership of a particular weapon system. Rather, it is to develop a tool that allows the comparison of similar weapon systems - in this case combat aircraft - designed for the same mission and operating environment. That said, it is necessary to try to calibrate each section of the cost model with real data to ensure that the LCC quantities are relatively accurate, so that meaningful trade studies can be performed.

The word 'useable' is included to indicate that the software tool must be able to be operated by any intelligent individual, without advanced qualifications in either aircraft design or in cost modelling. Obviously, aircraft design experience would make the understanding of the tool processes and model operation easier, but would not necessarily improve the quality of the results generated. Indeed, experience in classical aircraft conceptual design methodologies can be a hindrance, as the method used to design the aircraft using an optimisation tool, explained in Chapter 3, is the complete reverse of the classical methods. In order to facilitate this ease of use, the models have been written with a non-expert user in mind, and inputs have been kept to a minimum. Wherever judgement factors are required from the user, real-life figures are suggested, together with examples of how different technological levels may affect that particular variable. Those who have used the model have found it relatively easy to use, once they have adopted a new way of thinking about the design methodology, and the apparent reversal from the classical design process.

'Working' is intended to show that the model had to be operational at the completion of the research, with a trade-off being made between accuracy of the individual modules, and ensuring that all options had been tested. It was felt that the full capability of the

models would not be realised in one Doctoral study, so provision had to be made for later improvements and modifications. This was to be facilitated by writing the software in a modular format so that changes to individual models could be implemented simply and efficiently. The choice of a well-established language was also fundamental to the utility of the model, so that portability can be maximised. FORTRAN 77 was chosen to be the coding language, as it is widely used, compilers are available for most computing platforms, and the optimiser is already written in FORTRAN.

It is clear from the objective statement that the scope of the study is limited to combat aircraft. In this application 'combat' aircraft are limited to aircraft designed for air-to-air combat, ground attack, or advanced flying training, powered by gas turbine engines, and operated by the Royal Air Force. Aircraft having 'flying wing', 'blended wing-body', or other very low observable configurations/technologies are excluded.

The optimisation process is mentioned twice in the objective statement, confirming the use of an optimisation tool. As with many previous aircraft design studies, the optimiser utilised is the DERA-developed RQPMIN[92]. This is a very powerful and efficient optimiser, with a history of robust convergence. It is also well used and understood at both Cranfield and within DERA, where it is one of the main optimisation tools. It makes use of gradient-based search patterns, and has several features to enhance the speed and robustness of the search. It is explained in greater detail in Chapter 3, along with a brief explanation of the theory behind it.

The research study is limited to a peacetime operating regime, because in war the cost of loss is not normally measured in currency terms, and all available resources are made available in order to effect a satisfactory outcome. It would be inappropriate to try and model all of the factors that would contribute to the wartime cost of the aircraft, when much of this cost could be decided by the survivability and lethality of the aircraft and competing weapons systems. Whilst these are extremely important issues in the design of a combat aircraft, they would only serve to complicate the optimisation process by introducing conflicting requirements and noise into an already-complex system.

The final phrase in the objective statement, 'life cycle costs' is one of the most difficult values to quantify for any aircraft. Even civil airlines, which rely on knowing costs to calculate profits, have extreme difficulty in detailing life cycle costs for aircraft on their fleets, let alone predict costs for future purchases. There are many reasons for this, as was touched on earlier, but one of the most obvious reasons is not to do with aircraft design or operation, but rather one of accounting. The allocation of costs is a constant dilemma for analysts, and one that must be overcome early on if any progress is to be made. For this reason, it was decided that only costs that were a direct result of the design of the aircraft, or were scaled from other direct costs, would be included in the research scope. For military applications 'deep overheads', such as the cost of air traffic control or infrastructure costs, would not be included, as these do not depend on the design of the aircraft, and would be incurred regardless of the aircraft system being used. If the aim was to predict total weapon system costs, there would be a strong case for including these, so that comparison with other weapon systems could be made.

It soon becomes apparent that the scale of the work is vast, for not only must a series of relatively accurate cost models be developed and calibrated, but they must be driven by

a flexible, yet accurate, aircraft conceptual design package, and linked with a powerful optimisation tool. To complicate matters further, it was specified that the aircraft design model should not rely on externally supplied tabular data for the engine performance calculations and fuel burn estimation. It was therefore decided to make use of an existing suite of engine design and analysis codes, ONX and OFFX, written by Dr. Jack Mattingly, and presented with his first book[65]. These programs have added significantly to the capabilities of the aircraft design tool, but also to the overall program complexity.

The following chapters explain the theories and logic behind the software tool, highlighting some of the problems experienced during its development, and presenting relevant validations where possible. It is hoped that the knowledge and experience gained during the course of this research are well presented; this may entice the reader, and future students, to investigate some of the problems that have been encountered, and to follow up the 'areas for further work' that have been outlined. If that is the case, they will find themselves, as the present author does, asking more questions than they answer; for the subject, and for the industry in general, that can only be for the good. What is clear is that whilst this thesis may be a good starting point for the study of combat aircraft life cycle costs, there is still some way to go in the development of life cycle costing from an art to a science.

2 Literature Search

A literature search has many functions: it allows researchers time to familiarise themselves with the area of study; it gives the reader of this thesis an idea of the scope of the areas investigated, and the depth of knowledge gained; it allows weaknesses in the available literature to be identified and to be filled during the course of the research; and, during the course of the study, it will expose other researchers in the field who may be able to provide useful advice, guidance and data. Another useful feature of documenting the literature search is that it gives future developers of the models outlined later a head-start, so that many of the references found by scanning electronic databases can be re-visited or discarded, depending on their usefulness. Obviously, documents that have been rejected for this research may be very useful in later developments or other research studies.

Because of the scale of the research and the time limits imposed by budgetary and political influences, the references provided in this literature search are not intended to be exhaustive. It would be impossible in a three year programme to list all of the references available for all of the separate disciplines that make up this research - the snapshot provided should provide a solid starting point for the subjects listed.

2.1 Introduction

A literature search is an essential part of any research programme. In addition to the knowledge gained from studying the previous relevant information, the process helps to give a clearer definition of the research and determine the 'state of the art'. It provides a perspective on the contribution of the project to the literature available in the public domain, highlighting any gaps that are present and providing a focus. It prevents effort being wasted in repeating already documented research. Finally, the literature review provides the researcher with not only a list of what has been written, but also exposure to others performing similar work. This last point is very much underestimated by many people, as with an area such as costing, there is often much more to be gained through an unofficial chat than through formal correspondence. This is mainly due to the high level of sensitivity surrounding cost data, which can quickly be refuted if given verbally.

The project was broken down into the various relevant areas that were further investigated using the Cranfield library digital database, the NASA/AIAA Aerospace Database on CD-ROM from 1992-1998, the NASA and RAND Corporation on-line searches accessed from the Internet, which span many decades, and many reference books, journals and periodicals. One of the most useful search facilities is that made available by McGraw-Hill, the publishers of 'Aviation Week and Space Technology', which allows searches on specific aircraft programmes and other relevant keywords. There are now many costing pages published on the Internet, many of which provided useful data and references; a list of these has been provided at the end of this chapter.

2.2 Previous Work

So far, only two studies have been identified that specifically deal with the optimisation of conceptual aircraft designs for minimum life cycle costs, although there are undoubtedly more that are not documented in the public domain. The first was from a paper in 1989 published by Vicki Johnson[52] of NASA, and is briefly summarised below.

The paper explains the concepts of the 'window of opportunity', outlined in Chapter 1, and its applicability in both the military and commercial aircraft industries. The two major components of LCC stated are acquisition (Research, Development, Test and Evaluation (RDT&E) and production costs), and operating costs (direct and indirect operating costs (DOC & IOC)). Using LCC in the conceptual design process emphasises the importance of balancing the design between potentially conflicting parameters; e.g., low acquisition cost may be the result of low technology level and lead to high operating cost.

Johnson makes use of an existing conceptual design code - FLOPS (Flight Optimisation System), and utilises a separate module to evaluate LCC. Input to the system includes a baseline mission, aircraft data (geometry and propulsion), and economic assumptions. FLOPS is a multi-disciplinary system consisting of four primary modules: weights, aerodynamics, mission performance, and take-off and landing. It optimises the configuration using non-linear programming techniques to adjust the design variables, and the results are generated by parametric equations.

The design variables are; wing area, wing sweep, aspect ratio, taper ratio, thickness to chord ratio, gross weight, thrust (engine size), cruise Mach number, and maximum cruise altitude. There is the ability to vary technology level through complexity factors and the optimisation targets include minimum gross weight, minimum fuel burn, maximum range, minimum LCC, acquisition, direct operating, total operating costs, or maximum return on investment.

From the above paragraph, it becomes clear that the work is aimed at minimum LCC for civil transport aircraft, and this is one of the major differences between that work and the current research study. The LCC model incorporates several models for the acquisition and production phases and uses models developed by airlines for the operating cost modules. Three different classes of subsonic commercial transport aircraft were used, short, medium, and medium-to-long range, and the same economic assumptions were used for each aircraft. The design variables were not constrained to realistic values so that the full extent of the optimisation could be seen, although the mission requirements (particularly runway length) did help to maintain a certain amount of realism in designs.

A series of graphs and explanations is then given for each of the design variables optimised for minimum acquisition cost, take-off gross weight, LCC, DOC, and minimum fuel burned. A direct relationship is shown to exist between empty weight, acquisition cost and technology level. It is also apparent that DOC and LCC are very closely related, as the two are similar in magnitude for nearly all sensitivity studies. A study on fuel price is then conducted, along with a study on the lifetime of the aircraft and the residual value of the aircraft at the end of its useful life. The benefits of advanced technology are shown, with aerodynamic advances paying the most substantial dividends. Before aerodynamic advances are negated, cost penalties approaching 40% (on airframe R&D, airframe manufacturing, and airframe operating costs) are required.

Several problems were encountered during the study: convergence was not achieved if the initial parameter was too far from the optimum, resulting in the need to restart the program more than once; for all aircraft the taper ratio optimised to near zero. The latter problem suggests that the tip stall condition had not properly been addressed and that a tip-stall constraint had probably not been included.

Conclusions:

The addition of a LCC model to existing design modules is feasible and does give realistic results for transport aircraft. Results from the study show that varying the optimisation parameter has a definite effect on the aircraft, and that optimising an aircraft for minimum LCC produces a different aircraft than when optimising for minimum take-off gross weight, fuel burned, DOC or acquisition cost. In addition, results show that advanced technology can be worthwhile, even if it results in higher manufacturing and operating costs. The main differences found between this work and that proposed by the author are that this study concentrates on civil transport aircraft, the number of design variables considered is lower than the proposed study, and the present research aims to provide much more flexibility in the aircraft and engine design modules.

The second study was much larger and was initially conducted by the Vehicle Synthesis Branch, Flight Dynamics Laboratory, Air Force Wright Aeronautical Laboratories. It was a joint effort between Grumman and Lockheed, and was first revealed in a paper by Sternberger[97], who was the Modular Life Cycle Cost Model (MLCCM) project manager. The aim of the project was to develop a credible methodology/mathematical model to enhance design trade studies and predict LCC (RDT&E, Production and O&S) of advanced technology aircraft systems.

The MLCCM was developed as a modular computer code in three stages, with phase II being the computerised methodology for predicting advanced aircraft LCC to major sub-system level during the conceptual and preliminary design phases. The program uses a complete set of Life Cycle Cost Estimating Relationships, which provide the design engineer with a design-based engineering/cost tool for fighter/attack and cargo/transport/tanker aircraft. Sternberger gives some cost estimating relationships (CERs) used in the program, but not enough to make a real contribution.

Two other reports were also found which give some details of the MLCCM, those of Eberl[33], and Schwartz[85]. Both reports describe more of the trade studies that were performed using the MLCCM program and give some insight into further studies. It would appear that the MLCCM was highly regarded and was used by Grumman not only for the conceptual design stage, alongside the RAND model by Levenson[59] and the PRICE program[75], but also in the preliminary design stage, leaving only PRICE and proprietary grass-roots methods for the full scale design and production phases.

Unfortunately, the documents containing the full set of CERs and LCCERs are held only at Wright-Patterson AFB, and requests were originally refused on the grounds that they 'contain limited information and cannot be released to public'. The documents were eventually obtained, and were found to be very thorough, although they differ in approach from the methods that had already been built into the current LCC model.

Conclusions:

The MLCCM approach uses the slightly older direct product cost methodology, where the costs of individual aircraft systems are predicted directly from a number of parameters, or cost drivers. Recently, the concept of Activity-Based Costing (ABC) has become much more fashionable because "Appropriately applied, ABC provides a far more accurate portrayal of cost than previous accounting methods"[31], as discussed in section 2.3.1. As the current model has tried to use ABC wherever possible, many of the CERs contained in the MLCCM are not appropriate. However, where ABC models, or the necessary data to generate them, were not available, certain values, factors, and some models from the MLCCM were used.

The fact that the MLCCM was developed in the early 1980's, using data from the 60's and 70's, was another reason that many of those CERs were not used. Rather, newer methods were chosen that contained factors for new technologies, particularly for stealth and materials advances. It was also felt that the MLCCM methods, whilst calibrated and seemingly robust, did not provide as much visibility of costs as the ABC methods; it seems more intuitive that the time to complete a task should be calculated, from which the labour cost can be calculated using a known labour rate. The higher visibility of ABC makes checking of the models easier, and increases user confidence.

2.3 Areas of Study

2.3.1 General Costing Literature

Pugh[76] offers some valuable advice for the cost estimator, having been involved with the MoD Directorate of Project, Time, and Cost Analysis (DPTCAn) for a number of years. The advice varies from the common-sense - "The most important decision in the life of a project is whether to begin or not." - to the illuminating - "Do not be too eager to discount old data. Taken across complete projects, change is always progressive, with modest impact on total cost. e.g. avionics have racks, boxes, connectors, etc., as well as ever-cheaper micro-chips." There are many more useful tips to be learnt, given from the perspective of the estimator looking for absolute numbers, but it would appear that they offer sound advice in most respects. These were kept in mind throughout the literature search and were used as guidelines throughout the duration of the project.

Daniel[30], also of DPTCAn, promotes the need for data, and suggests that more time should be spent on the acquisition of data than on the development of models. He states that models, unless empirically based, are used as absolute values and therefore will not be as successful as they otherwise could be. This is a call echoed by the author, who would add that there should also be more visibility of cost data, and less division of the cost community. It is only through visibility of real cost data and models, and sensible, open discussion that real progress will be made in the area of cost estimating.

An excellent reference, and a necessity for those intending to create, or use parametric Cost Estimating Relationships (CERs) is the Parametric Cost Estimating Handbook[19]. It is an extremely useful guide, containing costing definitions, data collection and methods of improving models for the effects of technology, inflation, learning curves and production rates. The handbook also provides guidelines on the calibration and validation of parametric models, audit considerations, and two examples of the data normalisation process. Normalising means the conversion of data into a compatible format (consistent units, currency, accounting year) for the purposes of analysis.

The Cost Estimating Handbook continues to give guidelines on the development of CERs, when to use them, and a brief description of their strengths and weaknesses. The processes of curve fitting and regression analysis are described, including multiple and curvilinear regression techniques. "Goodness" of fit, and the statistical values of the correlation coefficient (R), and the coefficient of determination (R^2) are explained, as are the mathematics behind the statistical theory. The handbook concludes by giving advice and guidance on preparing cost estimates for hardware and software systems, and auditing and business application viewpoints. The appendices are also excellent, giving definitions of estimating terminology, work breakdown structures, more statistical explanations, examples of hardware estimating models, software estimation products, and a parametric estimating system checklist.

The US Department of Defense have also written a number of other Military Handbooks for specific cost-estimating tasks, such as that for Cost Engineering: Policy and Procedures[10], and the Life Cycle Cost Model for Defense Material Systems[6]. Both of are interesting in that they provide detailed procedures, data requirements and calculation procedures for the preparation of cost estimates for any system. However,

they are much too detailed for use in a conceptual design environment such as the one proposed for this research. They are still useful in that they give the structure and procedures used by the US DoD, arguably the world leaders in LCC, many of which have been used in the current suite of cost models.

Another useful DoD document is the Operating and Support Cost Estimating Guide[9]. This document is aimed at the preparation of in-service cost estimates, providing the conventions, definitions, methodology, and structure. One of the most useful sections of the document is the latter, the cost element structure, detailing where costs should be apportioned. This is a very significant contribution, as it allows comparison of one cost estimate with another, or estimates with real data. It has been found that because the guidelines were implemented relatively recently (1992), much data, even that produced by the DoD does not always conform to their guidelines. Also, the MoD use slightly different accounting and naming conventions for their cost capturing systems, which makes comparison of UK and US operation and support costs very time-consuming.

Activity-Based Costing (ABC) has been mentioned as the preferred method of estimating costs. As defined by Dean[31], "ABC is a special form of function cost analysis, where the cost of the functions required to bring forth, sustain, and retire the product are measured, as opposed to the functions of the product measured in value engineering". The emphasis is clearly shifted from the product parameters to the effort or resource parameters, allowing greater visibility of the manpower, materials, overheads, profit, and other cost components. This permits increased understanding of cost in all stages of the product life cycle, which should allow increasingly effective management to reduce overall LCC. Greater cost breakdown gives increased flexibility in the modelling process, especially when representing advanced technologies.

With time, the available literature on LCC is expanding as the emphasis on cost reduction and 'Cost as an Independent Variable' increases. Three other noteworthy references are mentioned here. The first, by Cole[25], draws attention to the cost of 'Manpower, Personnel, and Training' (MPT) for the operation and support phase of an aerospace system. This backs up the decision to use an activity-based cost model for this programme, allowing changes in the organisation of the operator to be represented without having to alter the whole cost model. The paper presents some useful data, showing that MPT costs account for approximately 48% of the total operation and support costs for military aircraft, which themselves form between 40 and 60% of total LCC. Cole praises efforts to try and standardise accounting methods, but also advocates the further development of operation and support cost data and models, since this is the least understood phase of the aircraft life cycle.

Yelverton[107] describes the development and application of a 'technology-insertion' LCC model, with particular reference to the avionics sector. The premise is that an upgrade in a particular technology will realise cost savings over the remaining life of the project for a number of reasons, the main being the reduction in maintenance support for newer systems. Mathematical formulae are given for the calculation of the cost savings available from new technologies, but require many inputs to be used successfully. However, the ideas and methods described are well presented, and could be applied to an LCC model, and this could be future development of the present LCC model.

Finally, Comey *et al*[27] present a paper discussing trade-off studies that were performed in the late 1970's with an interactive engine/airframe LCC model. The models appear to be very similar to those proposed for this research programme, although they are heavily weighted towards the propulsion of the aircraft. In addition, the operation and support phase costs are simplified to a large extent. The trade studies performed are also aimed primarily at engine technologies, and are intended to show that there is a need for airframe/engine integration to be considered more closely during the early design stages. As this is now common practice, and the cost models used are not given, it was felt that this paper did not live up to its full potential, but is a useful addition nonetheless.

2.3.2 Airframe Layout

For each major aircraft sub-system, a list of influential conceptual design variables has been drawn up, together with relevant information for the determination of mass and costs for each sub-system. Many of the design variables are determined by external requirements, and are not included in the trades, even though they will have an effect on the LCC. Thus, the options discussed are not exhaustive and are meant to show some of the areas of research that have been covered. Many of the ideas and concepts came from conceptual design texts, such as Raymer[78], Nicolai[71], Mattingly[65], Heunecke[50], and Whitford[102], as well as from studying past, current, and future aircraft designs. It should be noted that along with the four traditional areas of aircraft design (airframe, engines, avionics, and systems), software development and reliability and maintainability (R&M) have been included, as these areas are becoming increasingly important in the definition and design of aircraft systems.

In previous MVO studies, like those by Lovell[63] and Siegers[88], the fuselage shape was mainly determined by the summation of lengths of the individual items, e.g. cockpit, radome, etc., from the fineness ratio required for reduction of wave drag, and from 'area ruling'. This was thought to be too complex for this research due to the large number of variables required. Instead it was proposed that much simpler methods would be used, with checks to ensure that the fuselage was of an appropriate shape (mainly determined by the drag model), and that it contained enough excess volume for the storage of fuel not carried in the wing.

Wing parameters, such as area, aspect ratio, sweep angle, t/c ratio, and taper ratio all affect the mass and production costs of the wing. However, many of them will be set by mission requirements or adjusted automatically by the optimiser, adjusting the wing to give the best compromise between aerodynamic and structural requirements. As the configuration of the wing is itself not a specific problem in this design methodology, no specific references are given here.

The configuration of the empennage can have a significant impact on the design and will determine aircraft size and performance to some extent. As the conceptual design of the aircraft was to be kept simple, the sizing of the empennage components was examined. After consulting standard aircraft conceptual design texts, such as Nicolai[71], Raymer[78], and Roskam[82], it was decided that a simpler method could probably be devised using parametric equations. In an attempt to become familiar with parametric methods and the problems involved, the empennage sizing methodology described in section 4.2.5 was developed.

As with the aircraft conceptual design and sizing procedure, there are few references aimed at the level of complexity envisaged for this programme, and only one reference was found that gave accurate results from similarly simple methods. Sieron *et al*[93] present a number of sizing procedures and design data for the conceptual design of all types of aircraft, starting right from the basic aircraft requirements of range and payload. It was decided that although many of the methods seemed to give good results, that they could easily be replaced with bespoke algorithms, without resorting to the interpolation of graphs, as is the case with that paper. It is still a useful reference for those intending to produce very simple sizing methods, or students.

Aerodynamic and aero-elastic tailoring have become possible using advanced technologies, although many of these are in the very early stages of design. Obviously, if cost benefits are indicated, then the new technologies should be included in future designs, but as yet sufficient cost data has not become available to allow realistic trades to take place. It is also questionable whether levels of detail such as this should be included in a relatively simple conceptual design tool, but it has been mentioned to illustrate that new and complex technologies were investigated.

Many papers have been published on the cost estimation of aircraft airframes, the most famous of these being by Levenson *et al*[59] of the RAND corporation. It was one of the first documents to provide a full set of cost estimation equations for a wide range of aircraft using statistical analysis in the form of Cost Estimating Relationships (CERs). Since then, many other similar documents have appeared, acting more as an update to the original, rather than as new approaches to the problem of estimating the cost of aircraft airframes. Hess and Romanoff[43], also of the RAND corporation, and Burns[21], of Vought Aircraft Company, provided revisions in December 1987 and May 1994 respectively. The latter reference is particularly useful, as it concentrates on some of the new technologies that will be incorporated into the next generation of combat aircraft.

Most of the airframe cost estimation methods investigated thus far are based on empty weight, among other variables, as the main driver of airframe cost, and this would seem to be a firm foundation for work to begin on, especially when considering the 'top-down' approach. In addition, the CER format of the estimating method is an ideal basis for optimisers to work with, as it avoids discrete step changes and gives a smooth change in cost as the input is varied. It is also a very powerful tool, particularly in this investigation, as it allows cost/parameter trades to be accomplished very effectively. Lovell's design synthesis[63] also incorporates a cost estimation module, but it was felt that it would be easier to start afresh with a new cost model in a consistent modular format, rather than trying to borrow sections of code from other programs.

2.3.3 Airframe Materials

Many new combat aircraft are being designed from the start to incorporate advanced materials because of the advantages that can be gained from their use. Resetar *et al*[79], of the RAND Corporation, have produced an excellent paper on advanced airframe structural materials. This, together with the knowledge gained from the Department of Aerospace Technology (DAeT) short course on composite materials[62] and from journal articles, like that by Belbin[15], has provided much useful data and understanding on the effects of composite materials.

Resetar identifies, describes, and quantifies the cost effects of structural materials that will be incorporated into aircraft becoming operational early in the next century (aluminium, aluminium-lithium, steel, titanium, graphite/epoxy, graphite/bismaleimide, and graphite/thermoplastic). The first half of the report is a primer for advanced aircraft structural materials with a strong emphasis on polymer matrix composites. The second half of the report contains both cost data and a cost estimating methodology sensitive to material mix. For each material type, separate cost factors are presented for two time frames, the late 1980s and the mid-1990s, and for the following cost elements: non-recurring engineering, non-recurring tooling, recurring engineering, recurring tooling, manufacturing labour, manufacturing material, and quality assurance. These factors are based on data obtained from Boeing, General Dynamics, Grumman, Lockheed, LTV Aerospace, McDonnell Douglas, Northrop, and Rockwell International.

Resetar states that the new materials and processes that are most likely to be utilised in aircraft structures will be aluminium-lithium, superplastically formed/diffusion bonded (SPF/DB) titanium, and polymer matrix composites (for example graphite/epoxy)[a]. The other categories of composite materials (ceramic matrix, metal matrix composites and carbon/carbon composites), powder metallurgy, and titanium-aluminium alloys do not constitute substantial proportions of 1990's aircraft, although these will certainly become more available and more widely used in future. These materials either offer properties that are too specialised for widespread application to aircraft structure, or are behind polymer matrix composites or other metals in terms of development.

Two important categories of polymer matrix composites are thermosets and thermoplastics. Most composites used up to now on production aircraft have been thermosets, although continuous fibre reinforced thermoplastics have recently received considerable attention. The important distinction between the two is that thermoplastics, unlike thermosets, may be re-heated and reshaped, offering enormous potential for processing ease. However, little public-domain engineering data exists regarding advanced thermoplastics, creating uncertainty in both the design and fabrication processes.

Although composites offer several performance advantages, including reduced mass, reduced number of fasteners, increased corrosion resistance, reduced radar cross-section, and the potential for extended life, this increase in capability comes at a price. The cost data collected by RAND clearly show that composite materials are more expensive than aluminium on a cost-per-pound basis. In terms of overall recurring cost per pound[b], composites can be over twice as expensive as aluminium, for the following reasons:

- properties of these engineered materials frequently need to be verified
- tooling requires sophisticated design and durability because of the severe thermal cycling experienced in the autoclave
- current hand-lay-up techniques are very labour-intensive
- quality assurance processes are in their formative stages and are time consuming

---

[a] Aluminium, titanium, and steel will continue to be important materials in future aircraft structural applications.

[b] Total recurring cost includes recurring engineering, recurring tooling, manufacturing labour, material costs, and quality assurance.

While these reasons may be true for thermosets, thermoplastic production methods are much simpler, and it is for this reason that they will certainly become more common in future. Current figures indicate that the hours per pound for both conventional metal materials and composite materials will decrease. In addition to the structural cost factors described above, Resetar also presents an approach for estimating overall airframe costs. The method, which is suitable for use in a project's conceptual design stage, when little detailed design information is available, takes into account not only the cost of the airframe structure, but also the cost of the airframe subsystems (e.g., electrical, hydraulic, environmental) and the cost of final assembly/integration. Reduced to its simplest form, the method applies weighted material indexes to baseline cost estimating relationships that utilise aircraft empty weight and maximum speed as the principal explanatory variables. This is essentially another revision for the original work carried out by Levenson *et al.*[59].

An example is given where the method was applied to two hypothetical combat aircraft to get some feel for the net effect of composites on overall airframe costs. One of the aircraft weighed 13,000 lb. and had an all-aluminium structure; the other weighed 11,700 lb. and was based on the assumption that only 5,200 lb. of graphite/epoxy would be required to replace 6,500 lb. of aluminium (a 20% weight saving). A comparison of the costs of the two vehicles indicated that the substitution of graphite/epoxy for half the aluminium in the structure would increase non-recurring airframe costs by about 3 per cent and recurring airframe costs by roughly 35 per cent[c]. However, acquisition cost is by no means the only criterion for deciding whether to utilise advanced materials, such as composites. Operations and support costs must also be considered and the net life cycle cost effect then balanced against the performance advantages; it is towards this process that much of the remainder of this thesis is directed.

Warwick[101] states that Lockheed's objective with the F-22 Advanced Tactical Fighter (ATF) was to use composites for 40% of the aircraft structure by weight and thus achieve a structural weight reduction of 25% or better, compared to aluminium. The goal of 25% weight reduction was achieved, but with a usage of only 35% composites by weight. The reason for the shortfall was cost: in the trade-off of cost against weight saving, composites did not always win out over aluminium. The same cost/weight equation also seems to be limiting the application of composites in commercial airliner production - at least until material costs come down and the manufacturing technology matures. Of the 35% composite structure on the F-22, 25% is bismaleimide, and only 10% are thermoplastics, which are very much in their infancy in terms of structural usage, and therefore more costly. A discussion of damage tolerance and reparability is given, which highlights some of the problems encountered with composite materials and advocates the use of some of the advanced aluminium alloys currently available.

A paper from 'New-Tech News'[8] outlines many new production techniques being used in airframe production, including super-plastic forming and diffusion bonding, both of which will become increasingly important, it is claimed. By combining the two techniques, it is possible to produce components to their approximate final contour

---

[c] This conclusion assumes that the start point was an all-aluminium aircraft and that the only weight saving was that from material substitution. Different material mixes and aircraft re-sizing are ignored.

using minimum material and minimum machining. The components are then formed to achieve the final dimensions and are joined in a single operation. In addition to cutting down on production steps, this approach also simplifies assembly, in many cases doing away with sealing and joining work.

In aerospace industries these techniques can result in cost reductions (presumably recurring cost) of up to 50% and weight reductions of 35%, although most applications involve the use of titanium alloys, which are more expensive than aluminium alloys. The use of aluminium poses problems, because its impervious oxide layer prevents diffusion bonding from taking place efficiently. New techniques are being developed to reduce the formation of the oxide layer using a process similar to that used for etching printed circuit boards, making the use of aluminium more accessible to the designer and increase the opportunities for the use of this method, which will further reduce costs.

Incremental forging is also discussed, and is described as a pressure-forming technique, where the work-piece is formed by a tool whose contact surface is substantially smaller than the work-piece to be processed. In each forging operation, a number of strokes are made which successively cover the entire work surface. In this way, a simple geometry with a relatively small volume (e.g. a plate) is transformed by means of material flow, material displacement, and the elevation of ribs and lands into a forging blank. This, despite the need for final working, permits a reduction of material and machining as compared to conventional production methods. Thanks to the incremental tooling method, the plastified zone is reduced, and this goes hand in hand with a reduction in the shaping force. As with many other references, this paper presented many good tips for cost-saving, but little in the way of actual figures for the costs of the items being produced, or of the savings to be made.

Much research has been aimed at the reduction of corrosion, and airframe life extension in an attempt to save costs. An example of a recent paper is that by Chester *et al*[23], which describes the effects of water displacing corrosion preventatives on different parts of the airframe of military aircraft. Whilst the paper is useful, and would certainly be of interest to those involved in airframe maintenance, it is not significant to this research for two reasons. Firstly, the costs of corrosion prevention and maintenance are not specifically broken down in any of the reliability and maintainability data available to the author; it is therefore impossible to quantify the savings possible from this activity. Secondly, this level of detail is not captured in the data for the conceptual design models. Indeed, it is difficult to see how this kind of preventative maintenance would impact on the design of the aircraft, and should be considered as a separate cost-saving activity and trade study performed during routine operation of a designed aircraft.

One of the most important aspects of the use of composite materials is the mass saving available, and the subsequent downsizing of the aircraft that is possible. There are many references available detailing projected mass savings, such as that by Burns[20], but few that give averaged reported figures. The most useful reference found in this respect was that by Foye[35], a highly respected researcher and author, who has been involved in the study and application of composite materials to aero-structures since the mid-1960's. After describing some of the history of composite materials usage for each structure type, including some amazing facts ("The first composite primary wing structure was built in 1945"), the final section of the paper gives a table of average composite cost and

weight savings. As the cost effects are already dealt with in the airframe costing methodology, these were ignored, but the main facts gleaned were that composite tail surfaces resulted in 30% mass savings, whilst the effects for wings and fuselages are 16% and 22% respectively.

2.3.4 Propulsion Options

One of the most important aspects of any aircraft development is the design and integration of the propulsion system. It is probably the most complex single system on the aircraft, and, apart from the airframe, is the single most expensive component that will be used. In previous MVO studies, little emphasis was placed on the design and integration of the engine, with engine performance data being taken from digital data sheets. These programs treated the engine as a 'rubber' component, with simple equation used to scale the size and mass according to changes in the thrust value. It was decided that this research would develop and integrate a new way of modelling engine performance, giving the designer increased flexibility, whilst still keeping the design at the conceptual level.

There are many thermodynamics texts available, although it was quickly realised that in order to get realistic results from a thermodynamic model, it would have to be quite complex. This, in turn, would require a large amount of coding, with all the problems that brings, and the difficulty in validating such a complex code. It was thus decided to try to find an existing engine model that could be used for the purposes of this research, rather than starting from scratch. Several methods were investigated, which were not suitable or not easily available, including the Cranfield School of Mechanical Engineering model, Turbomatch[72]. This code is not suitable because of its size and complexity, although it is said to be very accurate. Similar codes are used by DERA and Rolls Royce, but neither of these were available, and would suffer the same problems as Turbomatch. Some of the basic thermodynamic texts were investigated, such as Cohen, Rogers & Saravannamuttoo[24], but it was the ONX and OFFX suite of thermodynamic codes, written by Mattingly[65], that were chosen for use in this research.

The advantage of these models is that they are written to help undergraduate students complete gas turbine thermodynamic analysis problems, and operate at the component level, rather than the sub-component level. Simply, this means that components, such as the compressor, are treated as single items with a prescribed pressure ratio and efficiency, making the analysis significantly simpler than complex industry methods, whilst still allowing the user to make realistic performance predictions, and alter the design of the engine cycle. The models can be used for the analysis of all two-shaft gas turbine engines, as well as ram-jets, although their use is limited to the normal engine types for use in this research. The other advantage of using ONX and OFFX is that Mattingly kindly provided the source code for the menu-driven stand-alone program, reducing the amount of programming required by at least an order of magnitude. After investigating some of the more complex installation loss calculation methods, such as that described by Goldsmith and Seddon[37], it was decided that the method suggested by Mattingly would probably be the most appropriate. A description of the thermodynamic and installation loss models and the changes made to incorporate them into the overall design synthesis is given in sections 5.2.2 and 5.2.3.

Now that a thermodynamic model had been found, an estimation method for the size and mass of the engine was required, as the method used by Lovell and others was invalidated by the move away from the tabular data sheets. Two separate methods were discovered, by Annear[4] and Gerend and Roundhill[36], but before the methods could be validated, a large amount of engine data was required. As was discovered, engine data well-guarded, and public-domain engine cost data is almost non-existent. This led to a long process of data collection and gathering while the development of the thermodynamic models was ongoing. The main source of data was Jane's[39], but much data was gathered directly from manufacturer's literature and from military and industry sources. It soon became clear that the methods mentioned above for the estimation of engine mass and dimensions were not as accurate as their authors were claiming for engines not contained in their databases, this trend becoming particular apparent for newer engines. This is probably caused by the dependence of those methods on many detailed engine variables, rather than fewer representative values. Therefore, continuing a theme suggested by Whittle[103], it was decided that a new method would be developed to predict the mass and dimensions for all types of turbofan engines, including those with re-heat. The results of this methodology development are given in section 5.2.4.

Several advanced propulsion technologies were investigated with the aim of including their effects on the design of the aircraft, engine, and life cycle cost. One of the concepts was for thrust vectoring nozzles, and multi-axis thrust-vectoring, or multi-function, nozzles (MFNs). They offer significant potential performance advantages for an aircraft for the following reasons:

- Fuel savings from improved aircraft cruise lift-to-drag ratio and possible reduced tail size leads to lower fuel burn, reducing aircraft size and fuel costs.
- Superior ground performance can lead to shorter runways and lower base maintenance, or a smaller wing (aircraft) for a given ground run.
- Increased control authority can reduce single-engine approach speeds, lowering peacetime attrition costs and reducing undercarriage size and maintenance.

However, MFNs do have drawbacks, such as:

- The relatively new technology leads to high development costs & high risk.
- Increased nozzle and engine installation mass will reduce some of the benefits outlined above, and the rearward shift in c.g. may introduce control problems.
- More actuators and flight control software will cause higher development and flight test costs, and maintenance costs may increase.

Several papers on advanced engine concepts were studied, the first of those being by Herrick[41]. The paper advocates the use of multi-function nozzles for the reasons outlined above, as well as many advantages in aircraft survivability and effectiveness. Multi-function nozzles (MFNs) add pitch and yaw thrust vectoring, thrust reversing and the ability to optimise the nozzle area ratio, to the current nozzle capabilities. These new propulsive forces and moments allow the aircraft to operate at higher angles of attack, higher angular rotation rates and accelerations, and quicker flight path turn and deceleration rates. A fundamental change in combat aircraft design philosophy will be possible by allowing the MFNs to augment or replace the tail surfaces. Herrick claims

that aircraft affordability will be increased through reduced size and cost, better fuel efficiency, less stringent basing requirements, lower peacetime attrition, and reduced pilot training. Also, survivability will be enhanced by reductions in detectability, susceptibility, and vulnerability; and mission effectiveness will be augmented by increased air combat capability and reduced ground attack response time.

MFNs may offer all of these possibilities, but there are drawbacks in the form of the increased control systems requirements and software required for both the engine and flight controls, if maximum advantage is to be gained. Higher maintenance is foreseen, as the flaps and seals of the 3-D nozzle will need to be replaced more frequently, and the software and mechanical actuators will require maintenance not normally necessary. Brian Rowe, Chairman Emeritus GE Aircraft Engines (GEAE), indicated[83] that the cost of the General Electric Advanced Vectoring Exhaust Nozzle (AVEN™) was only 15% more expensive than an ordinary nozzle. He did not give any details of the O&S costs for the new nozzle, nor for the cost of the increased control software.

Details of the AVEN™ nozzle are given by Mishler and Wilkinson[68] (Figure 2-1), along with several other developments being studied at General Electric. These include Performance Seeking Control systems, Variable Stall Margin, Rapid Thrust Modulation concepts, Integrated airframe/engine monitoring systems, More Electric Aircraft (MEA) concepts, and integrated thermal management for fuel/oil/hydraulic fluids. Of these, thrust vectoring is the most advanced, although several of the others should soon be ready for flight testing.

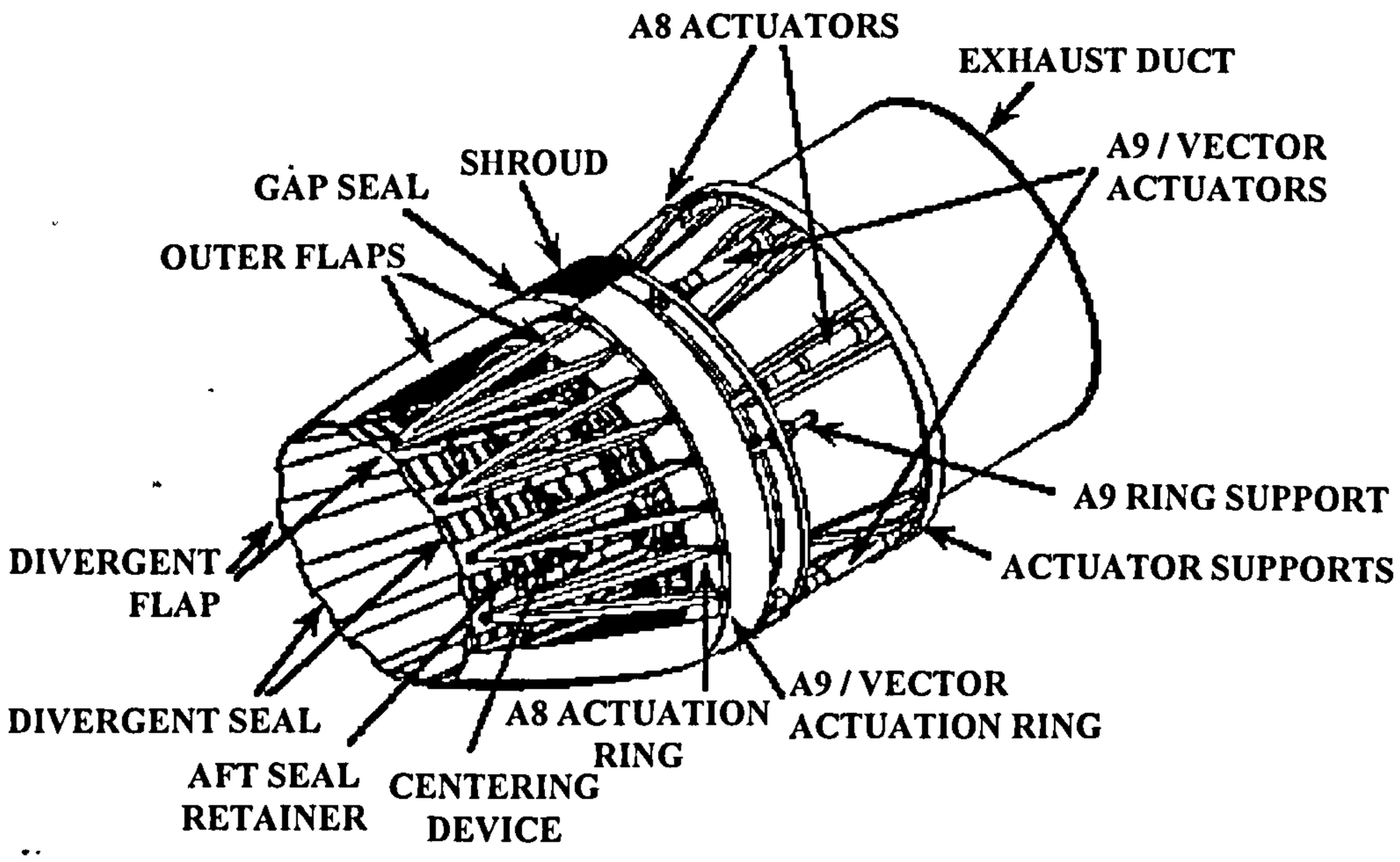


Figure 2-1. General Electric AVEN™ Nozzle.

GEAE has been working on thrust-vectoring exhaust nozzles for over three decades, with past systems including the Block and Turn vectoring system, the three bearing vectoring system and the Augmented Deflector Exhaust Nozzle (ADEN). They have been involved in many of the thrust-vectoring demonstrators, including the F-18 High Angle of Attack Research Vehicle (HARV), the F-15 S/MTD flight program, the X-31, and also provide the YF-120 engine that powers the F-22 ATF. The latter system, like

the F-15S/MTD and the F-18 HARV, is a 2-D system and it is this difference that distinguishes the AVEN™ design from 2-D systems, providing both pitch and yaw control in a light, compact, and relatively simple design. The nozzle has been tested at vector angles of up to 20° and at transient vector rates of more than 60°/s, providing more than 6500 lb. (28.9 kN) of vector force.

Liston and Small[60] discuss some of the concepts under study at Wright AFB, which include Active Flexible Wing, Multi-Axis Thrust Vectoring, and Forebody Vortex Control, among others. They have also studied the GEAE and PW nozzles, the level of airframe/propulsion integration and its effect on the weight, cost and aerodynamic efficiency of the aircraft. The three stages of nozzle integration are: (1) An engine-mounted nozzle; (2) An airframe-mounted, easily removed nozzle that does not carry aircraft loads; and (3) A structurally integrated nozzle which is part of the overall structure of the aircraft and carries internal gas path loads as well as external airframe loads. According to their studies, the progression 1 - 2 - 3 gives lighter weight, lower cost, and greater aerodynamic efficiency. Specifically for the AVEN ™ nozzle, they claim a 2-3% improvement of installed thrust minus drag for many flight conditions, due to the inclusion of the A9 (Vector Actuation) ring. They include Pratt and Whitney's (PW) F100 P/Y Balanced Beam Nozzle (BBN) and give weight penalties for both the nozzles of 200-300 lb. and costs of less than US$350,000. Mr. Rowe, when confronted with these figures, said that this weight was much more likely to be for a 2-D nozzle, and that the weight of the AVEN™ nozzle was "substantially less" than these values, although he would not elaborate further.

Kitowski[57] and Mace & Nyberg[64] outline similar programmes and studies at McDonnell Aircraft and General Dynamics respectively, and confirm some of the claims outlined above: MATV can reduce tail sizing, with subsequent benefits; increase survivability and combat effectiveness and improve both of these with increased integration of the nozzle system. Kandebo[54], interestingly, also claims a weight penalty for the AVEN™ nozzle of 200-300lbs, and states that thrust vectoring can reduce take-off rolls for the F-15 (presumably the F-15 S/MTD) by more than 35%. He states test data as having achieved slew rates of more than 45°/sec, maximum deflections of 17°, thrust ratings of up to 28,000 lb., and that the nozzle had passed its 2Hz tailplane emulation test with no extra cockpit control - nozzle control is built into the Flight Control System (FCS). The PW nozzle ran for 9 days on an F100-229 engine. This nozzle can rotate up 20° at 45°/sec. No further cost or weight data was given.

This technology would certainly be worth investigating, but would have to be performed as a separate detailed study, as the change from non-vectoring to vectoring is discrete, and would therefore be outside the control of the optimiser. Another problem with the integration of such a radical technology into the design synthesis, is the lack of data available for the system, especially with regard to R&M. In addition, the flight mechanics of an aircraft utilising such a system would be significantly different to a conventional aircraft, which would dramatically alter the point performance equations. This, coupled with the fact that the cruise drag benefit could not be modelled fully, due to the absence of a specific 'trim drag' module, led to the use of thrust vectoring being ignored in this development of the synthesis. As mentioned above, it is an important future technology, and an area that requires further work.

Several papers were investigated for their discussion of the impact of engine technology on combat aircraft. The first, by Hodder and Simm[44], discusses the effects of improving technology values (normally turbine inlet temperature and turbomachinery efficiencies) on aircraft performance. They point out that such technology improvements can produce significant reductions in aircraft mass while meeting a given level of performance. Alternatively, performance levels can be increased for a given aircraft mass. Empty mass savings of between 2% and 6% are quoted for representative Combat Air Patrol (CAP) and intercept missions for a turbine inlet temperature increasing from 1850K to 2000K. Raising the turbine inlet temperature by a further 150K raises the above mass improvements by half, showing that the benefits diminish with increasing temperature. Increasing the turbine inlet temperature has the effect of increasing the specific thrust of the engine, i.e. for a given air mass flow rate, a higher thrust is delivered. At current levels of airframe technology, increasing engine thrust/weight from 10 to 15 provides an empty mass saving of 15%, and increasing it to 20 gives a further 6% saving. A similar, but smaller, effect can be achieved by increasing the efficiency of the turbomachinery components, or enhancing the turbine cooling effectiveness. Studies with a variable cycle engine were also performed, but are not summarised here, as this technology is still very much in its infancy.

Skira[89] performed similar studies on the cost and performance improvements available under the DoD/NASA/Industry Integrated High Performance Turbine Engine Technology (IHPTET) Programme. The goal of this programme is to demonstrate the technologies required to double engine thrust/weight ratios and cut fuel consumption by 40%. The meeting of these goals is proposed primarily through improvements in materials technology and aerodynamic advances, leading to smaller, more efficient engines. The reductions in component mass using advanced materials will return much of the thrust/weight increase, whilst increasing turbine inlet temperature will also play a large part. This will lead to a higher cost per stage (40%), but the number of stages is predicted to reduce significantly, reducing engine purchase prices by 30%. Aircraft acquisition costs are also predicted to drop by 30% due to the down-sizing process, and the reduction in fuel burn will also contribute to cost savings. In-service costs are also predicted to be lower due to lower part count and an overall simplification in the design, but figures are not given for this.

Herrmann and Biehl[42] performed a comparable study, for an aircraft very similar to Eurofighter 2000. They place more emphasis on the life cycle cost savings available for the advanced technology engine, and draw the following conclusions. With an advanced design developed to fit an existing aircraft, the total costs increase, as production and in-service costs are not much affected, but the cost of developing the new engine is added to the LCC of the existing engine. For a given level of performance, the advanced engine, even with its larger development cost, has a lower acquisition cost than the existing engine, mainly as a result of the reduced cost of manufacturing the smaller engine. The simplified design helps both production and Operation and Support (O&S) costs, leading to savings on top of those achieved from fuel costs, which are the most significant. A similar study is done by re-sizing the aircraft with advanced engines for the same size and performance, with less-pronounced, but predictable results. Sensitivity analyses predict the point at which the engine development programme becomes viable, considering both acquisition and LCC.

From these references and others, it became clear that the costing methodology for the LCC models would need to capture the effects of both engine size and technology, which is not captured from thrust alone. This is because there are essentially two ways of generating higher thrusts; build a larger engine, or increase the technology level of an existing engine. The cost implications of the two methods are significantly different. Burns[21] gives some very simple CERs for low-bypass engines, although they have been "reverse-engineered" in that the engine costs for many aircraft were established by subtracting the calculated airframe cost from the aircraft recurring flyaway cost. The values obtained were normalised for the accounting year and correlated against engine thrust. This method was rejected for the reasons explained above.

Birkler *et al*[16] present a useful methodology for the development and production costs and time of arrival of US military gas turbine engines. The values are based on static sea-level thrust and the engine technology parameters Mach number and turbine inlet temperature. The correlation for production costs gives the cumulative average price at the 1000th unit, requiring a complex conversion back to first unit cost by using an assumed value for the learning curve before the true engine cost can be calculated. An equation for time of arrival of engines, based on an extrapolation of engine technologies, is also given, and makes use of engine thrust/weight ratio, turbine inlet temperature, and specific fuel consumption. The models seem to be well constructed and the statistical analysis is very thorough, but it is unfortunate that thrust and technology factors have been combined in the same equations. The accuracy of the models is difficult to check, except for the time of arrival prediction, which displays much poorer correlation that claimed in the RAND report, with an R^2 value of 0.28 instead of 0.96. It must be pointed out that the data collected during the dimension and mass estimation method development was taken from open literature and could be erroneous for some engines. However, the discrepancy for time of arrival exceeds that which one would expect from a few incorrect data points. The development cost model was checked against engine development cost values in a National Audit Office Report[28]; the accuracy was found to be good, as discussed in section 7.3.2. No further comments on accuracy can be made due to a lack of data for engine production costs.

The Modular Life Cycle Cost Model Report[96] also contains algorithms for the estimation of engine development costs based on several parameters, including air mass flow rate, compressor pressure ratio, maximum dynamic pressure, and a time-of-arrival value comprising engine thrust/weight ratio and turbine inlet temperature. The model also requires a value for the time (in years) for the development of the engine through to engine qualification. This value must be supplied "by a knowledgeable engine development engineer because various engine manufacturers will require differing development times, based on company experience and development approach". As a formula for the calculation of development time is not given, and access to such data is not available to the author for a wide range of engines, this method of engine development cost estimation had to be rejected. The engine production cost method given looks much more promising, with costs for the 1000th unit predicted using static sea-level thrust, a pressure term comprising the maximum compressor pressure ratio and sea-level maximum mach number, and a time of arrival value. However, when checked against the small amount of data that was available, accuracy was poor and correlation against the methods described above gave disappointing results.

Bailey and Overton[14] present a method for estimating individual aircraft parts using a comparative cost estimating approach. The Assistant Cost Estimator (ACE) program is an artificial intelligence system that uses object-oriented programming techniques incorporating rules and algorithms to derive cost from a database of existing parts. As a vast database of individual parts is required for this concept to be effective, and as the author had neither the time to analyse, nor access to such detailed information, this particular approach had to be ignored.

Information on operating and support (O&S) algorithms for aircraft engines has been located in a paper by Meitzler[67], which details the model developed by the Air Force Acquisition Logistics Centre, Wright Patterson Air Force Base. The model was developed to aid the acquisition decision-making process by providing a simple, computerised menu-driven program that would quickly provide support cost comparisons among engines. The model consists of twelve equations intended to be used with a three-level maintenance environment. Most of the algorithms are independent of each other although some use common terms. The twelve main equations cover Base Labour, Base Material, Depot Labour, Depot Materials, Condemnation spares, Second Destination Transport, Recurring Support Equipment, Spare Engines, Initial Spares, and Pipeline Funding Spread. Reliability, maintainability, spares levels, and other complex data is required to feed into the equations, together with details of the number of bases, distance between them, etc. Because of this, the method is well suited to an analysis of an existing engine programme, but is not useful for the conceptual design phase, when much of the detailed information is not available.

2.3.5 Avionics

For modern combat aircraft many of the avionics systems, and hence costs, are specified in the system and mission requirements, and are beyond the control of the airframe designer. For example, the LCC for the radar is likely to be a direct result of the mission requirements, and not affected by the design or cost of the rest of the aircraft. Interaction between the avionics and airframe is a difficult area to model, although it is well known that reducing vibration and increasing cooling to sensitive avionics components can deliver substantial avionics system cost savings. Increased cooling of the avionics suite has definite avionics cost advantages, as avionics reliability (MTBF) is increased, leading to lower maintenance costs. However, it requires greater cooling air mass flow rates, requiring larger/more ducting, increasing ducting mass and cost. If the bleed is taken from the engine, as is normally the case, the higher bleed rate gives lower efficiency, resulting in higher SFC and lower available thrust for a given engine type. Vibration can be reduced by the use of anti-vibration mounting mechanisms, which can be costly, and will add weight to the aircraft. The use of this heavier and more complex mounting system will reduce costs by improving reliability of the avionics suite, but the increased mass and aircraft size may negate any benefit gained from reducing avionics O&S costs. Only through detailed modelling can these changes be accurately captured and implemented. This level of detailed modelling requires months of effort to investigate complex cooling and avionics mounting components, and has therefore been disregarded at this stage in the development of the LCC methodology.

The inherent differences between avionics and other systems have caused many problems during the course of the research, not least because avionics and software are increasing in complexity and capability at a rate unmatched by other aircraft systems. The introduction of integrated avionics architecture, as opposed to the older federated systems, has completely changed the way in which avionics operate, leading to a reconfigurable, distributed network of high-performance computers, instead of single boxes for single functions. These changes, and the rapid advances in computers themselves, make cost analysis of avionics systems particularly challenging. There are several benefits for modular systems:

- Redundancy without extra boxes, giving higher reliability/fault tolerance.
- Allow carefree positioning of computer components, increasing design flexibility, maintainability, and survivability, leading to lower costs.
- Allow commonality of components, reducing the spares requirement and O&S costs.
- Allow use of self-test, reducing O&S costs.

However, modular avionics systems and their use can have pitfalls:

- Carefree positioning of distributed systems can result in long databuses, increasing costs and reducing survivability.
- Self-test equipment can increase the amount of avionics on the aircraft, causing more heat and lower reliability/increased cooling. There is also the danger of self-test failure, causing aircraft to be grounded unnecessarily.
- There is high risk and cost for development of new technologies such as integrated modular avionics systems.

Having investigated a small amount of the available literature, it soon became clear that it would not be feasible to realistically account for avionics system costs within the scope and time-scales of this research programme. In order to give future developers of the synthesis models a head start, and to provide comment on some of the useful literature investigated for interested readers, the following references are recommended.

Spitzer[95] provides an excellent guide to modern avionics systems, giving useful information on: avionics system requirement; data buses, crew interfaces and power; fault tolerance; reliability and maintainability; architectures; aircraft avionics packaging; hardware assessment and validation; software design, assessment and validation; and avionics cost estimation. It is this last section that has been studied most closely. The designer of the avionics system plays a central role in determining its cost. The segments of LCC shown overleaf are essentially the same for both civil and military programmes: acquisition cost, operating cost and support cost. Descriptions of these three items are given, along with an explanatory diagram, shown in Figure 2-2. A set of equations is given to estimate the LCC by approximating the costs of the two parts that can be significantly influenced by the designer, i.e. the investment costs (acquisition costs), and the O&S costs. These equations are based more on an accounting viewpoint rather than on CERs, in that the cost per unit is expected to be known, which is not the case. The cost analysis section of this particular reference is, therefore limited, but the book does provide a detailed view of modern avionics systems, especially for those who have had negligible experience with such systems.

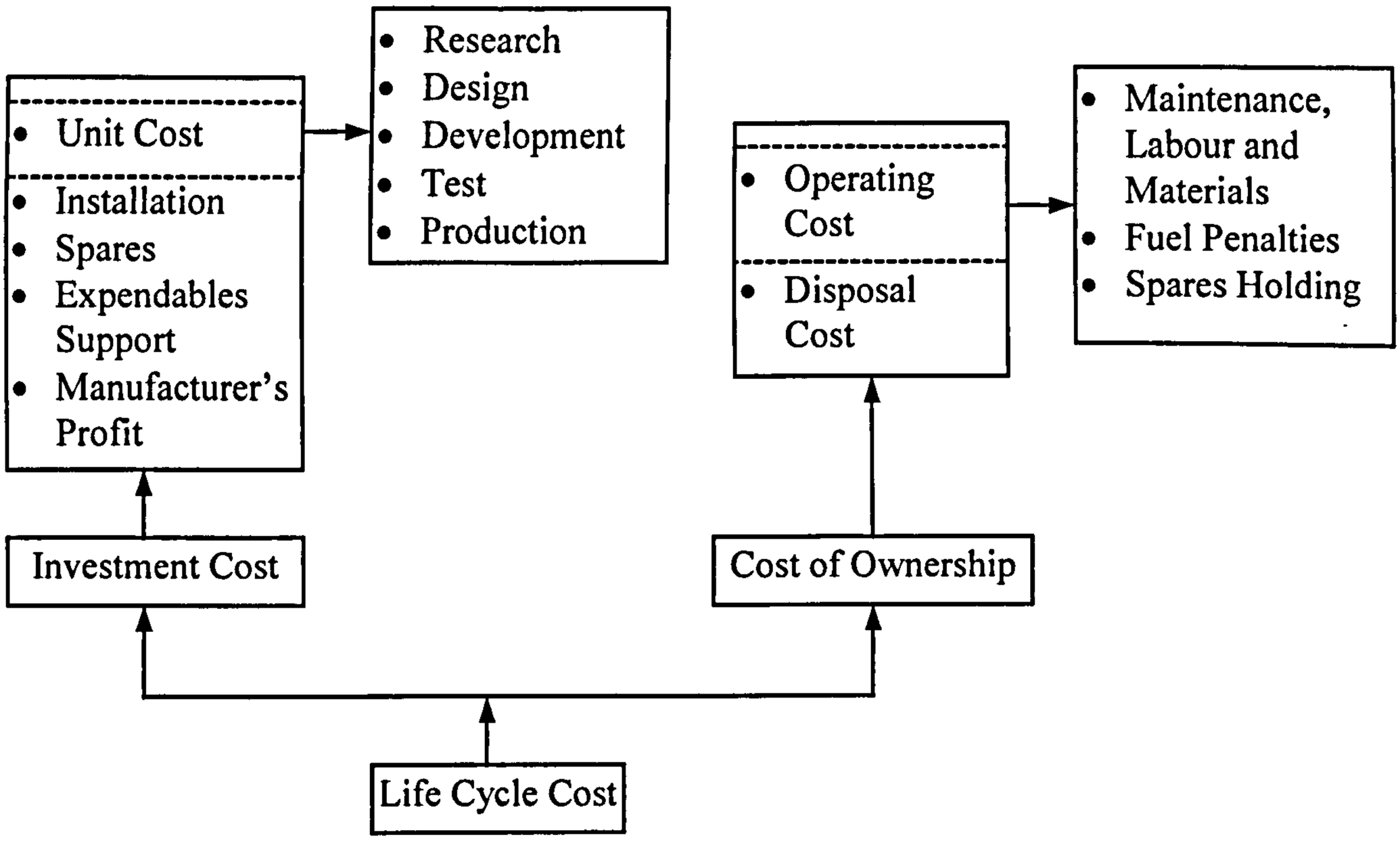


Figure 2-2. LCC for Digital Avionics[95].

Killingsworth and Jarvaise[56], of the RAND Corporation, provide a set of CERs designed especially for air force avionics systems. Because of the speed of the advances taking place in this field, the calculations are for the previous generation of stand-alone avionics, even though the paper was written in 1990. They are not compatible with integrated modular avionics (IMA) such as PAVE PILLAR, the system used for the F-22 ATF, RH-6 Commanche, and similar to the AIMES system used in the Boeing 777. However, the equations may be used to calculate the costs of a complete stand-alone sub-system such as the radar, provided detailed design information is known. The second section of the document contains a detailed review of the technology.

Section III is the most relevant, providing implications on LCC of advanced avionics. One of the most important points raised is the influence of 'performance push' on the costs of avionics in the 1990's and beyond. 'Performance push' is the tendency of the programme managers and engineers to design systems to the highest levels of functionality and performance permitted by current technology. Thus, instead of the costs of the avionics being reduced by use of more advanced technology for a given performance level, as the technology becomes available, the performance level is increased to match the technology. This leads to increased costs through increased research, development and test time, and use of state-of-the-art (expensive) processors and chips. Many useful time- (and cost-) saving systems have been developed recently, e.g. CAD, which help to increase productivity, but the effects of these systems can very quickly be negated through performance push. Coupled with this is the vast expense of the CAD systems (some can cost US$10-20 million) and their relatively short useful lives (estimated at only 3-5 years). Another implication of the high levels of integration achieved with very high speed integrated circuit (VHSIC) electronics is the amount of simulation required during the design of the systems into which they go. Simulation as a testing process at the system level can nearly double avionics design costs.

Military systems normally require even higher levels of testing than commercial systems, especially in the operation and support stages, where it is necessary to repeatedly verify that individual parts will work as they should. The old style of Very Large Scale Integrated (VLSI) chip could take up to ten hours to test - this is clearly not acceptable when considering the hundreds of chips that are present on aircraft such as the F-22 ATF. Testability is therefore one of the most important challenges in the practical development of VHSIC electronics in military systems. The acquisition process discourages early investment in system reliability, maintainability, and testability. Programme managers are motivated to get a design that works and let others take care of the logistics details later. With systems incorporating highly integrated circuit technology, this would be a disastrous philosophy. Testability is a systems concept and cannot be tacked on later. It must be mandated by corporate policy; without such measures, design engineers will ignore extra requirements, in order to meet deadlines and achieve cost goals. Some estimate that the leverage of funds invested for testability early in the design systems would be 20 to 1 in savings on O&S costs.

Self-test, or Built-In Test (BIT), mentioned earlier, is a concept that has been in use for some time. It has been implemented with varying degrees of success, but is now considered essential for advanced avionics systems. Without the help from the chip itself, the use of external test equipment to exercise and verify the devices on a VHSIC chip with an acceptable level of coverage is time-prohibitive, because there are so many devices and so few leads with which to access them. Another reason for advanced systems to incorporate BIT is that they will be susceptible to higher incidences of intermittent faults, which must be detected and isolated as they occur. It is envisaged that systems incorporating VHSIC will have extensive BIT circuitry, not only at the chip level but at the module and subsystem level as well. A fault will be reported to maintenance processors at the module and subsystem level, which will record the problem and its location. The goals for the avionics on the ATF are 99% fault detection and 98% fault isolation to the defective module. If this is actually realised, most of the jobs performed by the Avionics Intermediate Squadron (AIS) could be performed on the flight line. Faulty modules would be sent directly to the depot for repair, coming closer to a two-tier maintenance hierarchy. Intermediate-level maintenance would then be done away with, reducing personnel and support equipment costs.

Although BIT will be essential to the implementation of maintainable advanced avionics systems, it does not come without a price. One VHSIC contractor expects 21% of the area of Phase 1 chips to be BIT circuits. Phase II chips could be from 30% to 50% BIT circuitry, and by increasing the area of the chip, BIT decreases the chip yield, thereby increasing the fabrication costs. By increasing complexity, BIT itself could increase failure rates, and by using more power, it generates more heat, thus potentially lowering the mean time between failures (MTBF). High levels of BIT become self-defeating at some point. The real cost effect of BIT will again be felt in the design of the system.

Many more details of costs and the arguments for and against modern improvements are given throughout the paper, along with some estimated figures for the ATF, e.g. there will be approximately 320 avionics modules on the ATF, each one costing US$20,000 - $50,000. These high acquisition costs will be offset somewhat by the lower O&S costs expected from these advanced systems, specifically through the use of BIT and increases

in reliability. The potential for increased reliability results primarily from the lower number of electrical interconnections in VHSIC systems above the chip level. While 55% of total aircraft failures are related to avionics failures, 60% of these failures are related to interconnections above the IC level. Other estimates place this figure as high as 75%. The extreme level of integration that VHSIC offers means that many more of the interconnections are in the chips themselves and not made with solder or mechanical connectors. If the systems reliability is improved, it would mean cost savings in both spares and repair costs.

Details of the mean time between maintenance (MTBM) and its effect on the overall maintenance costs for several aircraft is given, along with tables for replacement spares cost per flying hour and depot maintenance costs against suite cost and reliability. Graphs are given of avionics suite cost versus year of first flight and suite cost per pound of weight versus year of first flight. Tables are given of average costs per pound weight for several typical systems, e.g. fire control radar, controls and displays, etc. The level of integration is accounted for by a basic factor and some assumptions as to what that factor might be for a range of starting year dates. Thus, the three main cost drivers for the avionics suite, not including software, are weight, level of integration and technological maturity.

Logan[61] discusses the modular avionics systems architecture (MASA), its implications on the design of combat aircraft, and the retrofitting of such an architecture to existing designs. It is claimed that the use of Common Avionics Modules that could be used on a variety of aircraft and DoD equipment (such as GPS systems) could result in significant cost savings. Conservative estimates claim LCC savings of 25%, 20% improvement in R&M, and 10% reduction in support personnel over equivalent technology point designs for integrated multi-function avionics applications across multiple aircraft. There are still many issues to resolve with the implementation of MASA, not least of which are the loss of traditional product lines and the change in the way that Defence contracts are awarded. A strong case is made for the use of modular systems, though many of the problems associated with them are highlighted.

As far as individual technologies are concerned, Williams[105] expands on the concepts of Built-In Test, by introducing Wrap-Around Test (WAT). WAT builds on the BIT principle existing in the aircraft by utilising the aircraft self-monitoring equipment and redirecting it to encompass the entire weapon system. Examples given are mainly for the weapons system, where a small adapter replaces the complex ground test equipment for the weapon system. The use of this adapter reduces costs, as the adapter is a fraction of the cost of the test equipment. The real savings come, though, in the reduced numbers of test equipment packs (2 per squadron), and the much reduced logistics trail cost for supporting the WAT. The only drawback is that the WAT adapter resides on the aircraft, increasing mass, but only by a very small amount.

Another avionics technology, or technology usage, is the use of Commercial Off-The-Shelf (COTS) technology, for which there are many references available. It is claimed that COTS technology in military avionics can substantially lower development costs while reducing weight, power needs, and volume. It has come about through the relaxation of military standards, which previously specified the use of bespoke military

hardware and software for military systems. The cost savings will be realised through sharing of common components, reduced development costs, and better heat dispersal and storage, using standard electronic modules. COTS technology cannot yet replace all the functions required of military avionics suite. The areas where military technologies are still required include:

- An integrated sensor system that furnishes the radar, communication/navigation/ identification, and electronic warfare functions.
- Electro-optical sensor front-end electronic systems.
- Digital signal processing for sensor data.
- Aircraft-unique interfaces with digital, analogue, and cockpit controls.
- The vehicle flight and management system.

Inserting COTS technologies into military avionics systems will be meaningless if they cannot survive and perform reliably. In order to ensure that commercial PC cards and components can be used in military applications, the components can either be tested to military specifications, or the components must be kept in an environment similar to that experienced on the ground, by way of an environmental protection enclosure. In previous studies, COTS PC cards were militarised by testing to ensure compatibility, enclosing them in a hermetically sealed stainless steel case, and testing them for environmental changes, such as temperature, humidity, and salt spray. This process added about $1300 for a PC flash RAM card, enabling production costs of about $1500-$3000 for small quantities (in the hundreds) to $700-$2000 for larger runs. This equates to a reduction in cost of 8:1 in favour of the COTS card, whilst similar reductions are shown for power (6:1), weight (10:1) and volume (15:1) for a standard electronic module. It is yet to be determined whether COTS components can deliver the required reliability, but should this prove to be the case, the COTS concept could fulfil the promise of affordable military avionics systems.

2.3.6 Software

One of the most difficult areas in avionics systems cost estimating is software. As software costs are rapidly becoming some of the largest and most unpredictable cost values in the development of any new aircraft, it is important that programme managers have a good idea of the real costs of software development. That said, it is debatable how much software development costs influence the design of an aircraft, as the air vehicle control software for the F22 (including engine management and all aircraft flight control functions) accounts for only 11.2% of the total software size[55]. If it has been established that the aircraft will make use of a fly-by-wire (FBW) control system and associated software, it is unlikely that there will be a significant difference in the software size from one aircraft to another, provided the mission profile is reasonably similar. However, software size is not the same as software cost, as the air vehicle software must be the most reliable present, increasing development and testing time, and hence cost. The other 89% of the software lines of code control all other aircraft avionics functions. It can be assumed that production costs for software are negligible, as the majority of the work is aimed at algorithm development, coding, and testing, which should be non-recurring tasks, except for the purposes of de-bugging.

Difficulties in estimating software costs can come from imprecise specifications of performance, poor understanding of the requirements, and/or poor estimates of the software team capability. It is essential that requirements be clearly written and equally clearly understood; only then is it possible to reasonably estimate the number of instructions and size of the databases in the software, based on prior similar situations. The best possible quality of estimate of the software and database size is essential, since all subsequent software costs are based on it; the more accurate the size estimate, the better the cost estimate will be. Given a good estimate of the number of lines of code and database size, the next principal uncertainty is the skill of the programmer, which can change the cost of a line of code by a factor of two.

One of the most widely used software cost estimating models is the COnstructive COst MOdel (COCOMO), devised by Boehm[17]. COCOMO can operate at three levels of detail, from macroscopic to microscopic (which includes a complete work breakdown structure and phase sensitivity multipliers). The intermediate level of COCOMO is claimed to achieve a cost estimate within 20% of the actual cost, 70% of the time, as long as it is used within the class of projects for which it is calibrated. This is not as accurate as one might like, but it does provide a good deal of help in software engineering economic analysis and decision-making. Using COCOMO is basically a two-step process - the first being the fundamental beginning point for all software cost models: the most reliable estimate of the number of delivered source instructions, and the development mode of the project. Eight features of the software are correlated to the three possible development modes: organic, semidetached, or embedded. Avionics, complex transaction processing systems, and ambitious command control systems are all given as examples of an embedded mode, as shown in Table 2-1. With the size of the program and its development mode known, the number of programmer months and development time in months can be calculated from simple equations.

Feature	Embedded Mode
Organisational understanding of product objectives	General
Experience in working with related software systems	Moderate
Need for software conformance with pre-established requirements	Full
Need for software conformance with external interface specifications	Full
Concurrent development of associated new hardware and operational procedures	Extensive
Need for innovative data processing architecture, and algorithms	Considerable
Premium on early completion	High
Product size range	All sizes
Examples	Avionics Ambitious command control

Table 2-1. COCOMO Embedded Mode software features.

Step 2 is to determine the effort multipliers that are applied to the 'nominal programmer months' value to achieve the most probable number of programmer months. There are 15 cost driver attributes shown below that have varying degrees of impact on the cost of the software. Boehm gives a set of tables to establish the rating for each attribute from very low to extra high, and another table for the complexity attribute. Once the ratings are established, the effort multipliers are applied to each attribute to extract the total programmer months. Assuming all 15 attributes are rated, the product of all effort multipliers can range from 0.0886 to 74.7408. Thus, it is obvious that careful attention must be given to the ratings of each of the attributes.

COCOMO Development Effort Ratings

- Required software reliability - high, as there is a risk to the aircraft, and human life. Higher reliability leads to higher development costs, but lower operating costs.
- Database size - larger database requires more programmer time and effort.
- Product complexity - depends on type of module (control, computational, device-dependent, and data management).
- Execution time constraint - amount of processor capability utilised. This is often limited, as in the F-22, to increase reliability through redundancy of resources.
- Main storage - as above, except for memory usage.
- Virtual machine volatility - danger of having to replace computers, which is obviously not desirable.
- Computer turnaround time - possibility of repairing computer should an error occur.
- Analyst capability - difference between good and bad programme managers.
- Applications experience - number of months (years) experience.
- Programmer capability - as for analysts.
- Virtual machine experience - as for analysts, but times will more likely be shorter.
- Program language experience - as above.
- Use of modern programming practices.
- Use of software tools.
- Required development schedule - faster schedule requires more personnel.

Wanstall[100] gives some indication of the cost of avionics and software. For the first 30 Gripen aircraft, the cost of the fly-by-wire (FBW) system and associated software was estimated to be US$35 million. It is not clear as to what stages of the development process this includes (if any), as this seems to be a very conservative figure for an extremely complex and risky development. Kandebo and Hughes[55], in their article on the F-22, claim that the software size is approximately 1.558 million lines of code. Using the COCOMO method detailed above, and using sensible values for all of the input values therein, a development effort rating of 1.0216 was arrived at for the Air Vehicle code, giving a timescale of 4 years 10 months, and a cost of FY90$250 million. The development effort rating for the rest of the aircraft was lower at 0.5611, but the increased size of the code means a development time of 8 years and 8 months, at a cost of FY90$1.54 billion. This gives a total development time of 8 years 8 months, and a total cost of approximately FY90$1.8 billion. The cost figures above cannot be checked, but the timescales correlate well with the published data, giving some confidence in the cost values.

There are a number of other excellent references dealing with software costing, but the only other one mentioned here is that by Bruce and Pederson[18]. The most relevant section is Chapter 3 - Planning the Project. All aspects of software costing are covered, and many of the points discussed in this section are extended and examined in different ways. One of the most useful points for would-be software developers is the explanation of cost estimating approaches and the example of the Project Plan. An interesting point raised in this and other references is a common misconception in many disciplines, not just software development. That is: in order to speed development, or correct a slip in the programme schedule, adding personnel will not necessarily be beneficial, as productivity drops. This is due to the number of possible communication paths increasing according to the simple formulae; Paths = N(N-1)/2. As can be seen, the number of communication paths increases faster than the number of personnel, requiring a very disciplined approach to adding personnel. This is particularly true when rescuing a slipping project, as standards and policies are ignored to speed development. This, in turn, will lead to lower productivity and the perceived need for yet more personnel to be added.

It should be noted that software does not suffer 'failures' similar to those encountered in mechanical systems. Rather, problems are encountered in the software that are due to development or programmer errors that were not detected during the test stages. Thus, investment cost can be thought of as the time spent developing and testing the required software. The higher the level of testing and de-bugging, the higher the development costs incurred, but there should also be fewer errors remaining in the software, resulting in fewer errors, or 'failures' occurring during the service life of the software system.

2.3.7 Reliability and Maintainability

Reliability and maintainability (R&M) is perhaps becoming one of the most important design areas, not only when applied to modern military aircraft, but to all aircraft, and indeed to many commercial and consumer items, probably as a result of LCC becoming an every day, but unnamed phenomenon. For instance, when buying an expensive item, such as a car, most people want to know how much the item will cost to buy, how much it will cost to maintain/ operate/insure, what life they can expect from it, how reliable it will be, and what its residual value will be at re-sale.

Reliability

Reliability is the probability that an item will carry out its mission satisfactorily for the desired period when used according to specified conditions[32]. In the context of aircraft operation, it is a measure of the time or usage (normally in flight hours) between failures of a component or system. Thus, every component on the aircraft will have a reliability, which can be expressed in failures per flight hour, or more commonly as a mean time between failures (MTBF). MTBF and mean flight time between failures are normally synonymous, as far as aircraft are concerned, as it is very unlikely to suffer a failure whilst not in operation. The measurement of reliability is not an exact science, as reliability can be affected by many variables; heat, vibration, operating environment, operator, poor maintenance (giving 'installed' faults), usage, manufacturing tolerance, etc. This makes the prediction of reliability a very difficult task, and yet one that has a major impact on the in-service cost and availability of a particular aircraft.

Alexander[2] investigates the costs of achieving greater reliability in military equipment, the benefits of improved reliability in reduced support costs and increased availability, and strategies for attaining reliability goals. Three kinds of evidence were examined: reliability improvement programmes, new product developments, and statistical analyses of reliability costs and outcomes in new programmes. Many of the case studies used for that report are not valid for this work, but those that are include the F-18 Hornet, the F-16 Falcon reliability improvement programme, the CH-47D Chinook helicopter modernisation, and the F100 turbine engine development. These programmes show that when reliability was a goal of equipment users and developers, substantial reliability improvements were possible (some up to 15 times the original level) and that original reliability did not constrain the size of the improvements. It was also found that by comparing the changes in RDT&E investments attributable to reliability with the percentage changes in reliability, that increasing returns on investment were realised. A 10% increase in reliability would cost, say, 5% more in RDT&E expenditures, whereas a doubling of reliability would cost 20% more, and a five-fold reliability gain would require only a 50% increase in development costs. In most cases, unit production cost did not rise with increased reliability, suggesting that the majority of the effects are in non-recurring investments.

Performance trade-offs that reduce stress can increase the life of parts and components and reduce the probability of failure. Review of these cases and other experience suggests that reliability is often improved at a cost of reduced performance. Although this route to reliability was not universally applied, it is becoming a more explicit choice among users and developers. Special cases where this method is not relevant include: gust alleviation systems, which can reduce load amplitudes without performance penalties; manoeuvre load alleviation, which can reduce limit loads, but requires highly reliable systems to enable structural mass savings; and smart structures, which again must be reliable to avoid catastrophic damage. In the stress-reduction cases examined, well-chosen reductions or limits to performance often yielded substantial reliability gains. What is not detailed in the report is whether the reliability gains achieved always resulted in significant mass increases, and if so, of what magnitude these were. Some figures are given for the specific programmes, as shown below.

Reliability improvements have direct effects on maintenance time and personnel, spare parts usage and investments, operational availability, logistics loads, and life cycle costs. Availability rates can increase by one-fifth to one-half as much as the rate of reliability improvement (a 100% reliability gain could yield availability increases of 20-50%). The additional cost of the reliability increase for the F-18 was estimated by McDonnell Douglas to be FY85$155 million, and the total development cost to be $3.7 billion for a procurement quantity of 1377 aircraft. This comes to an investment of $113,000 per aircraft in 1985 dollars. The RDT&E investment in increased reliability represented about 0.6% of the $18.7 million production costs in 1985. There was also a 500lb increase in structural weight, equivalent to an increased production cost of $310,000.

Using the same methods for comparing aircraft reliability, the F-18 demonstrated a fleet average of about 1.75 to 1.9 MFHBF five years after production, compared with about 0.8 for the F-14 and F-4 and 0.95 for the A-7E. Thus, the F-18 is demonstrating levels of reliability roughly twice that of other aircraft. This translates into an estimated

increase in sortie generation rate (SGR) of 17%. To a first approximation, this is assumed to be equivalent to the same increase in force size. 17% of 1377 is an increase of more than 234 aircraft added to the inventory, and at 1985 prices of $18.7 million, this force increment would cost some $4.4 billion. At most, the cost of improving the reliability of the engine and airframe to the previously stated levels would be $700 million. Thus, spending $0.7 billion initially can be shown as a direct saving of $4 billion in 'decreased' force size. Examples of some other improvements are given in Table 2-2. Abell *et al*[1] provide more detailed, but similar information for the F-16A/B.

Case	Improvement Category	Original Level	Improvement	Percent Change
F100	Component Improvement Programme (CIP)			
1-Year CIP	Maintenance (MMH/FH)	1.72	0.26	15
	Support Cost (1985 $/FH)	530	175	33
10-Year CIP	Life Cycle Cost (1985 $/FH)		600	
F-16	Investment per aircraft (1985 M$) (Assumes constant 630 aircraft)			
Double reliability of only engines and fire-control	Engines and engine modules	1.25	0.54	43
	Recoverable peacetime operating spares	1.43	0.31	22
	Replacement for condemned spares	1.33	0.48	36
	Depot level repair of components	1.87	0.48	34
	Total	5.88	1.81	34
Double reliability of all subsystems	Sortie rate			17
	or			
	Maintenance manpower			9

Table 2-2. Benefits of Improved Reliability[2].

Many other systems are covered, including the modernisation of the CH-47 Chinook helicopter, but whether enough of a parallel exists between such systems to be able to draw accurate conclusions remains to be seen. Detailed reliability information has been obtained on the F15, F-16, F-18, and on several other NATO aircraft. Figure 2-3 shows the occurrence, by system, of faults leading to unscheduled maintenance per 1000 flying hours for typical British combat aircraft. This Pareto plot clearly shows those systems that require the most attention in terms of R&M improvements.

Jhanjee[51] proposes the use of computers in design, particularly the use of CAD and CAD/CAM systems, to produce better designs in order to raise R&M achievements. He also gives a list of 27 design processes to improve R&M characteristics, including: minimise part quantity; minimise part types; maximise part quality; minimise avionics part/junction temperatures, minimise the number of printed circuit boards, etc.

Dhillon[32] offers a detailed description of reliability modelling, the theory behind it, and a number of very useful references. This book also contains some extremely useful data and models for life cycle costing, including some basic models for aircraft LCC evaluation. By far the most useful aspect of this text is the number and the quality of references it provides for every part of LCC analysis. It has proven to be an excellent source of data, models, and references from the cost of airframe production and software development, to reliability modelling and economic theory.

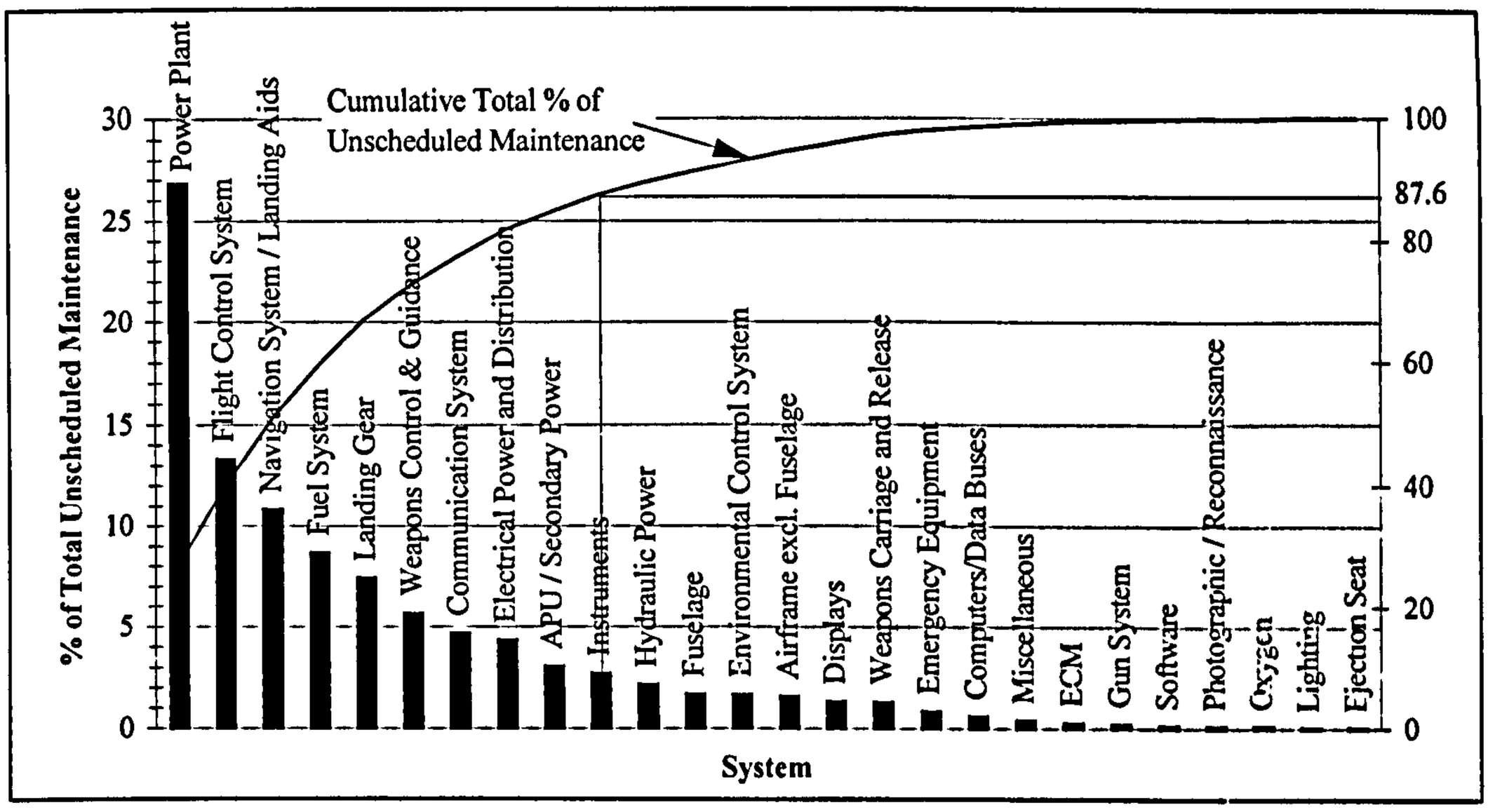


Figure 2-3. % Unscheduled MMH/FH, by System - British Combat Aircraft.

Maintainability

Maintainability is the probability that a failed item will be restored to its satisfactory operational state within a specified total down-time when maintenance action is started according to specified conditions[32]. It is a measure of the ease with which a component or system can be repaired or maintained, which is affected by a large number of factors, including accessibility, complexity, and many other so-called 'ilities'. Dhillon defines maintenance as all scheduled and unscheduled maintenance actions appropriate for keeping an item in a serviceable condition or restoring it to serviceability. It includes inspection, repair, remove and replace, servicing, testing, modifications, etc.

While much emphasis has been placed on reliability improvements in aircraft design, the maintainability of modern combat aircraft is posing a much greater strain on military maintenance facilities. As mentioned above, unscheduled maintenance (i.e. work to fix a fault) accounts for approximately 1.43 MMH/FH for British combat aircraft, and probably slightly higher values than this for comparative American aircraft. Total maintenance times for similar aircraft would not be under 20 hours: the lowest total MMH/FH for any military non-training aircraft, before the introduction of the Rafale, was held by the F-16, at approximately 14 MMH/FH. The unscheduled MMH/FH figure for the F-18 for the period January 1993 - December 1993 was 3.7, accounting for approximately only 20.5% of its total MMH/FH. The F-18 was specifically designed for both high reliability and maintainability, which would not appear to be the case given the high scheduled maintenance requirement indicated. Other aircraft show figures much worse than this, with the F-111 having MMH/FH ratios as high as 44MMH/FH. It would appear that improved detailed design is required to ensure lower scheduled maintenance requirements and improve aircraft maintainability.

Retterer and Criscimagna[80] provide figures for other USAF aircraft, including the F-4, F-16A&B and F-15C. They give the cost of maintenance time as FY87$30.71/MMH and the total O&S costs of the USAF as 43.6% of its total budget. They also present a small discussion of software maintainability, which is not often covered in available texts. Nearly 5% of the total USAF budget, or nearly $3000 million per year, is spent on software. Software has become a dominant factor in modern weapon systems, and its importance as a force multiplier makes software maintainability a critical concept. Software should be designed to be highly testable, so that it can be de-bugged prior to use and can be easily modified and maintained during operational use. Modular programming and standard higher order programming languages can help to make software more testable, easier to maintain, and enhance reusability.

It became apparent during the course of the research that there was a need to be able to predict reliability and maintainability of combat aircraft. Two models were found early in the research, one based on the other. The initial work was by Harmon *et al*[40], and is based on the correlation of maintenance data for ten USAF aircraft with capability, performance, and design parameters. The model uses historical maintenance records for the aircraft, which were collated and grouped by major aircraft systems. The relevant aircraft design/performance characteristics were chosen and correlated with the maintenance data by system and maintenance level. The model is very thorough, and gives good results, although it is outdated now, and is heavily reliant on the aircraft mass, which is not desirable for mass/technology trade studies.

Serghides[86] performed an update of this work for British combat aircraft. The methods used are very similar, although the number of systems is smaller - 13 in Serghides' work, as opposed to the 26 systems in the Harmon model[40]. Also, Serghides does not include a separate model for the prediction of scheduled maintenance, as Harmon does, but does provide an estimate of system reliabilities as well as what is called the Defect Man-Hour Rate (DMHR). This is effectively a measure of the total maintenance per flight hour required by the aircraft, and can be thought of as the product of defect rate and mean time to repair for each major system.

Serghides claims good results for the models, but as these are based on the last two generations of combat aircraft, the algorithms are out-dated. If the algorithms are re-worked, the systems should be split differently, as systems that were notoriously labour-intensive on past aircraft now contribute a much smaller percentage of the total maintenance requirement. A good example of this is the oxygen system: on older aircraft, the use of liquid oxygen (LOX) made the replacement of oxygen a difficult and dangerous task. The location of the storage bottles was dictated more by mass savings than by logical placement - LOX bottle replacement on the Gnat requires the removal of the tail cone! In modern aircraft, the use of on-board oxygen generating systems (OBOGS) means that the Oxygen system now accounts for less than 1% of total maintenance. In future, it is suggested that the oxygen system should be grouped together with other crew-determined systems, such as cockpit displays and controls, ejection seats, etc., to create a group called 'Crew Systems', correlated to the number of crew (among other things). In this way, if an uninhabited design was proposed, all of the relevant systems would show zero maintenance, allowing the accurate prediction of aircraft maintenance without the need for factors on several different system algorithms.

Serghides and Fielding[87] present a maintainability-prediction methodology for use in aircraft design. The design must be fairly advanced for this particular method to be employed, preventing its use at the conceptual design stage. It is noteworthy, however, that three checklists are given from MIL-HDBK-472 for scoring certain attributes that contribute to the overall maintainability of the aircraft. Checklist A is used to score physical design factors, checklist B scores facilities dictated by design, and checklist C scores maintenance skills. Some, or all, of these factors may be able to be included in the conceptual design stage, although exactly how this may be done is not yet clear.

Some further points that have become clear from the references in this section and from other reading are: R&M can only be achieved if quantified targets are set and specifically designed for during the early stages of the design process; a two-level maintenance system can significantly reduce maintenance costs; built-in-test (BIT) does help to reduce maintenance time, and if, in future, maintenance strategies can be altered to rely on BIT results, many maintenance functions can be delayed until necessary - "If it isn't broken, don't fix it."; and "Reliability/Maintainability cannot be 'tested in' - you have to spec. it, check it, and pay for it"

2.3.8 Other Data Sources

As will become more common in future, much of the useful data and many of the models available are published on the Internet. In order to provide those with an interest in LCC with some idea of possible data sources, a list of Uniform Resource Locators (URLs) is included below, with a brief description of the sites and their contents. The list below is by no means exhaustive, as sites operated by private individuals have been avoided, in an attempt to safeguard the integrity of the references with regard to time.

Air Force SAF/FM Home Page: **http://www.hq.af.mil/SAFFM**
This is the official Home Page of the Assistant Secretary of the Air Force (Financial Management & Comptroller). The site contains many useful links to may other USAF costing resources, one of the most useful of which is the AFI 65-503 Cost And Planning Factors page, found at: http://www.saffm.hq.af.mil/SAFFM/FMC/afi65503.html. This page contains much useful data, including aircraft acquisition costs, pay scales, training costs, logistics cost factors, attrition rates, squadron strengths, etc.

Air Force VAMOSC Home Page: **http://www.vamosc.tasc.com**
VAMOSC stands for Visibility and Management of Operation and Support Costs. This site gives some idea of the extent to which data is gathered and disseminated throughout the US DoD. The individual data tables cannot be viewed by those outside the DoD, but the site is useful nonetheless.

Army Cost and Economic Analysis Center: **http://134.11.192.15/ceac.htm**
This page gives very similar data to the Air Force SAF/FM home page above, but for the US Army. It contains pages for Army Military-Civilian Cost Tool (AMCOS); Cost & Economic Analysis; Discount Rates; FORCES cost models; Managerial data; Activity-Based Costing (ABC), Service-Based Costing (SBC), and Standard Service Costing (SSC); and details of reports and other useful information generated by the Army costing cell.

Design for Competitive Advantage: **http://dfca.larc.nasa.gov/toc.html**
This page is managed by Edwin B. Dean, author of numerous papers on cost estimating, design for competitive advantage, and other costing issues. The table of contents leads to many other headings, including; Technology, Business Technologies, Cost Technologies, Engineering Technologies, Human Technologies, Mathematical Technologies, Quality Technologies, System Technologies, Designing for Value. Although not giving much in terms of raw data, the site is very useful for costing concepts, especially some of the more abstract.

General Accounting Office: **http://www.access.gpo.gov/gao**
The General Accounting Office (GAO) performs many similar tasks to those of the National Audit Office in the UK. The Reports section of this web site contains all publicly released GAO reports since 1995, with a searchable index. All reports are available for download in electronic format, making this an excellent resource. The GAO conducts detailed studies on behalf of Congress, many of which concern military procurement, such as the report on the Operational Improvements of the F-18E/F[81].

NASA Technical Reports Server: **http://techreports.larc.nasa.gov/cgi-bin/NTRS**
The NASA Technical Reports Server is possible the single most important electronic resource utilised during the course of this research. It provides access to a huge amount of data by allowing detailed searches of all data stored at all NASA sites. Although much of the data is restricted to space and satellite applications, there are a number of abstracts contained in the databases that are aerospace-specific. Although few whole documents are available from the server, the abstracts often provide a good indication of the document content.

Naval Center for Cost Analysis (NCCA): **http://www.ncca.navy.mil**
This page is very similar to those of the Air Force and Army cost analysis agencies. This site provides details of the Cost of Manpower Estimating Tool (COMET), gives examples of Costing Research taking place and inflation indices, explains the uses and implementation of the Navy VAMOSC and other databases, and explains some of the Software cost estimation work taking place.

Parametric Cost Estimating Reference Manual: **http://www.jsc.nasa.gov/bu2**
This is a very useful page, containing many useful links to other sites, but also provides the Parametric Cost Estimating Reference Manual as an electronic document available for download, or in full text form on the web site. It provides a list of costing and aerospace acronyms, a number of links to professional associations, and a list of available cost models, many of which can be used on-line to save programming, or even obtaining the reference in some cases. The site also contains a list of space costing databases, economic and inflation data, job and career sites for cost-estimating engineers, and many other useful links and resources.

RAND Corporation: **http://www.rand.org**
The RAND Corporation has been a World leader in military aerospace analysis for many years, providing technical support, policy advice, and advanced parametric models to the USAF for over fifty years under the single contract heading Project Air Force.

Many papers referenced in this thesis were produced by RAND, and many others are emerging all the time. The Web Site provides a means of directly tracking research areas, giving a searchable database of abstracts of all non-classified research, as well as an on-line ordering system. This has proven to be one of the most useful resources found during this research programme, and is strongly recommended for all military costing or reliability engineers.

Society of Cost Estimating & Analysis: **http://www.crols.com/scea**
The main purpose of this Web Site is to further the effectiveness and efficiency of cost estimating and analysis and related disciplines in the public and private sectors. The site contains many useful cost-estimating links, especially for cost estimators. It also details a professional accreditation programme and gives a number of resources available to cost estimating professionals, including a comprehensive list of potential employers.

2.4 Summary

The literature search was an invaluable first step in the development of the design for life cycle cost synthesis. Problems have been encountered in information-gathering, in that many people produce data that is not properly referenced, or is introduced in a manner that is so vague as to be unusable, introducing doubts as to the accuracy of the quoted figures. In some cases this is probably intentional, to protect the source and technical secrecy of the companies/individuals providing the data. Another problem encountered was that of papers whose titles suggest promising data, but end up being nothing more than ideological approaches to the subject of LCC and do not contain any significant information at all. Those obstacles aside, the Literature Search has:

- Introduced the wide-ranging topics that make up LCC estimation.
- Provided a vast amount of information for use in the synthesis and LCC estimation program.
- Allowed the collection of a large number of costing references and ideas for future developments of the program.
- Introduced the author into the costing community, publicising the work, and enabling the gathering of otherwise unavailable data.
- Provided a useful starting point for others following on from this work, and for those interested in costing techniques and models in general.

3 Optimisation Processes and Tools

The aim of this chapter is to provide the reader with sufficient information to understand the optimisation process, without it becoming a major section of the thesis. Although optimisation plays a significant role in the work as a whole, the optimisation tool is a commercial product, and no development work was performed on it during this study. A brief description of the basic principles of the optimisation method used in this research is given, whilst avoiding excessive mathematical details

This first section gives an introduction to optimisation, explaining unconstrained and constrained optimisation problems. It also covers some of the problems that are inherent with these types of optimisation, and the ways in which they have been tackled in this research programme. Later sections describe the method chosen for the application to the research as a whole, and explain some of the details of the chosen optimisation tool. Again, mathematical details are kept to a minimum, although the basic formulae are presented in a simple form.

3.1 Introduction

Optimisation is a process that deals with problems of minimising or maximising a function of several variables, usually subject to equality and/or inequality constraints[92]. The methods and processes of optimisation have undergone rapid development in recent years, and many new names have been generated to describe the procedures that the mathematical functions mimic. Only those relevant to this research are discussed.

The most basic non-linear optimisation problem in continuous (non-discrete) variables is the *unconstrained optimisation* problem. This is the problem of finding the minimum point of a non-linear function of n real variables, and is denoted such that:

$$\text{given } f:\Re^n \to \Re, \ \underset{x\in\Re^n}{\text{minimise}}\, f(x) \ . \tag{3-1}$$

It is assumed that $f(x)$ is at least twice differentiable, so that a local minimum (or maximum) value may be established. The basic methods for unconstrained optimisation are most easily understood through their relation to the basic methods for a second non-linear algebraic problem, the *non-linear equations* problem. This is the problem of finding the simultaneous solution of n non-linear equations in n unknowns, denoted:

$$\text{given } F:\Re^n \to \Re^n, \text{ find } x_* \text{ for which } F(x_*)=0 \ , \tag{3-2}$$

where $F(x)$ is assumed to be at least once differentiable.

Unconstrained optimisation problems arise in virtually all areas of science and engineering, and a significant number of real-world examples are data-fitting problems, such as the one used to produce the engine dimension and mass estimation method in section 5.2.4. The size of such problems may vary considerably, from small problems, with n in the region of 2 to 10, to problems where n is in the order of hundreds or thousands. In many cases, this one included, the *objective function*, i.e. the function to be minimised, is generated by a computer routine that is computationally expensive to calculate, so that often even small problems are difficult and expensive to solve.

The user of an unconstrained optimisation method is expected to provide the function $f(x)$ and a starting guess to the solution, x_0. The routine is expected to return an estimate of a *local minimiser* x_* of $f(x)$, the lowest point in some open sub-region of $\Re^n$. The user optionally may provide routines for evaluating the first and second partial derivatives of $f(x)$. In most cases they are not provided, and are instead approximated in various ways by the algorithm. Approximating these derivatives is one of the main challenges of creating unconstrained optimisation methods. The other main challenge is to create methods that will converge to a local minimum even if x_0 is far from any minimum point. This is referred to as the *global* phase of the method; the part of the method that converges quickly to x_*, once it is close to it, is called the *local* phase.

In many engineering cases, unconstrained optimisation is not appropriate, as there are physical, environmental, and manufacturing constraints to the problem. In this research, constraints have been added to prevent the solution entering a region for which empirical data does not exist. The problem is then known as a *constrained optimisation problem*; constrained optimisation problems involve the minimisation of a smooth non-linear function subject to smooth constraints on a finite set of continuous variables[70]. The implementation of the constraints can vary with context, and is left for later

discussion during the explanation of the chosen optimiser, RQPMIN. This work is not concerned with the detailed mathematics behind the optimisation process; interested readers are directed to Chapters 1 and 4 of 'Optimization'[70] for a fuller explanation.

3.2 Limitations and Pitfalls

Optimisation algorithms have progressed considerably in recent years, and are capable of providing solutions to complex problems in a very efficient and cost-effective manner. Yet, they are not without faults. Two significant problems, which have been encountered during this research, are those of discontinuities and local minima, although those with a deep knowledge of optimisation will undoubtedly point to more.

3.2.1 Discontinuities

From the first section of this chapter, it should be clear that much emphasis has been placed on both functions and constraints being smooth and continuous. This is a very important point in the implementation of the user-defined function $f(x)$, as it means that any kind of discontinuity can cause a false minimum, or more commonly, cause the optimisation process to stop at a sub-optimal point. A common cause of discontinuities is the inclusion of integer values in the optimisation process, and causes problems because of the way that optimisers evaluate the partial derivatives of $f(x)$.

Integer discontinuities have been prevented in the synthesis models, described later in the thesis, by ensuring that all of the available program variables are real and continuous between all reasonable values. Integer values can be used, but must be the subject of separate optimisation problems. For example, if the user wanted to investigate the effect of changing from a single crew member to two crew members on the design and cost of the aircraft, two separate runs would be required; one with a single crew member, and one with two aircrew. The results must then be compared manually to decide which one gave the lower value of the objective function. Provided the objective function and all of the constraint values had been left unchanged between the two runs, then the lower of the two values would therefore be the global optimum. The same process would need to be repeated for engines, fins, and other discrete variables.

Many optimisation algorithms use *gradient-based* approaches to predict the direction in which the search for the minimum objective function should proceed. Integer discontinuities can cause particular problems, as the differential of a function or constraint with respect to the relevant variable $\partial f / \partial x$ will tend to infinity. Thus, should a discontinuity be reached, or a continuous function be ill-conditioned, then the linearisation process used to determine the gradient of the function will break down, probably causing the optimisation to fail, as described above. Figure 3-1b shows a simplified representation of the problem for false positions returned from a line search due to an ill-conditioned function, $f(x)$, which has a distinct 'knee' in the output for incremental values of x. A gradient search method interpolating between the values x_1 and x_2 is unlikely to be able to determine the value for which $x = 0$, (x^*), unless much smaller increments than those shown are used, making the search inefficient. The well-behaved function shown in Figure 3-1a returns a value that is closer to the desired result, and gives a new gradient of the function that directs the search closer to x^*.

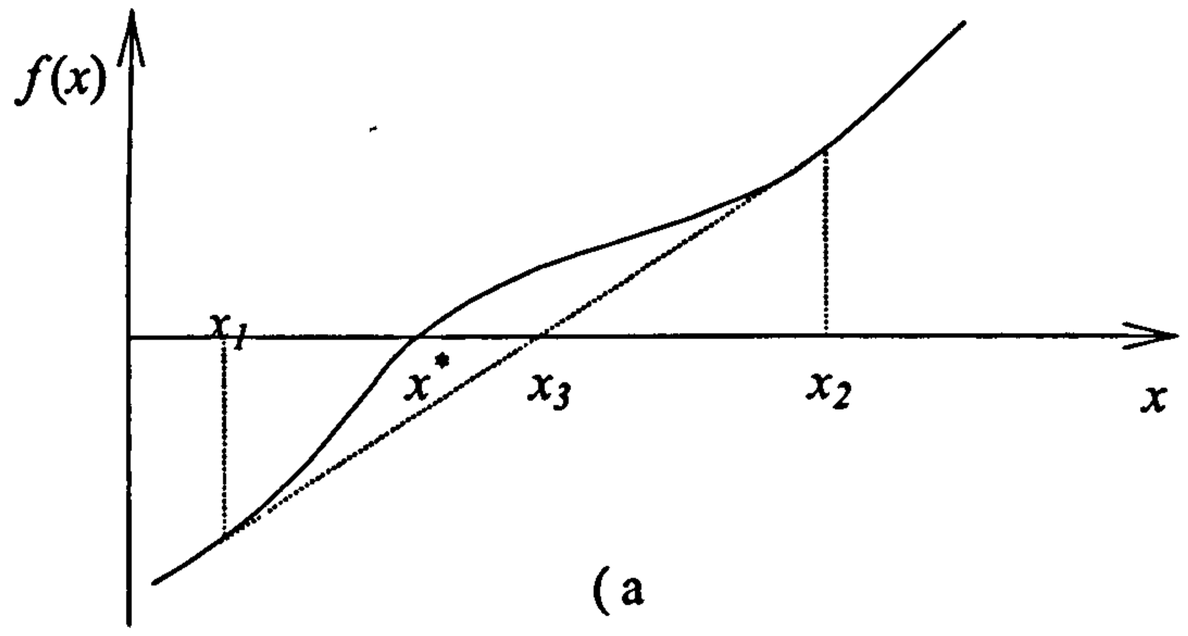


Figure 3-1. Examples of (a) Well-behaved Function (b) Ill-conditioned Function[69].

Ill-conditioned functions and smaller discontinuities are much harder to eradicate, as they often only occur on the boundaries of certain constraint functions, and when a function evaluation procedure changes from one method to another. The only way to solve this is by careful algorithm design, and exhaustive testing of the program, which is time-consuming. One area that is particularly difficult to specify is the level of convergence of internal iterations of the user-provided functions. In certain circumstances, a very small change in a particular variable can cause the number of convergence iterations to change, with a significant change in the value of the particular function (e.g. mass iteration to converge on gross mass). If this happens, the optimisation can stop or converge to a false minimum, because of an unduly large change in the value of the internal function.

One way of preventing this situation from arising is to tighten the convergence criterion for the iteration so that differences in the result are insignificant at the global level. However, this can cause dramatic increases in calculation time, as the rate of convergence often diminishes when close to the correct point. Another way around the problem is to specify a number of internal iterations to be performed, giving consistent results regardless of the changes being made to the variables. The problem with this fix is that the results are not always as accurate as one would like, depending on the starting point and the particular combination of input variables and/or function constraints. Finding a good balance between accuracy and run-time is more art than science in some cases, and can be a very time-consuming problem in itself. For this, and other reasons, it is often suggested that the number of internal iterations should be kept to a minimum, as this can reduce run times and improve convergence.

3.2.2 Local Minima

The optimisation process used in this research is described in section 3.3.7, and readers are directed there for an explanation of the minimisation process. Occasionally, the solution returned by the optimiser is later found not to be the global minimum of the optimisation problem. When this occurs, it is known as a local minimum. It most commonly occurs when the optimiser cannot move away from a best trial point in a direction in which $f(x)$ decreases, without leaving the feasible region. If the value of the objective function is higher in all other feasible parts of the domain than at the trial point, then that point must be the *global minimum*[69]. Clearly, a global solution must be a local solution, but the reverse is not necessarily true. Figure 3-2 shows a function of one variable, which, in the absence of any constraints, has two local minima in addition to the global minimum.

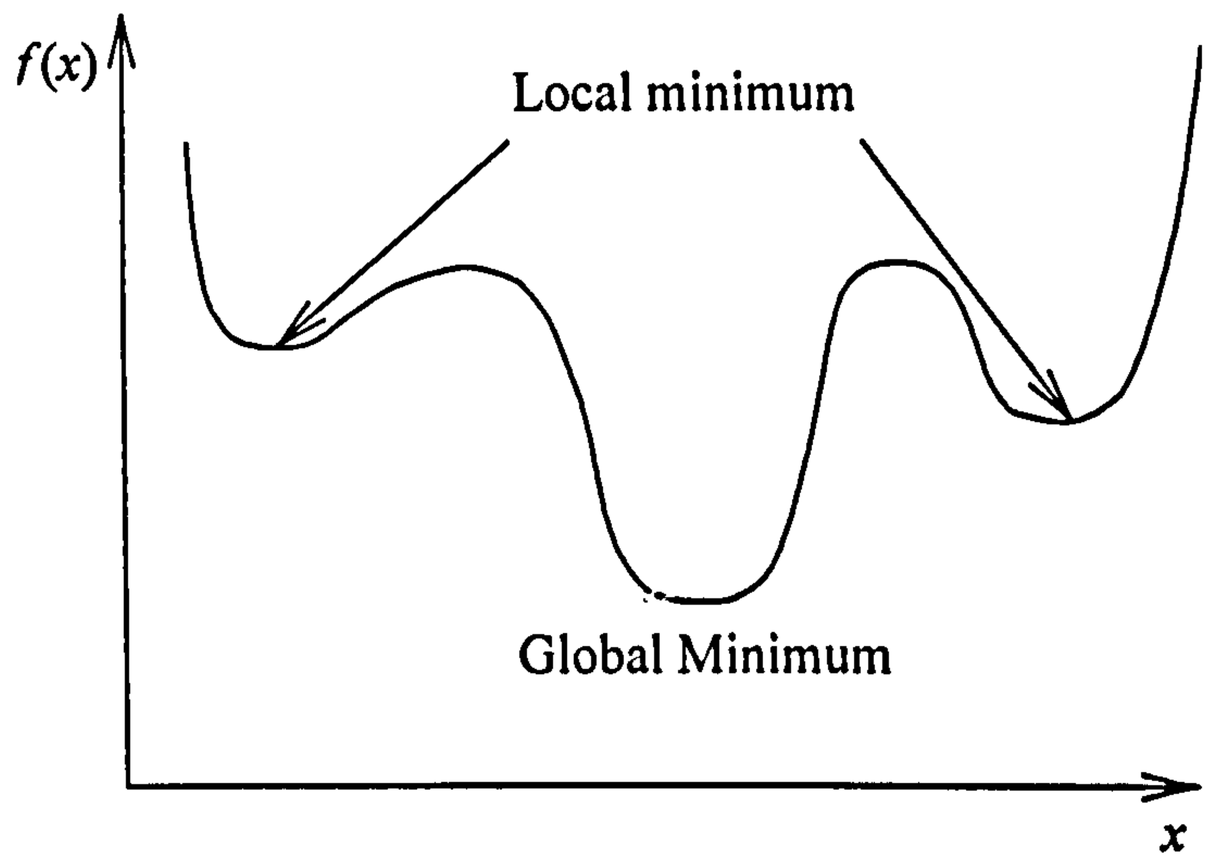


Figure 3-2. Example of Function with Local and Global Minimum Values[69].

There is no easy way to detect that the value given as the solution is indeed the global optimum, and to be sure, the problem is normally initiated from several starting points. In this way, the true optimum is more likely to be found, particularly if the result is the same from several different starting points. This is widely documented, and has been found to be the case during the course of this research, although most of the time the optimiser has been found to converge to a global optimum.

One way of reducing the likelihood of a local minimum being returned is to minimise the number of variables used in the optimisation problem. The fewer the number of variables, the less likely the optimiser is to settle in a local minimum, as there will be fewer conflicting drivers which can act to decrease the size of the feasible region. In this research, the number of variables has been kept deliberately small (12 in all), and the number of separate constraints has also been kept low, in order to reduce the number of dependent variables. The measures taken above to prevent discontinuities, i.e. smoothing of functions, etc., will also help to prevent local minima, as the optimiser relies on correct linearisation values to predict the next feasible point. If no feasible points are found due to incorrect linearisation, the calculation will terminate at the best point found so far, producing a local minimum.

3.3 Description of RQPMIN

RQPMIN is a computer program designed to solve constrained optimisation problems, i.e. problems requiring the minimisation or maximisation of a function subject to optional side constraints. RQPMIN is fast, reliable, and relatively easy to use; once the method has been established and the required programming links created, the code does not require any maintenance when changing the objective function, constraints, or any other optimiser variables. It can be used on a variety of computer platforms, including PC's, Sun Workstations, VAX or any machine running a VMS-compatible operating system, and Cray supercomputers. It is written in FORTRAN 77, and the user needs only compile and link with one relevant system-specific file and one standard file for immediate operation. As RQPMIN is a commercial software tool, the following section is not intended to be a detailed description of the search algorithms, rather an overview of operation; those interested in the theoretical background should consult Skrobanski[91], and the RQPMIN User Guide[92].

3.3.1 Problem Formulation

RQPMIN assumes that the optimisation to be solved is posed in the form:

$$\text{minimise } f(x) \ . \tag{3-3}$$

The function being minimised is called the *objective function*, and the symbol x is mathematical shorthand for a list of n real (non-integer) variables, $x_1, x_2, \ldots x_n$. At the solution to any problem, each of the variables x_i must satisfy a pair of inequalities

$$x_i^{(L)} \le x_i \le x_i^{(U)} \quad (i = 1, 2, \ldots, n) \ . \tag{3-4}$$

The values $x_i^{(L)}$ and $x_i^{(U)}$ are called *lower* and *upper bounds*. They are constants and are specified in the optimiser input file.

In addition, the x_i may satisfy a set of $m \ge 0$ constraints:

$$\left.\begin{array}{c} c_i = 0 \\ or \\ c_i \le 0 \\ or \\ a_i \le c_i(x) \le b_i \end{array}\right\} \ (i = 1, 2, \ldots, m) \ , \tag{3-5}$$

where a_i and b_i are constants specified in the optimiser input file. The functions $c_i(x)$ are called *constraint functions*, where each constraint can take one of three distinct types:

- an equality constraint, where $c_i(x) = 0$
- a single inequality constraint, where $c_i(x) \le 0$, or
- a double inequality constraint, where $a_i \le c_i(x) \le b_i$.

Each constraint type is defined in the input file, when the constraint limits are also set. Constraints are considered in three separate ways by RQPMIN: equality constraints; active inequality constraints; and inactive inequality constraints. An active inequality constraint is one that is temporarily treated as though it were an equality constraint of

the form described earlier. The collection of all equality and inequality constraints is called the *active set*, and is RQPMIN's guess as to which variables are satisfied, as strict equalities, at the solution of the optimisation problem. As the calculation progresses, a constraint may be removed from, or added to, the active set at any time, in some cases many times for certain variables.

3.3.2 Variable Categorisation

RQPMIN divides variables into the following categories:

- permanently fixed variables,
- variables held temporarily at their lower bounds,
- variables held temporarily at their upper bounds,
- dependent variables, and
- independent variables.

A variable can be defined as permanently fixed by setting its status to zero in the optimiser input file; RQPMIN will not change the value of this value, though the values of the remaining variables may be changed as normal. Some variables may be temporarily held at either bound, although the value of the variable will have already reached that bound. A variable may be fixed and freed from its bound several times during the calculation, although it is the author's experience that this can cause a higher number of function calls, and reduces the probability of a solution being found when used with the synthesis model described later.

Variables that are not fixed (permanently or temporarily) are called *free variables*, and are further classified as follows. The constraints are first *linearised*, and the resulting set of linear equations is used to eliminate a single free variable. Each variable eliminated in this way is called a *dependent variable*, indicating that, in general, the number of dependent variables is equal to the number of active constraints. The remaining variables are called *independent variables*.

3.3.3 Quantities Calculated by RQPMIN

RQPMIN works by generating a series of trial values of the problem variables. Each set of trial values can be interpreted geometrically as a defining point in n-dimensional space. At each trial point, RQPMIN generates the following auxiliary quantities:

- Lagrange multipliers, λ_i,
- the Lagrangian function, $L(x)$,
- the sum of squares of the constraints, $g(x)$,
- the penalty scale factor, nu,
- the penalty function, $p(x)$,
- the feasibility search vector, u and its norm, and
- the minimisation search vector v and its norm.

The Lagrange multipliers λ_i are a set of auxiliary variables calculated internally, and there is one Lagrange multiplier for every active constraint. The Lagrangian function $L(x)$ is defined by:

$$L(x) = f(x) + \sum \lambda_i c_i(x) \quad , \tag{3-6}$$

and the sum of squares of the constraints is given by:

$$g(x) = \sum c_i(x)^2 \quad . \tag{3-7}$$

In both cases, the summation is over all active constraints.

By default, the penalty function is defined by:

$$p(x) = L(x) + \text{nu}\, g(x) \quad , \tag{3-8}$$

where nu ≥ 0 is the penalty scale factor. However, RQPMIN can be instructed to apply a separate penalty scale factor $n_i \geq 0$ to each active constraint $c_i(x)$, using a process called *cheating*, so that the penalty scale factors are increased in order to force the penalty function to be reduced with the sum of the squares.

The search vectors u and v define directions along which RQPMIN will search for an improved trial point. In ordinary, 3-dimensional space, a direction is defined by three values, but RQPMIN works in n-dimensional space so that u and v consist of n entries, each entry corresponding to a problem variable.

The norm of u is simply a measure of u's length. If u_1 , u_2 , ... , u_n are the individual entries of u, then the ℓ_∞ norm of u is given by:

$$\max_i |u_i| \quad , \tag{3-9}$$

where $|u_i|$ is the absolute value of u_i .

3.3.4 Convergence Criteria

The norms of u and v play important roles, as they define whether or not a trial point is a satisfactory solution of the optimisation problem:

- the norm of u is an estimate of how close the trial point is from the *feasible set*, i.e. the set of points satisfying the constraints exactly;
- thus RQPMIN considers the constraints to be satisfied if the norm of u is less than a specified value, called XTOLU;
- similarly, the norm of v is an estimate of how close the trial point is from a minimum of the Lagrangian function;
- if the norm of v is less then a specified value, XTOLV, then RQPMIN considers the current trial point to be at a minimum of the Lagrangian function;
- it follows that, if both tolerances are satisfied, then the current trial point is considered a satisfactory solution to the optimisation problem.

RQPMIN can also use another measure of convergence, based on the norm of the active constraint functions, their violation tolerances, and the gradient of the Lagrangian function, but this has not been used during this research programme.

3.3.5 Inner Iterations

One of the reasons that RQPMIN has been to found to be a fast solver of optimisation problems is its use of the concept of *perifeasibility*. RQPMIN allows the constraints to be violated by a specified, but variable tolerance, called the radius of perifeasibility, RTOL. Initially, RTOL is given a relatively large value, but as RQPMIN approaches the minimum of the objective function, the constraints automatically become satisfied. Thus, RQPMIN is fast because it does not waste time trying to solve the constraints exactly when far from the minimum.

A trial point is said to be perifeasible if its distance to the feasible set is less than RTOL. As the distance of the trial point from the feasible set is measured by the norm of the feasibility search vector, u, x is perifeasible if

$$\max_i |u_i| \leq \text{RTOL} \quad . \tag{3-10}$$

RQPMIN applies a sequence of iterations, moving from one perifeasible point to another. At each new point, the penalty function is less than at the previous one, and in this way, a minimum value of the penalty function will eventually be reached. The perifeasible points are obtained by applying a series of inner iterations that progressively reduce $g(x)$. If at any iteration RQPMIN is unable to obtain a perifeasible trial point at which the value of the penalty function is reduced, then RTOL is reduced. It can be shown that if the computations were made with infinite precision, then RQPMIN is guaranteed to converge to a valid solution.

During these inner iterations, the set of active constraints remains constant, because the aim of the inner iteration is to find a point that is perifeasible with respect to the current active set. If, during an inner iteration, an inactive constraint becomes violated, the inner constraint is said to have failed. Sometimes the aim of an inner iteration is to find a point that is perifeasible with respect to all the constraints that are not violated at that point. Such an inner iteration may be required at the start of the calculation, if the starting point is not perifeasible, or when the current trial solution is no longer feasible because the active set has changed. In these circumstances, the active set is allowed to change during the course of the inner iteration. To distinguish them, such inner iterations are called *preliminary* inner iterations.

3.3.6 Newton Iterations

The feasibility search vector u is normally computed by the *Gauss-Newton* method. This is the step required to move from the current trial point to a least-squares solution of the linearised constraints. The Gauss-Newton method is generally fast and reliable, but its performance can deteriorate when far from the feasible set. RQPMIN sometimes applies a sequence of inner iterations using the search vector u calculated by a quasi-Newton method; these are called *Newton Iterations*.

Newton iterations are invoked if the sum of squares of the active constraints $g(x)$ exceeds a specified tolerance, or if the standard Gauss-Newton direction is not in a direction that reduces $g(x)$. Once RQPMIN enters a sequence of Newton iterations, it does not emerge from them until no further progress can be made, or the norm of the

standard Gauss-Newton search vector is less than XTOLU and RTOL, and the sum of the squares of the active constraint, g satisfies:

$$g < \text{GTOL} \max_i \left| \frac{\partial g}{\partial x_i} \right| \quad . \tag{3-11}$$

3.3.7 The Minimisation Step

Before carrying out the minimisation step, RQPMIN checks the norms $\|u\|$ and $\|v\|$ of the search vectors u and v. The values are compared with pre-set maximum values, and, if required, reduced so that the value of the norm is equal to the maximum allowable value. If desired by the user, the steplength can be updated automatically by RQPMIN following successful feasibility steps. The command will only be applied provided the steplength is less than the length of the feasibility search vector, the feasibility search vector did not cross any upper or lower bound, and there was no back-tracking required. If back-tracking was required, the steplength is reduced by a value larger than that by which it would have been increased (normally 4 and 0.5 respectively).

If the best trial point found so far is denoted by $x^{(\text{BEST})}$, then during a minimisation step RQPMIN generates a new trial point:

$$x^{(\text{DATUM})} = x^{(\text{BEST})} + u + \beta v \quad . \tag{3-12}$$

where u and v are the feasibility and minimisation step lengths corresponding to $x^{(\text{BEST})}$ and $0 < \beta < 1$ is a steplength initially set to unity. The norms of the vectors u and v may be scaled if $x^{(\text{DATUM})}$ violates the variable bounds, as mentioned earlier, so that the value of x_i is set at the variable bound. In addition, the steplength β may be reduced to ensure that any inactive constraints are not violated at $x^{(\text{DATUM})}$. If necessary, violated inactive constraints may be added to the active set and thus become active after either a preliminary inner iteration or during or after a minimisation step. The process used to add a constraint after a preliminary inner iteration or minimisation step is quite straightforward, but the handling of constraints during a minimisation step is more complicated. Interested readers are directed to Section 3 of the RQPMIN User Guide[92].

By assumption, $x^{(\text{BEST})}$ is perifeasible, but if $x^{(\text{DATUM})}$ is not perifeasible, then a sequence of inner iterations is performed to obtain a trial point that is. Having obtained a perifeasible point $x^{(\text{DATUM})}$, RQPMIN next tests the value of the penalty function at $x^{(\text{DATUM})}$; if this is sufficiently less than at $x^{(\text{BEST})}$ then $x^{(\text{DATUM})}$ becomes the new best point found so far and the minimisation step ends. To be considered a sufficient improvement, the value of the penalty function must satisfy the *descent criterion*:

$$p\left(x^{(\text{DATUM})}\right) \le p\left(x^{(\text{BEST})}\right) + \text{MU}\, \beta v^{\text{T}} \nabla f\left(x^{(\text{BEST})}\right) \quad . \tag{3-13}$$

where $0 \le \text{MU} \le \frac{1}{2}$ is a user-specified constant that prevents the acceptance of infinitesimally small reductions in the penalty function. The complicated expression on the right of Equation 3-13 is simply the scalar product of the minimisation search vector v with the gradient vector ∇f, i.e.:

$$v^{\text{T}} \nabla f = \sum_i v_i \frac{\partial f}{\partial x_i} \quad . \tag{3-14}$$

It can be shown that this is equal to the slope of the penalty function along the search vector v. This slope is always strictly negative, so the above descent criterion ensures that $p(x^{(\mathrm{DATUM})}) < p(x^{(\mathrm{BEST})})$. It is the calculation of the descent direction based on the slope of the penalty function that leads to the description of such processes as **gradient-based optimisation.**

If $x^{(\mathrm{DATUM})}$ does not satisfy the above descent criterion, or if the inner iteration fails (in particular because an inactive constraint has been violated), then the steplength β is reduced and a corresponding new $x^{(\mathrm{DATUM})}$ is generated. If necessary, a new iteration is performed to obtain a new perifeasible point, and the process is repeated until either:

- $x^{(\mathrm{DATUM})}$ satisfies the descent criterion, when the minimisation step is deemed to have succeeded, or
- one or more of the inactive constraints were violated during the inner iteration and β is reduced below the user-specified constant CTOL (in which case the violated constraints are added to the active set, β is reset to unity, and the minimisation step is repeated), or
- β is reduced below a user-specified minimum, in which case the minimisation step has failed.

RQPMIN's efficiency would suffer if it performed inner iterations at points distant from the feasible set. To prevent this, an outer radius of perifeasibility is used, so that if the distance of $x^{(\mathrm{DATUM})}$ from the feasible set is greater that this radius then no inner iteration is attempted. Instead, the steplength β is reduced exactly as though the inner iteration had failed.

3.3.8 The Feasibility Step

The details of the feasibility step are similar to that of the minimisation step, and $x^{(\mathrm{DATUM})}$ is again used to denote the best point found so far during the current inner iteration. As in the minimisation step, if $\|u\|$ is greater than its corresponding limit, it is scaled, and a new trial point $x^{(\mathrm{TRIAL})}$ is generated:

$$x^{(\mathrm{TRIAL})} = x^{(\mathrm{DATUM})} + \alpha u \quad , \tag{3-15}$$

where u is the feasibility search vector corresponding to $x^{(\mathrm{DATUM})}$ and α is the steplength, initially set to unity. As with the minimisation step, u and its norm may be scaled if $x^{(\mathrm{TRIAL})}$ violates the variable bounds.

RQPMIN tests the value of the sum of squares of the constraint function at $x^{(\mathrm{TRIAL})}$, and if this is sufficiently less than $x^{(\mathrm{DATUM})}$ then $x^{(\mathrm{TRIAL})}$ becomes the new best point found so far and the feasibility step ends. To be considered sufficient improvement, the sum of the squares of the constraints must satisfy the descent criterion:

$$g\left(x^{(\mathrm{TRIAL})}\right) \le g\left(x^{(\mathrm{DATUM})}\right) + \mathrm{MU}\ \alpha u^{\mathrm{T}} \nabla g\left(x^{(\mathrm{DATUM})}\right) \quad . \tag{3-16}$$

The rest of the explanation is identical to the for the minimisation step, with 'feasibility' and 'α' replacing 'minimisation and 'β', respectively.

3.3.9 Other Functions

The RQPMIN User Guide[92] contains details of the other functions and processes performed by RQPMIN; the noise reduction filter, how RQPMIN fixes and frees variable from their bounds, addition and removal of constraints from the active set, the Hessian matrix, QR factorisation, and others, are explained. The only other point pertinent to this thesis is the approximation of the derivatives, which, in the chapter introduction, was stated as one of the biggest challenges in optimisation.

In the optimiser input file, the user can indicate whether the partial derivatives of a particular problem function should be calculated analytically or by finite differences. The methods used in the aircraft design and LCC procedure involve the calculation of complex objective functions, and as such do not lend themselves to analytical solution. To overcome this, RQPMIN usually employs the *forward differences formula*:

$$\frac{\partial f}{\partial x_i} \approx \frac{f(x + \text{DIFTOL}\, e_i) - f(x)}{\text{DIFTOL}} \quad , \tag{3-17}$$

where f denotes an arbitrary problem function (not necessarily the objective function), e_i is the ith co-ordinate vector (the vector consisting of n zeros, except for the ith entry, which is unity), and DIFTOL is the user-specified stepsize. Thus the symbol x + DIFTOL e_i denotes the same vector as x, except that the DIFTOL has been added to the ith entry.

If $f(x + \text{DIFTOL}\, e_i)$ cannot be computed, because x would violate the function bounds, then RQPMIN attempts to use the *back ward differences formula*:

$$\frac{\partial f}{\partial x_i} \approx \frac{f(x) - f(x - \text{DIFTOL}\, e_i)}{\text{DIFTOL}} \quad , \tag{3-18}$$

or alternatively the *central differences formula*:

$$\frac{\partial f}{\partial x_i} \approx \frac{f(x + \text{DIFTOL}\, e_i) - f(x - \text{DIFTOL}\, e_i)}{\text{DIFTOL}} \quad . \tag{3-19}$$

Central differences are more accurate, but require twice as many function evaluations as the other two difference calculations. RQPMIN attempts a compromise by first solving the problem using forward differences only; when the calculation is finished (whether successfully or not) it is restarted from the best point using central differences.

3.3.10 Program Termination

When RQPMIN stops progression of the synthesis, either because it has found a solution, or because it has encountered an error, the reason will be indicated on the screen. If an error has been detected, a diagnostic message is also printed in the optimiser results file. Stop messages and diagnostics are discussed in Section 6.6 and Appendix A, respectively, of the RQPMIN User Guide. Regardless of the reason for termination of the optimisation process, the relevant output files are written for the optimiser and other program modules. From the results of the above files, and the stop message produced by RQPMIN, the user will be able to check whether a satisfactory solution has been found.

3.4 Summary

Chapter 3 has:

- Presented the basic theory of unconstrained and constrained optimisation in a clear and relatively simple fashion.
- Described some of the problems inherent with gradient-based optimisation processes, namely discontinuities and local minima.
- Explained the operation of the chosen optimisation tool, RQPMIN, including the method of linearisation of the function constraints and the approximation of the derivatives.
- Given details of some of the processes particular to RQPMIN to improve its efficiency, including the concept of perifeasibility.

This page is intentionally blank.

4 Program Structure and Airframe Definition

This chapter describes the overall structure of the aircraft synthesis, costing, and optimisation models, and their interrelation. It also explains the methods and calculations required to determine the size of the aircraft, estimate its mass, and predict the internal volume available for fuel storage. The model structure is explained first, and lists are given of some of the other models available to the individual synthesis and optimisation modules. A brief description of the aircraft design and costing variables available is given, along with some of the initialisation calculations.

The aircraft sizing method is explained, and is split into the main airframe components; fuselage, wing and empennage. Separate sub-sections detail the sizing procedure for each, and explain the choice of variables and constraints imposed to produce a realistic design. All of the sizing algorithms have been developed specifically for this model, and are aimed very much at the conceptual design level, with a minimum number of variables being used to generate realistically accurate models.

Once the major aircraft dimensions have been determined, the mass estimation models for the major aircraft components and systems are described in detail. The mass model is made up of a set of parametric equations, each of which is used to estimate the mass of a single component or system. The individual system masses are summed to give the aircraft empty mass, which, with the addition of flight crew, mission fuel, and weapons, forms the aircraft all-up mass. The iteration to convergence of the total aircraft mass is explained, and a comparison of results for a range of modern combat aircraft is given.

Building on the aircraft geometry, component mass, and standard system density values, volume estimation calculations are performed to complete the aircraft synthesis phase of the design. The total fuselage volume is calculated, from which the engine bay, intake ducts, and other major component and system volumes are subtracted. A fraction of the remaining space is added to the total wing volume available for fuel storage, to calculate the overall fuel storage volume available.

4.1 Description and Overall Structure

The aircraft synthesis and optimisation model is implemented via a large FORTRAN code containing many separate modules. This chapter, and the following three, describe the modules, and Figure 4-1 gives a schematic representation of the overall operation of the whole program, which is called COMBAT for ease of reference.

It can be seen that the optimiser has ultimate control of the program, and is responsible for altering the aircraft and engine design parameters such that all constraints are met, and a minimum objective function is achieved. The nature of the optimisation process, explained in Chapter 3, and the iterative nature of the design procedure, may prevent the linear solution path described below from being followed, but this should not detract from the explanation of the individual models.

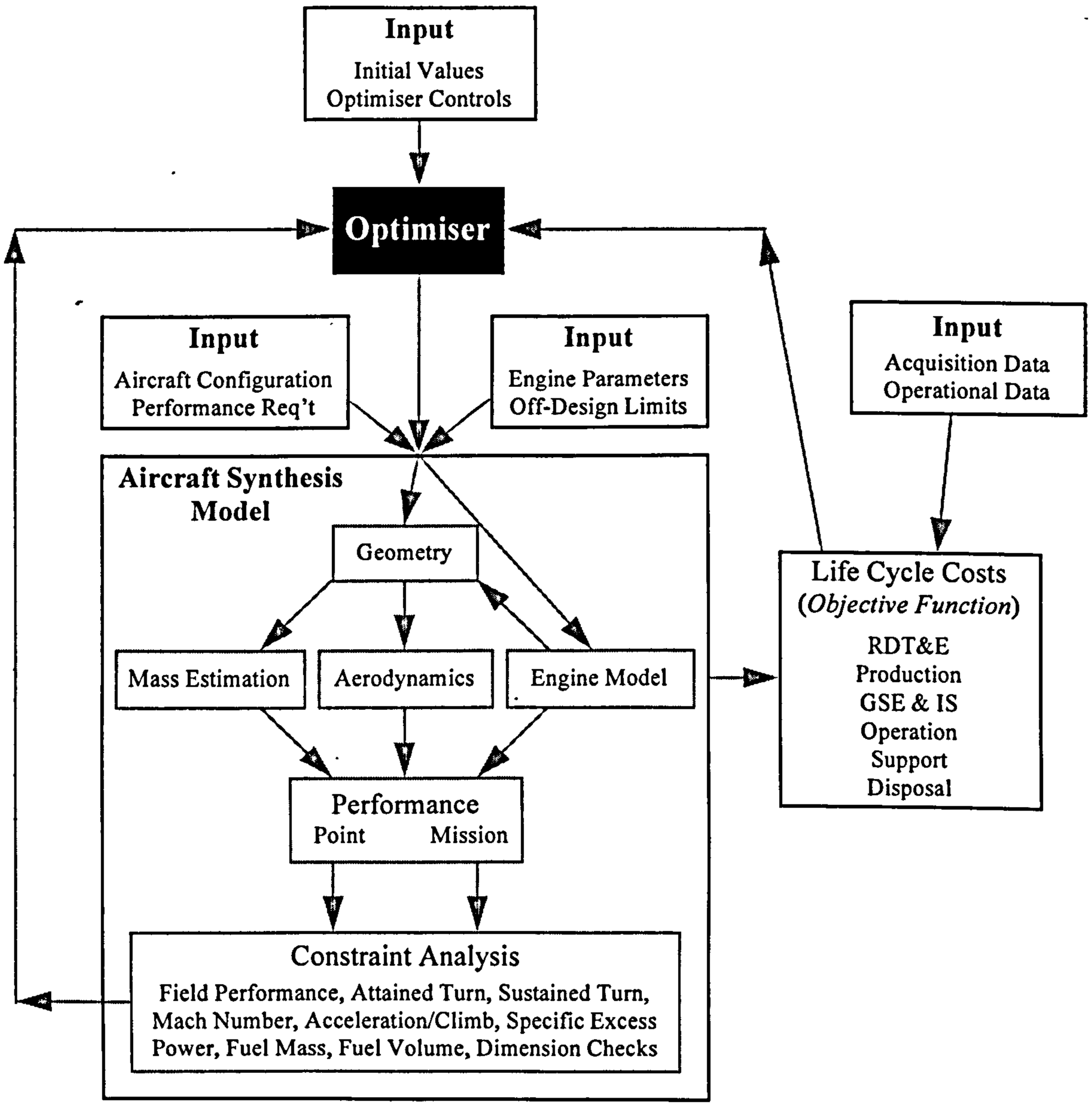


Figure 4-1. Overall Program Flowchart.

4.1.1 File Read and Variable Initialisation

Once the program is started, RQPMIN opens the optimiser input file, which contains all of the data required for the initial trial solution, constraint values and tolerances, and optimiser control values. Trial solution data gives start values for all of the aircraft variables controlled by the optimiser in order to produce the 'baseline' design, whether they are being included in the optimisation procedure, or not. The constraint values are used to check that the final geometry and performance of the aircraft lie within the prescribed limits, enabling the user to ensure that a feasible design is generated.

RQPMIN then calls the aircraft synthesis routine, which captures the trial design data and opens the aircraft, engine, and aircraft life cycle files. These files contain values that describe the aircraft configuration, performance requirements, mission, and engine parameters in detail, and feed into the aircraft synthesis and analysis models. Many aircraft design parameters are then initialised to their starting values, based on the optimiser, aircraft, and engine data input files, and simple empirical algorithms.

The parameter initialisation process also calls the engine design program, ONX, and the engine thermodynamic cycle design is performed using the data from the engine input file. This file also contains the off-design limits for the engine, adding to the realism of the methodology, by restricting its thermodynamic operating envelope. A description of the ONX engine thermodynamic design model is given in section 5.2. Once the thermodynamic cycle of the engine has been set, it is not altered, and all subsequent engine calculations are performed to analyse the engine performance away from its design point.

The above actions (i.e. file open & read, and engine thermodynamic design) are only performed on the very first call to the synthesis module. All of the following procedures are performed every time the synthesis is called.

4.1.2 Component Sizing

Although the engine thermodynamic cycle has been specified, the physical size of the engine is yet to be defined. The main parameter used to determine the engine size (in terms of both engine thrust and physical dimensions) is the air mass flow rate, the value of which is supplied by RQPMIN. The engine off-design analysis program, OFFX, is called at sea-level static conditions, and the values from this run, together with the original engine design data, are used to calculate the physical dimensions of the engine, as shown in section 5.2.4. The engine intake area and maximum nozzle area are generated at this stage of the design, and are used for the aircraft geometry, mass estimation, and drag prediction methodologies described in later chapters.

The next stage in the design process is to size the remainder of the aircraft so that an overall configuration can be studied. The methods used have been developed specifically for this purpose, and enable the rapid sizing of components based on historical data. The dimensions that are predicted in this section of the program are: fuselage height, width, and height to the centreline; wing centreline, root, tip and mean aerodynamic chord; fin and tail areas, aspect ratios, thickness/chord ratios and mean chords. Approximations are also made for flap parameters, for use in the aerodynamic drag model. Details of airframe sizing calculations are given in section 4.2.

4.1.3 Mass Estimation and Volume Accounting

The mass estimation method[45] used in the aircraft design synthesis model was developed at Cranfield, and makes use of other in-house methods, which are particularly suited to the estimation of aircraft component and system masses at the conceptual design stage. The methods are empirical derivations based on historical data, although some changes and improvements have been made to the basic algorithms to account for the inclusion of composite materials and other design options.

In order to produce a realistic aircraft design, the volume available for fuel storage must be estimated, so that necessary provision for all of the required design mission fuel can be made. This has been achieved by estimating the volume available for fuel storage in integral tanks in the wings, and by allowing a fraction of the available volume in the fuselage to be filled with fuel. This fraction is applied to the volume remaining after that required for the separate systems has been subtracted from the total fuselage volume. The volume of the internal systems is calculated using pre-defined densities, while the overall fuselage volume is estimated using simple geometric shapes. Separate, distinct, fuselage fuel tanks are not considered, allowing a check of fuel capacity to be made without the need for individual fuel tank variables, avoiding the reduction in optimisation robustness that this can cause. A full description of both methods, and results of a mass correlation for many combat aircraft are given in section 4.3.

4.1.4 Aerodynamic Estimation

The aerodynamic model is made up of three sections that predict available lift coefficient, angle-of-attack (AoA), and aircraft drag, respectively. All three models are based on empirical correlations, with the first two making use of relatively simple empirical algorithms for their aerodynamic prediction. The drag model is much more complex than the lift and AoA models, as it breaks the overall drag value down into smaller parts, and relies on the interpolation of many sets of empirical tables for the estimation of each part.

The models were chosen and developed for their applicability to low-aspect ratio wing configurations, and their ability to predict the relevant aerodynamic properties in both sub- and supersonic speed regimes. Although there are many more complex and accurate methods for the prediction of aerodynamic performance, they rely on detailed aircraft design data, can be computationally expensive, and do not always provide the smooth changes in constraint derivatives required by the optimiser. It is felt that the models chosen provide the correct balance of accuracy and efficiency, and are in keeping with the overall aims of the COMBAT aircraft design and optimisation synthesis.

4.1.5 Propulsion Modelling

As mentioned in previous chapters, COMBAT makes use of the ONX and OFFX models to estimate the thermodynamic performance of the engine. The source code for these models was kindly provided by Dr. Jack Mattingly of the University of Seattle, which has enabled a new level of accuracy and flexibility of propulsion modelling to be used. Although the ONX and OFFX models are slightly larger and more accurate than required by the rest of the synthesis model, they provide an excellent base on which to build.

The original models are capable of dealing with all types of two-shaft gas turbine engines, including turboprops and ramjets, though the current use of the models is restricted to two-shaft turbofan and turbojet engines, with and without reheat. Many separate engine parameters can be altered to represent different levels of technology and the size and performance of the engine can be completely changed using the bypass ratio, pressure ratio and air mass flow rate variables.

ONX and OFFX are very similar models, but perform completely different roles within the synthesis. ONX is responsible for the thermodynamic design of the engine, and the initialisation of all engine-dependant parameters, whilst OFFX performs the off-design analysis, returning available thrust, and throttled thrust and TSFC, at any reasonable flight condition. A separate module controls the engine models, and accounts for the installation losses incurred during the propulsion system integration.

The final section of the propulsion modelling procedure is the prediction of the engine mass and dimensions. The algorithms were developed specifically for the COMBAT synthesis model, producing accurate results that reflect the variation of key engine variables. This has proven to be a major boon to the synthesis model, as changes in the thermodynamic cycle of the engine not only affect the performance of the aircraft, but impacts on its performance indirectly, by affecting the aircraft external shape and internal volume distribution.

4.1.6 Performance Calculations

The aircraft synthesis model is capable of establishing the point performance of the designed aircraft for a number of different design manoeuvres and flight phases. The manoeuvres and phases that can be compared to required values are; take-off, landing, attained turn, sustained turn, maximum level Mach number, acceleration/climb time, and specific excess power. For each of these phases, the user can set the altitude, Mach number, the required performance value, and four other parameters. These include fuel and payload fractions, reheat switch, and one other manoeuvre-specific value, as shown in Table 6-1 and Table 6-2, section 6.1.

Up to 10 separate point performance requirements can be entered in any order, giving a very powerful aircraft performance definition tool. By restricting the use of reheat, the user is also able to specify that the aircraft should have 'supercruise' capability, and other features, such as thrust reversal effectiveness for landing, have also been included. The calculations are based on classical performance methods, and are described in detail in section 6.1.

Up to 30 different mission legs can be supplied for analysis in the aircraft input file. The phases are; engine run (maximum or military thrust for a given time), take-off, climb/accelerate/descend, cruise, manoeuvres, weapons drop, loiter/Combat Air Patrol (CAP), and landing. For each of the mission legs, altitude, Mach number, two phase-specific data values (such as range for the cruise leg), and the use of reheat, can be specified as shown in Table 6-3 and Table 6-4, section 6.2. Time, distance travelled, and fuel burnt during the mission leg is calculated, although horizontal distance is **not** accounted for during climb or descent manoeuvres.

The large number of mission phases and options available make the mission performance calculations very powerful and flexible, and further options exist within some of the mission phase calculations. For instance, when the aircraft is in turning flight or combat, the duration can be given in time (minutes), or the total angle through which the aircraft must turn. The required turn rate (in g's) and flight condition can be specified at the end of any mission leg, and this will be treated as another constraint. This feature is very useful, as it is often not known, at the mission specification or operational analysis stage of the design, exactly what the fuel load will be at this point in the design mission. Supercruise can also be guaranteed during the mission, by limiting the use of reheat. The ratio of available to required thrust is an optimiser constraint, ensuring that there is sufficient thrust to overcome drag.

The time, distance, and fuel load are summed at the completion of the mission, to give the total values. These feed into the relevant section of the rest of the synthesis model, as explained in later chapters.

4.1.7 Life Cycle Cost Model

The Life Cycle Cost (LCC) model is a stand-alone section of the code, and is made up of several subroutines, one for each of the main life cycle phases. These are; Research, Development, Test & Evaluation (RDT&E); Production; Ground Support Equipment and Initial Spares (GSE&IS); Operation and Support (O&S); and Disposal. A full description of the models used is given in Chapter 7.

The total costs for each phase are calculated using a combination of activity-based and function-driven parametric equations. Extensive use of existing models was made, as the data required for the generation of these models has proven extremely difficult to obtain, and the analysis required is very time-consuming. Where possible, each of the separate phases of the model has been calibrated against known values to increase confidence in the overall prediction capability.

The RDT&E, Production, and GSE&IS costs are calculated on the total number of aircraft built, whilst the O&S costs take account of the basing strategy employed for the operation of the designed aircraft. This gives another variable to be investigated, should the particular study require it. As the Operation & Support and Disposal life cycle phases take place over a period of years, and some time in the future, respectively, a discounting method is used to predict the total deflated costs. The discounting process follows the procedures outlined in the Government Green Book[7], and allow for a very efficient calculation process.

4.1.8 Aircraft Synthesis Output Routines

Once RQPMIN has determined a result of the optimisation, converged or not, the output routines are initiated, and results of the design study are written to the respective files. These are written to allow the accurate comparison of competing designs, and contain enough data for a sketch of the aircraft to be produced, a feature that has not yet been included, but could be added at a later stage in the program development. The results of the LCC analysis are also given, and are broken down into the life cycle phases explained above, together with a breakdown for a single aircraft, for easy comparison.

Two other sets of data are optionally produced: the drag polar file, and the turn rate file, which contains two sets of data, one for the attained turn rate (instantaneous turn), and one for the sustained turn rate. The drag polar file is produced mainly to provide a 'reality check' for the design that has been produced, and gives the drag coefficient for varying Mach numbers between M = 0.1 and the maximum Mach number. Drag coefficients varying with lift coefficient values between 0 (zero-lift drag), and 0.7 are also given, as this would be the maximum normal range of C_L for wings operating without trailing-edge high-lift devices (e.g. in a sustained turn).

The attained turn rate and sustained turn rate are written in a single file, and show the predicted performance for the designed aircraft. These charts could be used for the comparison of competing conceptual designs, and are not intended to exactly reproduce performance charts of existing aircraft for which real turn rate data has been obtained. Whilst every effort has been made to ensure that all the methods are as accurate as possible, slight differences between real and modelled performance in the engine performance or drag models are amplified at the extreme levels of performance being modelled in these cases. Inaccuracies will also occur due to the highly non-linear and configuration-dependant aerodynamic drag forces generated at high angles of attack. A description of the methods and comparison of the results is given in Chapter 6.

4.1.9 Other Modules

The COMBAT aircraft synthesis and sizing model makes use of several other models that are contained in the main program. An example of one such model is the atmosphere module, which forms an important part of the synthesis program, as changes in air properties will have a significant impact on the performance of the aircraft and engine. In order to model the above effects, one subroutine performs all necessary calculations for the changes in atmospheric properties with altitude, and the values correspond very well with those given in tables of properties for the International Standard Atmosphere.

The model uses linear equations to predict changes in temperature, and exponential algorithms for pressure drop; the remaining atmospheric values are calculated from these two values and the altitude. The inputs required are altitude and temperature difference from ISA, from which are returned density, pressure, speed of sound, kinematic viscosity, gravitational acceleration, relative pressure and relative temperature. The change in gravitational acceleration with geometric height is based on a simple linear decrease in g with altitude, such that $G = 9.80665 \times (1 - 3.14\times10^{-7} \times HT)$, where HT is the geometric altitude. This may seem an unnecessary complication, but as it was already contained in the atmosphere subroutine, it would have been more work to remove it than to leave it in place.

Another set of algorithms frequently made use of is the interpolation subroutine, which contains equations for the interpolation of data points from linear models, conic/cubic splines and combinations of the two. The routines are used extensively by the drag model, and are also called by the engine modelling routines during the estimation of inlet and nozzle losses.

4.2 Airframe Component Sizing

The aircraft sizing process has been kept deliberately simple, in order to keep the number of variables to a minimum, and improve robustness of the code. A large number of design variables can cause the optimiser to become trapped in local minima, and reduce the chances of true convergence, due to the increased number of solution paths. It is also undesirable to generate variables that are not properly reflected in the other sections of the code, particularly the mass and aerodynamic modules. Figure 4-2 shows the overall sizing of the aircraft, and the relevant major airframe design variables. The following sections provide the equations for the derivation of those variables, and give method correlations where applicable. All variable names shown are the same as contained in the aircraft synthesis code, and conform to a standardised naming convention, based on the method used by Lovell[63].

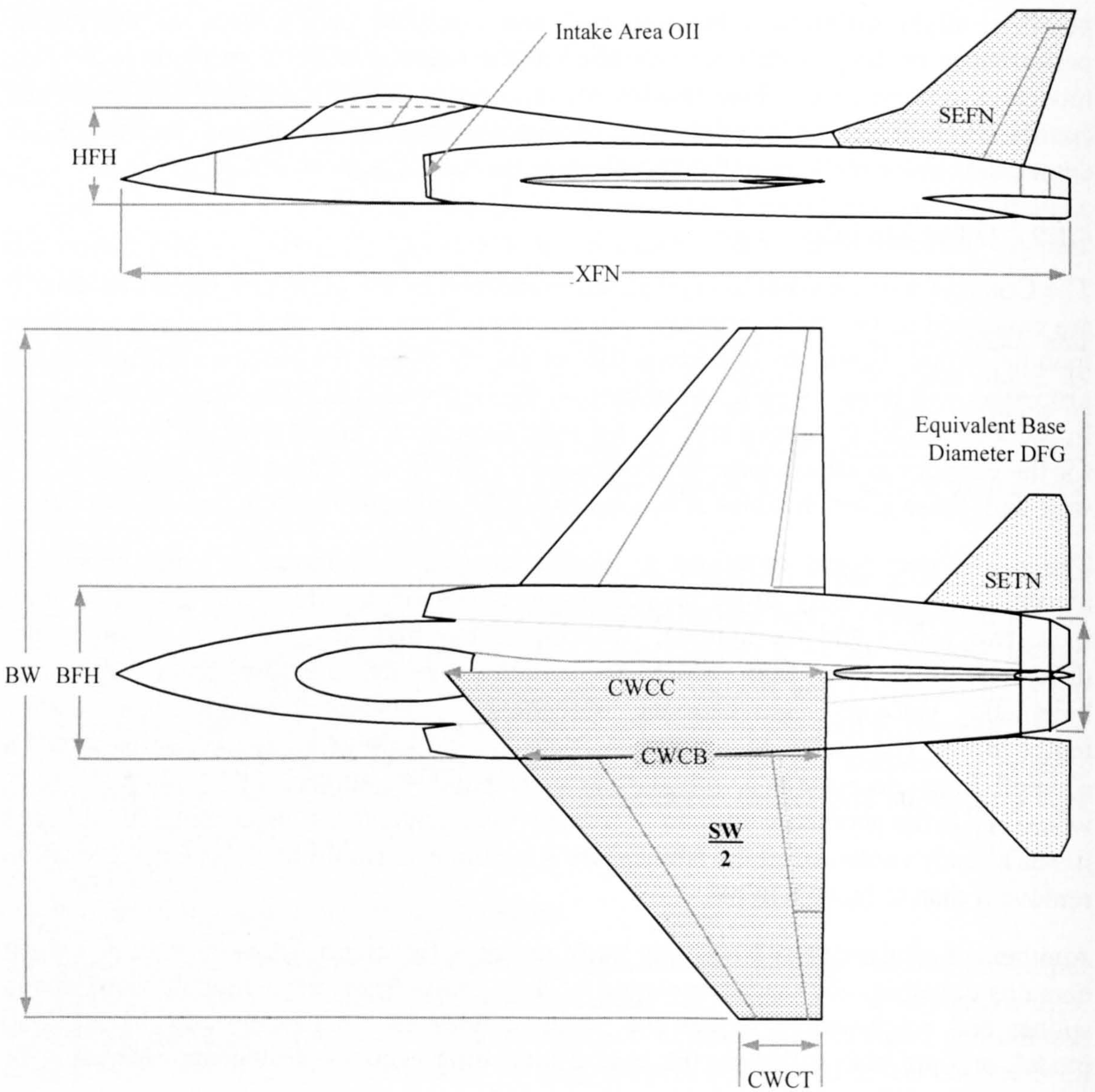


Figure 4-2. Aircraft Variable Definitions.

4.2.1 Aircraft Description Variables

The text below gives a list and brief description of the type and definitions of the aircraft definition variables available in the COMBAT design synthesis program. A range of values is shown, so that the reader has some idea of the flexibility and application of the models, which are explained in the following sections.

Variable	Values	Description
IOPT	1 - 4	objective function selection. 1 - all-up-mass, 2 - recurring flyaway cost, 3 - acquisition cost, 4 - life cycle cost
ISCRN	0 - 1	write independent variables, function constraints, objective function, and design convergence checks to screen. 0 - off, 1 - on
IDRAG	0 - 1	write aircraft drag polar at end of optimisation. 0 - off, 1 - on
IPRF	0 - 1	write turn performance at end of optimisation. 0 - off, 1 - on
ROLE	1 - 3	intended role of aircraft. 1 - fighter, 2 - strike, 3 - trainer
TAIL	1 - 4	aircraft configuration. 1 - conventional (aft-tail), 2 - conventional & leading-edge extension (LEX), 3 - delta-wing, 4 - delta-canard
NCP	1 - 2	number of crew members
NENG	1 - 2	number of engines
IAPU	0 - 1	auxiliary power unit on aircraft. 0 - no, 1 - yes
NEF	1 - 2	number of fins
TYPE	1 - 2	aerofoil type for subsonic wings. 1 - conventional, 2 - supercritical
IFLAP	1 - 2	flap type. 1 - single-slotted, 2 - double-slotted
SLAT	0 - 1	leading-edge device (flap or slat) available. 0 - no, 1 - yes
QFLA	0 - 40	landing flap setting angle for flap pressure drag calculations. (°)

The user is also required to enter data that determines the overall design requirements of the aircraft:

Variable	Values	Description
MMAX	0 - 2	maximum high-altitude (> 11000m) design Mach number.
MLOW	0 - 1.5	design diving Mach number for structure mass calculations.
GND	0 - 10	aircraft design limit load. (g's)
MBMAX	0 -	maximum allowed aircraft payload. (kg)
MAXI	0 -	installed avionics mass. (kg). If 0.0, will be estimated by COMBAT
NHARD	0 - 15	number of pylons available for weapons/fuel carriage.
MBX	0 -	mass of external payload to be carried on design mission. (kg)
MTGFXI	0 -	mass of external fuel allowed for design mission. (kg)
MBI	0 -	mass of design mission internal payload - no drag penalty. (kg)
VBI	0 -	volume required for internal weapons bays. (m^3)
MGCA	0 -	mass of gun and ammunition. (kg)

4.2.2 Fuselage Sizing

The size of the fuselage is determined by RQPMIN using the variables XFN and DFH, the maximum fuselage length, and the maximum effective fuselage diameter. From these two variables, and engine parameters calculated during the initialisation routines, the remaining fuselage dimensions can be estimated.

The width and height dimensions are driven by the maximum effective diameter, DFH, whether the aircraft has one or two engines, and the size of those engine(s). Both the height and width could have been given to RQPMIN for individual control, but a single variable was felt to be more appropriate, as explained at the start of section 4.2. The diagram and equations below give a simple explanation of the sizing methodology.

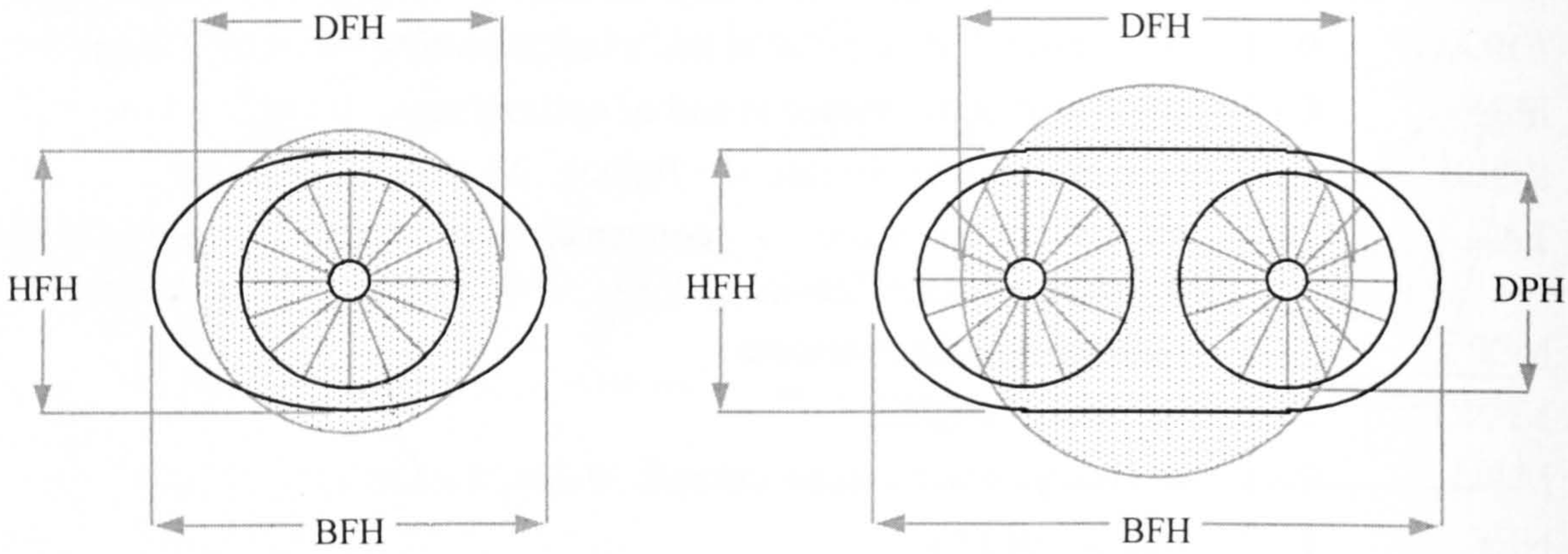


Figure 4-3. Definition of Fuselage Width and Height Dimensions.

For the single-engine case it is assumed that the fuselage will have an elliptic shape and a cross-sectional area equivalent to that given by the effective diameter, with a ratio of major to minor axis of 1.2, and a minor axis at least 10% larger than the engine maximum diameter:

$$HFH = \sqrt{\frac{DFH^2}{1.2}} \ \text{(m)}, \ BFH = 1.2 \times HFH \ \text{(m)}, \text{ and} \tag{4-1}$$

$$FUNC(4) = 1.1 \times DPH - HFH \ ,$$

The dimensions for the twin-engine case are calculated from the maximum engine diameter and the effective fuselage diameter:

$$HFH = 1.2 \times DPH \ \text{(m)}, \quad BFH = \frac{\frac{\pi DFH^2}{4} - \frac{\pi HFH^2}{4}}{HFH} + HFH \ \text{(m)}, \text{ and} \tag{4-2}$$

$$FUNC(4) = 2.4 \times DPH - BFH.$$

Where:

BFH and HFH are the fuselage maximum width and height, respectively (m); DFH is the effective fuselage diameter (m), which is an input from RQPMIN; and DPH is the maximum engine diameter (m), calculated earlier in the synthesis code.
FUNC(4) is the RQPMIN constraint responsible for controlling the minimum effective fuselage diameter for both the single and twin-engine cases.

The engine intake area, OII, is calculated from the number of engines (NENG), the ratio of the intake to entry streamtube area (RA1A0), and the reference streamtube area (OFFDAT(5)), one of the values from the off-design analysis program OFFX, so that:

$$OII = NENG \times RA1A0 \times OFFDAT(5) \ (m^2). \tag{4-3}$$

The fuselage base drag area, OFGB, is calculated similarly, making use of the ratio of the exit-to-entry streamtube area (A10A0), the reference streamtube area (OFFDAT(5)), and the engine intake area calculated above. The base drag area is used in the drag prediction routines, and is shown in greater detail in section 5.1.3.

$$OFGB = 1.25 \times NENG \times A10A0 \times OFFDAT(5) - OII \ (m^2). \tag{4-4}$$

The equivalent base diameter is calculated from the base drag area (OFGB), and the engine intake area (OII), and is used in the calculation of the total fuselage volume.

$$DFG = \sqrt{\frac{4 \times (OFGB + OII)}{\pi}} \ (m). \tag{4-5}$$

The fuselage centreline height off the ground, HFM, is calculated by assuming that the undercarriage will be situated at 40% from the rear of the fuselage, and allows a 15° rotation angle before a rear fuselage strike occurs. Thus:

$$HFM = 0.4\ XFN \times TAN(15) + \frac{HFH}{2} \ (m). \tag{4-6}$$

It should be noted that all angular functions in FORTRAN are calculated in radians.

4.2.3 Wing Sizing

All of the main wing sizing parameters are determined by RQPMIN, and the remaining dimensions are calculated from the optimiser-provided values of gross wing area (SW), aspect ratio (AW), taper ratio (UW), leading-edge sweep (QWL), and thickness/chord ratio (RTW). The wing sweep angle from RQPMIN is given in degrees for ease of use, but is immediately converted to radians, as mentioned above.

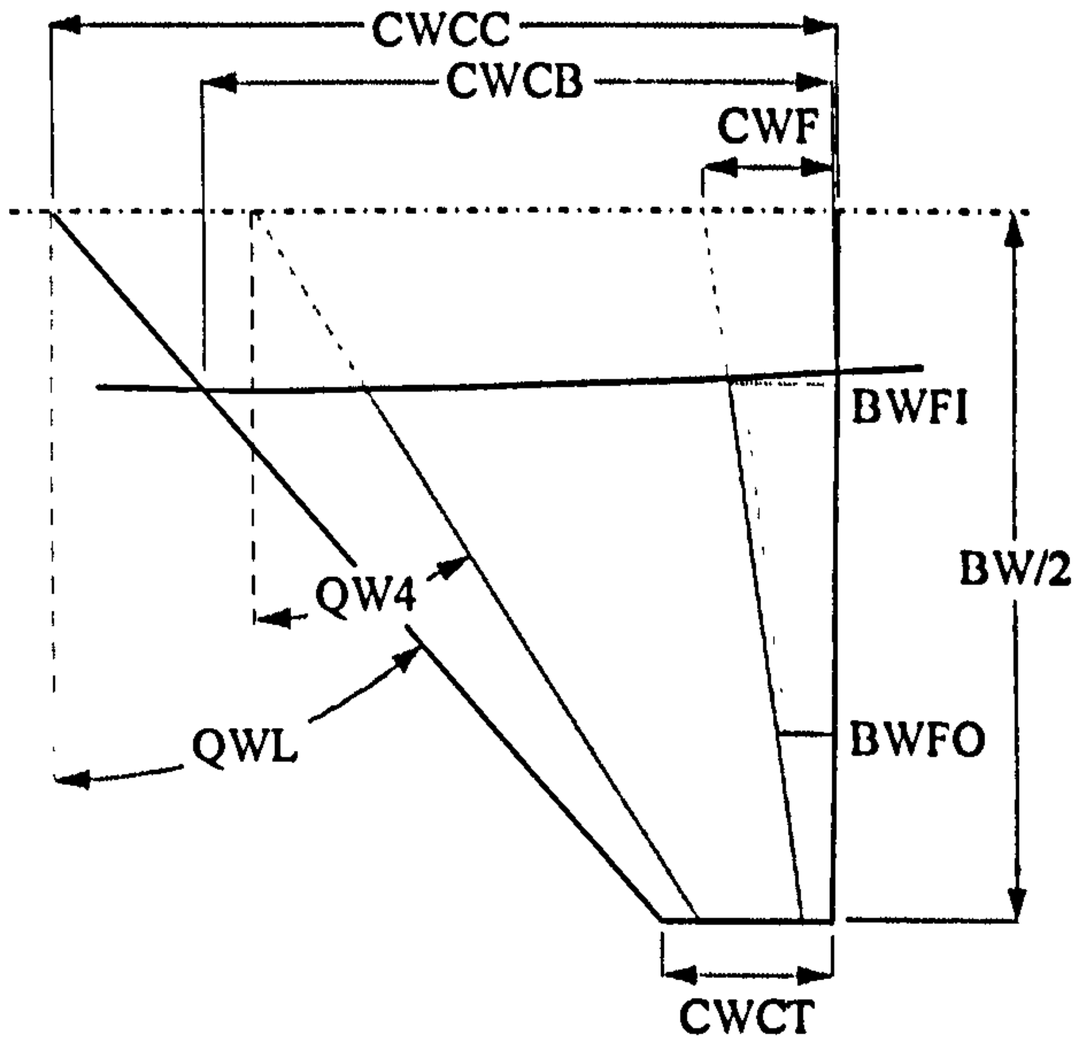


Figure 4-4. Definition of Wing Dimensions.

One of the most important parameters for the calculation of aerodynamic lift is the sweep of the quarter-chord line (QW4). This is calculated from the above values as follows:

$$QW4 = TAN^{-1}\left[TAN(QWL) - \frac{(1-UW)}{AW \times (1+UW)}\right] \text{ (radians).} \quad (4\text{-}7)$$

The wing span is simply calculated from the wing area (SW) and the aspect ratio:

$$BW = \sqrt{SW \times AW} \text{ (m).} \quad (4\text{-}8)$$

The centreline chord, CWCC, is a useful parameter, as it makes the calculation of the other wing dimensions much simpler, so that:

$$CWCC = \frac{2}{1+UW}\sqrt{\frac{SW}{AW}} \text{ (m).} \quad (4\text{-}9)$$

From the centreline chord, the tip chord (CWCT) and the root chord (CWCB), which is used to calculate the wing volume available for fuel, are calculated using the above values and the fuselage width (BFH):

$$CWCT = UW \times CWCC \text{ (m), and} \quad (4\text{-}10)$$

$$CWCB = CWCT + \left[\left(1 - \frac{BFH}{BW}\right) \times (CWCC - CWCT)\right] \text{ (m).} \quad (4\text{-}11)$$

The mean aerodynamic chord is used in the drag estimation procedure, and is calculated as follows:

$$CWMA = \frac{4}{3}\sqrt{\frac{SW}{AW}}\left(\frac{1+UW+UW^2}{(1+UW)^2}\right) \text{ (m).} \quad (4\text{-}12)$$

The flap dimensions shown in Figure 4-4 are never calculated explicitly, but are set as ratios of the wing span and chord. Thus, the inner flap-joint parameter is called FBWFI, and is the ratio of the length from the centreline to the inner flap joint, divided by the wing semi-span. The outer flap length ratio, FBWFO, is treated similarly, whilst the flap chord is expressed as a ratio of the flap chord to the wing chord at any point.

These last two values are simply set from historical data, and the flap chord ratio, RCWF, depends on the type of wing. The values are not especially significant, as they are only used in the calculation of flap pressure drag for take-off and landing.

The equations for the flap parameters are:

$$FBWFI = \frac{BFH + 0.2}{BW},$$

FBWFO = 0.75, and (4-13)
RCWF = 0.2 for aft-tail aircraft, or
RCWF = 0.15 for delta and delta-canard configurations.

All necessary dimensions for the wing have now been specified, except for the available wing fuel volume, which is calculated in section 4.3.4.

4.2.4 Sizing Correlations

In order to determine the equation for a sizing correlation, cost estimating relationship, or any other parametric equation, some measure of the accuracy of the equation is required. In order to establish how good the forecast will be by using the derived equation, the equation must be checked for "goodness" of fit; this is done using the **coefficient of correlation** (R), and the related **coefficient of determination** (R^2). There are other statistics that could be checked to expand knowledge of the regression equation, but normally the two mentioned above are sufficient.

"The coefficient of determination (R^2) represents the proportion of variation in the dependent variable that has been explained or accounted for by the regression line."[19] The value of the coefficient of determination may vary from zero to one. A coefficient of determination of zero indicates that none of the variation in Y is explained by the regression equation; a value of one indicates that 100 percent of the variation in Y has been explained by the regression equation.

In order to calculate the R^2, the following equation is used:

$$R^2 = \frac{\left(\sum xy - n\bar{x}\bar{y}\right)^2}{\left(\sum x^2 - \bar{x}\sum x\right)\cdot\left(\sum y^2 - \bar{y}\sum y\right)} \qquad (4\text{-}14)$$

Graphically, when the R^2 is zero, the observed values appear as in Figure 4-5c, and when the R^2 is one, the observed values all fall exactly on the regression line, as in Figure 4-5a and b.

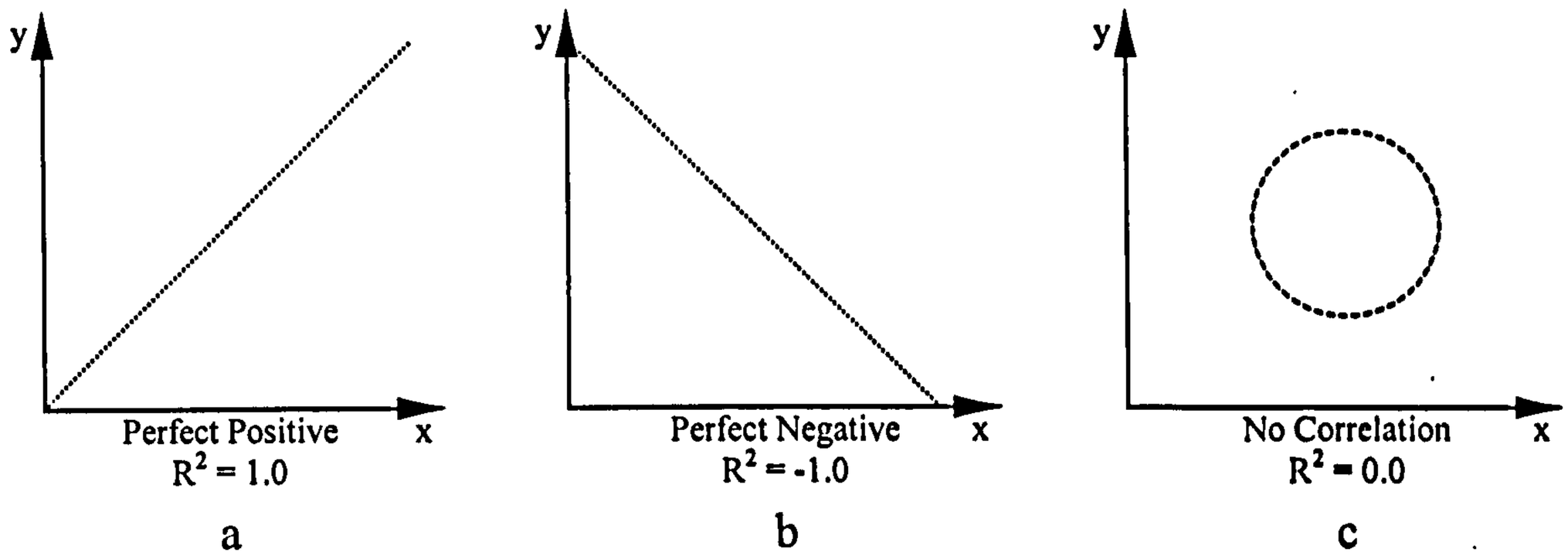


Figure 4-5. Graphical Representation of Coefficient of Determination.

The coefficient of correlation (R) measures both the strength and direction of the relationship between X and Y. The meaning of the coefficient of correlation is not as explicit as that of the coefficient of determination. The value of R will be either positive or negative, depending on the gradient of the regression line, i.e. R takes the same sign as the slope.

It is important to note that R does not indicate how much of the variation in Y is explained by the regression line. R is only valuable in indicating whether the relationship is direct or inverse, and as a general indicator of the strength of the association. For this reason, where a graph of correlation is presented, R is ignored.

4.2.5 Empennage Sizing

The sizing of the empennage is performed by COMBAT using some simple, yet highly accurate parametric sizing equations that were developed in-house, specifically for this methodology. The equations are based on data gathered from references such as Jane's All the World's Aircraft[98], and the results are used in the aircraft mass modules, and the drag estimation procedures.

The tailplane sizing methodology is a function of maximum fuselage length, XFN, and gross wing area, SW. The equation for calculating the net tailplane area, SETN, is:

$$SETN = A \times \left[\frac{0.9438}{XFN^{0.2344}} \times SW^{0.6588} \right]^{1.1877} \quad (m^2). \qquad (4\text{-}15)$$

Where:

A = 1.0 for conventional, aft-tail configurations,
A = 0.0 for delta-wing aircraft, and
A = 0.2612 for canard-delta aircraft.

Aircraft with variable-geometry wings, although not catered for in the rest of the design methodology, can be dealt with by using the factors A = 3.1801, and replacing the last exponent (1.1877) with a value of 0.7569. These last factors are derived from a very small data set, and should therefore be treated with caution.

Although the equation is very simple, the results obtained are as accurate as many other more detailed sizing methods. This is because the method is only trying to capture the final size of the tail, and not predict it according to several different criteria, which it does so using only a few aircraft parameters. The results of the correlation are shown in Figure 4-6, together with the gradient of the best-fit line, and the coefficient of determination (R^2). A value of R^2 of 0.9533 is very good, and the gradient of the line, which would ideally be 1, is good enough not to cause significant errors.

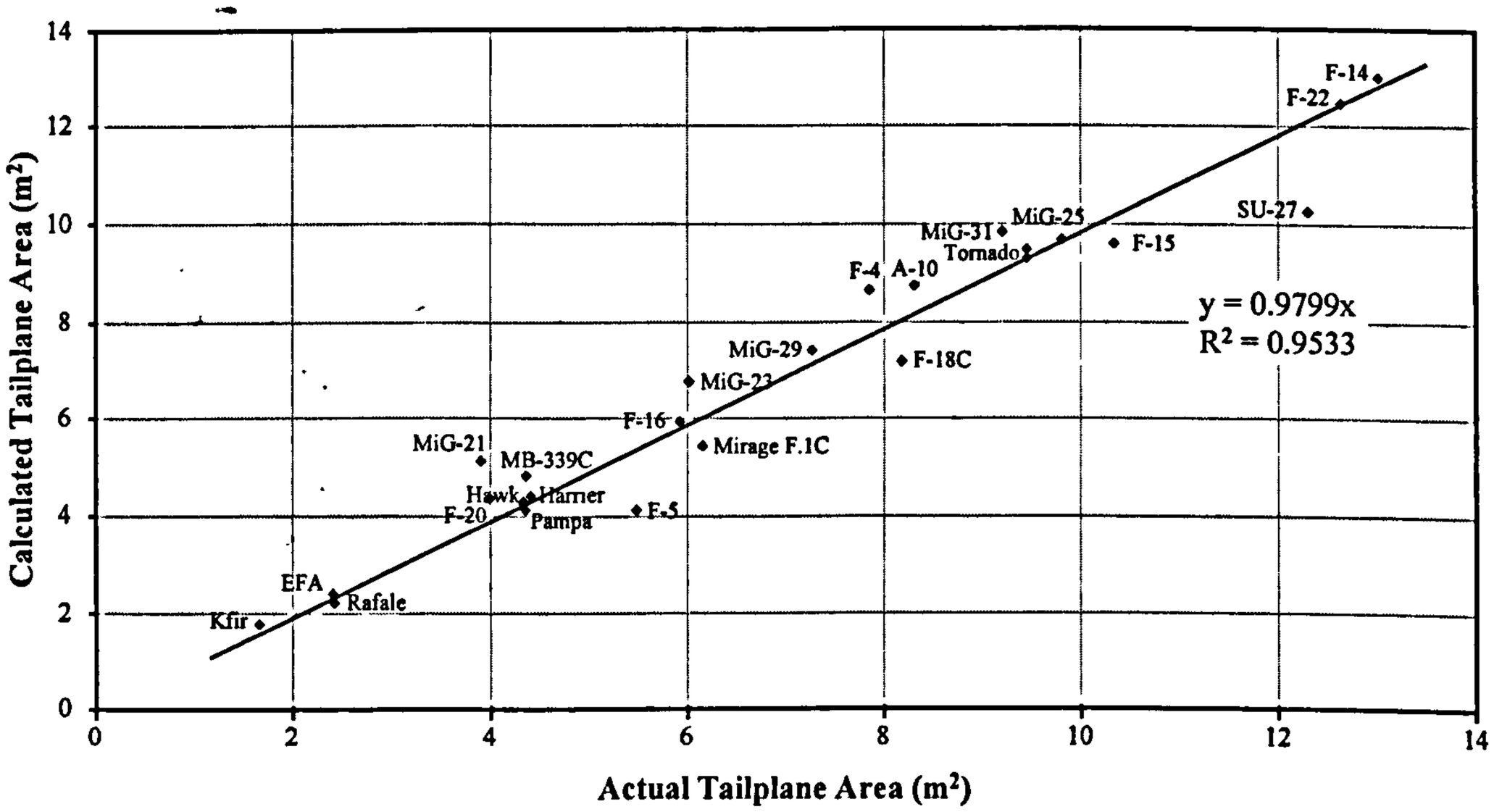


Figure 4-6. Tailplane Sizing Correlation Results.

A similar method is used to produce a sizing equation for the net vertical surface (fin) area, SEFN, and the results for this correlation exercise are found to be slightly better than the equivalent values for the tailplane area comparison. The equation makes use of the maximum fuselage length (XFN), fuselage effective diameter (DFH), and the number of fins (NEF), as set by the user in the aircraft input file.

The equation used to predict net fin area is :

$$SEFN = 0.6273 \times (0.6053 \times XFN \times DFH)^{0.6960} \times NEF^{1.0735} \quad (m^2). \qquad (4\text{-}16)$$

The results of the correlation are shown in Figure 4-7, and it can be seen that the coefficient of determination is higher than for the tailplane sizing procedure, at 0.9834, and that the gradient is closer to the optimum value of 1.0. The best-fit line gradient for the tailplane sizing correlation could be artificially raised, but this would have a slightly detrimental effect on the value of R^2.

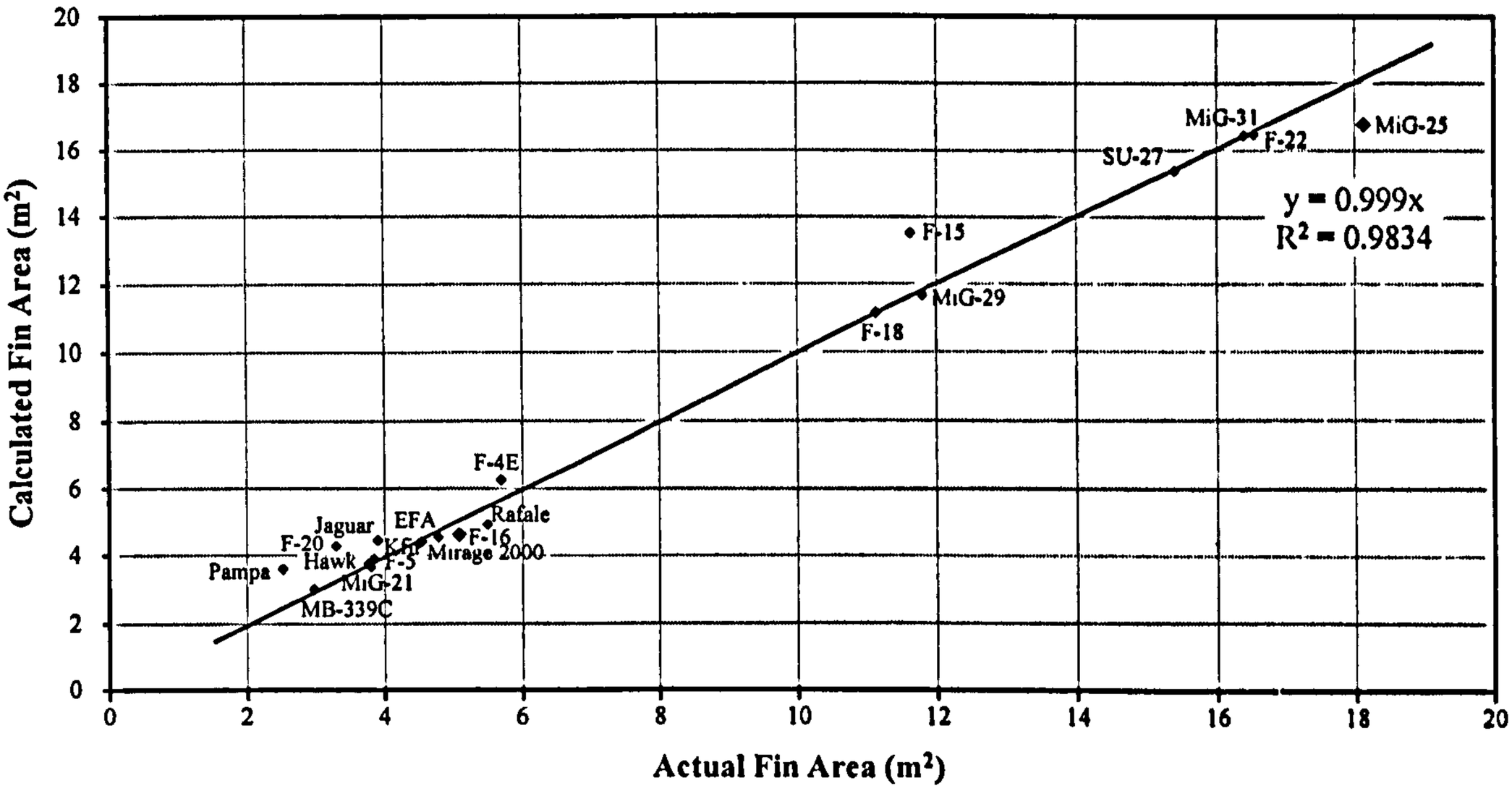


Figure 4-7. Fin Sizing Correlation Results.

For both the tail and fin, other parameters are also calculated, namely aspect ratio, thickness/chord ratio, and mean chord. Of these only the thickness/chord ratios are used, in the drag estimation module, whilst the others are provided to help the user visualise the design, and may eventually be used for graphical output

In both cases, the thickness/chord ratios, RTET and RTEF for tail and fin respectively, are set to 90% of the wing thickness/chord ratio (RTW). The tail aspect ratio, AETN, is set to 80% of that of the wing (AW). The fin aspect ratio is set to one-third of the value of the wing aspect ratio for delta and canard-delta aircraft, and to one-quarter of the wing aspect ratio for aircraft have aft-tails. All of the above values are based on observation, and on some simple rules laid down by Raymer[78].

All fin and tail parameters required by the synthesis, as well as some for the user's benefit, have now been generated. This concludes the geometrical definition of the aircraft.

4.3 Mass Estimation and Volume Accounting

One of the most important processes in the design of any aircraft is the estimation of the aircraft mass, calculated from the sum of the individual component masses, in this methodology. However, many of these components are themselves power functions of the aircraft all-up-mass, and the process becomes an iterative procedure to converge on the correct mass of the current design configuration.

The equations used have been developed over a number of years, and are presented in DAeT notes[45], which form a major part of the Cranfield MSc course in Aerospace Vehicle Design. They follow a similar method to many other mass-estimation techniques, and show surprisingly accurate results for the level of input detail required. The mass estimation equations used in COMBAT have been re-checked and modified where required; fuselage mass was updated to deal with non-circular fuselage shapes. Other changes include factors to account for composite materials, reduction in wing mass with increasing fuselage width, and other minor adjustments.

4.3.1 Structural Mass Estimation

Two parameters having a major impact on the aircraft structural mass are the design load factor, and design diving speed. Although design load factor (GND) is input by the user, the ultimate load factor (NULT) varies according to the horizontal stabiliser position (aft-tailed aircraft typically have to overcome an additional download of 0.1g), and is calculated by multiplying the corrected design load factor by 1.5. The design diving speed (VD) is a measure of the maximum dynamic pressure to be endured by the airframe, and is calculated from the low-level Mach number, MLOW.

$$NULT = 1.5 \times GND \text{ or } NULT = 1.5 \times (GND + 0.1), \text{ and} \quad (4\text{-}17)$$

$$VD = 340.29 \times MLOW \text{ (m/s TAS)}. \quad (4\text{-}18)$$

Before the mass estimation routine begins, the composite mass factor (UMATAD) is re-set to the optimiser-provided value, and the suggested empirical constants for the wing mass estimation[49] are set, as shown in Table 4-1.

	Conventional Tail Aircraft			Delta & Delta/Canard Aircraft	
	Trainers	MTG ≤ 9250	MTG > 10,000	MTG ≤ 15,000	MTG >30,000
A	0.0014	0.00095	0.00058	0.00065	0.00044
B	0.085	0.04	0.0	0.0	0.00058

Table 4-1. Empirical Wing Mass Factors.

The values of A and B thus established are used to establish the value C, which scales the wing mass, and takes account of the factor for fuselage width (BFH), as shown below. C values for canard/delta aircraft between 15000 and 30000 kg are interpolated.

$$C = 0.5\left(A - \frac{B \times MTG}{1 \times 10^6}\right)\left[1 + 0.89 \times \left(\left(1 - 5 \times \left(\frac{BFH}{BW}\right)^2\right) + \left(1 - 3 \times \left(\frac{BFH}{BW}\right)^2\right)\right) \times UW\right]. \quad (4\text{-}19)$$

COMBAT makes use of two values of aircraft mass, MTG and MTT. MTG is the maximum all-up-mass of the aircraft, and is the final iterated mass using the maximum allowable payload mass (MBMAX). MTT is the mission mass of the aircraft, and is found by adding the design mission payload to the empty mass (MTE) and fuel mass (MTGF) already found using MTG. Therefore, unless the design mission payload is equal to the maximum payload, then MTT will always be less than MTG.

Historically, when composite materials have been used in aircraft design, they were included as small structural components, and only recently have composites been used for major load-bearing structure. The mass estimation method is implemented in such a way as to mimic this procedure, and makes use of an optimiser-provided value for the fraction of empty mass made up of advanced materials. The mass fraction of the items calculated first is subtracted from the total mass fraction composed of advanced materials, which include titanium alloys, carbon composites, and more exotic materials.

The first structural component mass estimated is that for the empennage (ME), followed by the wing mass (MWC), and finally the fuselage (MFG). Each one is factored by a fraction of the remaining advanced material, until there is none left, or until the whole airframe is made of advanced materials. The maximum mass saving for each component, based on work done by Foye[35], is as follows:

Empennage: 30% Wing: 16% Fuselage: 22%

If the total advanced material fraction (UMATAD) is less than the factored mass of the next item to be calculated, a linear scaling factor is used to reduce the mass of the next item to be calculated. Thus, there will always be a continuous reduction in the mass of the component with increasing advanced material fraction, although the fraction gradient will change from one item to the next. This has been found to work well in practice, resulting in a robust solution to a complex problem. An example of this mass-reduction procedure is given below.

For conventional, aft-tail, aircraft the empennage mass is calculated as follows:

$$ME = 0.039 \times VD \times (SEFN + SETN)^{1.2} \quad (kg), \tag{4-20a}$$

while for canard-delta aircraft

$$ME = 0.0727 \times VD \times (SEFN^{1.12} + SETN^{1.12}) \quad (kg). \tag{4-20b}$$

Once the initial mass of the structure has been calculated, the minimum possible mass of the component is calculated. If the factored mass of the structure is less than the remaining mass of advanced materials, the mass is reduced by the full amount, and the remaining advanced material mass (RMATAD) is calculated:

$$ME = ME \times 0.7 \text{ , and } \quad RMATAD = RMATAD - \frac{ME}{MTE} \text{ .} \tag{4-21}$$

If the factored mass of the structural system is less than the remaining mass of advanced materials, the component mass is given a reduced factor, and the remaining advanced materials mass is set to zero:

$$ME = ME - \left(\frac{1}{0.7} - 1\right) \times RMATAD \times MTE \quad (kg), \text{ and } RMATAD = 0.0 \text{ .} \tag{4-22}$$

The same mass-reduction procedure as for the empennage is sequentially applied to the other major structural components.

The wing mass includes the empirical factor C, along with geometric and planform parameters, and the ultimate load factor, NULT, and is calculated as follows:

$$MWC = C \times \left[\frac{AW^{0.5} SW^{1.5}}{COS(QW4)} \times \frac{(1+2\,UW)}{(3+3\,UW)} \times \frac{MTG}{SW} \times NULT^{0.3} \times \left(\frac{VD}{RTW} \right)^{0.5} \right]^{0.9} \text{ (kg). (4-23)}$$

The fuselage mass has a small factor to allow for non-circular sections, but is otherwise based on the fuselage dimensions, and the design diving speed, VD:

$$MFG = 0.04 \times \left(\frac{BFH}{HFH} \right)^{0.09} \left(XFN \times (BFH + HFH) \times VD^{0.5} \right)^{1.5} \text{ (kg).} \quad (4\text{-}24)$$

4.3.2 Systems Mass Estimation

Undercarriage mass:

$$MU = 0.077\, MTG^{0.93} \text{ (kg).} \quad (4\text{-}25)$$

Propulsion system mass derivation is given fully in section 5.2.4, but the results of that method are presented here for completeness.

For engines without reheat:

$$MPG = \frac{NENG \times 17.7638 \times MPAD^{0.9739} \times NSHAFT^{0.2709}}{RPCH^{0.1385}\,(1+RPBY)^{0.4053}} \text{ (kg),} \quad (5\text{-}47a)$$

whilst there is an extra mass for the reheat system, making the total mass:

$$MPG = MPG + \frac{NENG \times 0.3628 \times MPAD^{1.9796}}{\left(\frac{TPGH}{TPGD} \right)^{0.0021} \times (1+RPBY)^{0.8468} \times RPCH^{0.5264}} \text{ (kg).} \quad (5\text{-}47b)$$

Where:

NENG = number of engines
MPAD = SLS air mass flow rate (kg/s)
NSHAFT = Number of main engine shafts/spools
RPCH = SLS compressor pressure ratio at maximum power
RPBY = SLS engine bypass ratio at maximum power
TPGH = uninstalled SLS engine maximum thrust (with reheat) (N)
TPGD = uninstalled SLS engine military thrust (max. dry thrust) (N)

The fuel system mass is based on the maximum all-up mass and the total internal fuel volume, VTGFI, which is calculated from the internal fuel mass, MTGFI:

$$MSF = 0.019 \times MTG + 5 + 12 \times \frac{MTGFI}{779} \text{ (kg).} \quad (4\text{-}26)$$

The mass of the flight control system depends on the aircraft all-up mass, and whether the intake system has moving ramps. For non-moving ramps (fixed geometry intakes):

$$MSC = 0.11 \times MTG^{0.8} \quad \text{(kg), or} \tag{4-27a}$$

$$MSC = 0.115 \times MTG^{0.8} \quad \text{(kg) for variable-geometry intakes.} \tag{4-27b}$$

The hydraulic and pneumatic system mass is based only on the aircraft all-up-mass.

$$MSH = 1.6 \times MTG^{0.5} \quad \text{(kg).} \tag{4-28}$$

The mass of the electrical system includes generation, switching, and batteries, and is dependent on the role of the aircraft. For Trainers:

$$MSE = 0.15 \times MTG^{0.84} \quad \text{(kg), whilst for other aircraft} \tag{4-29a}$$

$$MSE = 0.06 \times MTG^{0.84} \quad \text{(kg).} \tag{4-29b}$$

The air system mass includes the environmental control and anti-icing systems, and is based on the number of crew, NCP, and the all-up-mass:

$$MSA = 0.006 \times MTG \times NCP^{0.5} + 0.08 \times MTG^{0.7} \quad \text{(kg).} \tag{4-30}$$

The 'cockpit systems' group contains the mass estimates for the cockpit instruments, controls, oxygen system, fire extinguishers, furnishings, and external paint.

$$MCI = \left[3.6 + 21.4\ NENG^{0.676}\right] + \left[12\left(1 + NCP^{0.263}\right)^{1.356}\right] + \left[0.006\ MTG\right] + \left[4.8\ NENG^{1.008}\ (2\ XFN)^{0.222}\right] + \left[98.5\ NCP\right] + \left[0.5\ SW\right] \quad \text{(kg).} \tag{4-31}$$

If the avionics mass is not specified in the aircraft input file, the program makes an estimate of the mass, based on a very simple equation to estimate the installed avionics mass:

$$MAX = 0.0162\ MTG + 279 \quad \text{(kg).} \tag{4-32}$$

Combat aircraft will have a certain mass of fixed armament items, based on the amount of payload that can be carried. Although pylons, etc. can be removed, the airframe will still be strengthened for the extra loads:

$$MWCX = 0.1 \times MBMAX \quad \text{(kg).} \tag{4-33}$$

Finally, an estimate of the auxiliary power unit mass (MPA) is made for aircraft specified as having an APU in the aircraft input file:

$$MAPU = 0.1 \times MPG \quad \text{(kg).} \tag{4-34}$$

The masses of all of the separate aircraft structures and systems have now been calculated, and they are simply added, with the mass of the gun, if allocated, to obtain the aircraft empty mass, MTE:

$$MTE = MWC + MFG + ME + MU + MPG + MPI + MSF + MSC + MSH + MSE + MSA + MCI + MAX + MWCX + MPU + MGCA \quad \text{(kg).} \tag{4-35}$$

The crew mass is estimated as 100 kg per crew member:

$$MCP = NCP \times 100 \quad (kg). \tag{4-36}$$

The internal fuel mass, MTGFI, is calculated using the fuel fraction, FMFMT, which is an input from RQPMIN:

$$MTGFI = FMFMT \times MTG \quad (kg). \tag{4-37}$$

The maximum all-up mass, MTG, is calculated using the maximum payload mass, MBMAX:

$$MTG = MTE + MBMAX + MTGFI + MCP \quad (kg). \tag{4-38}$$

The mass estimation is iterative, as stated above, and the initial gross mass is re-calculated until the initial and calculated mass agree to within 0.1 kg. This limit may appear small, but larger values can lead to discontinuities in the optimisation functions, reducing the robustness of the code.

The mission mass is calculated from the mission payload, which is made up of the mission payload (MB), internal fuel (MTGFI), external fuel (MTGFX), the empty mass (MTE) and the crew mass (MCP).

$$MTT = MTE + MCP + MTGFI + MTGFXI + MB \quad (kg). \tag{4-39}$$

It is this mass which is used for the calculation of all aircraft masses for the mission performance calculations.

4.3.3 Mass Estimation Results

Although the above methods are simple, Figure 4-8 shows that surprisingly accurate results are achieved when the aircraft is treated as a whole system. It should be noted that the calculated empty mass is the result of only the first iteration, and not the final iterated result.

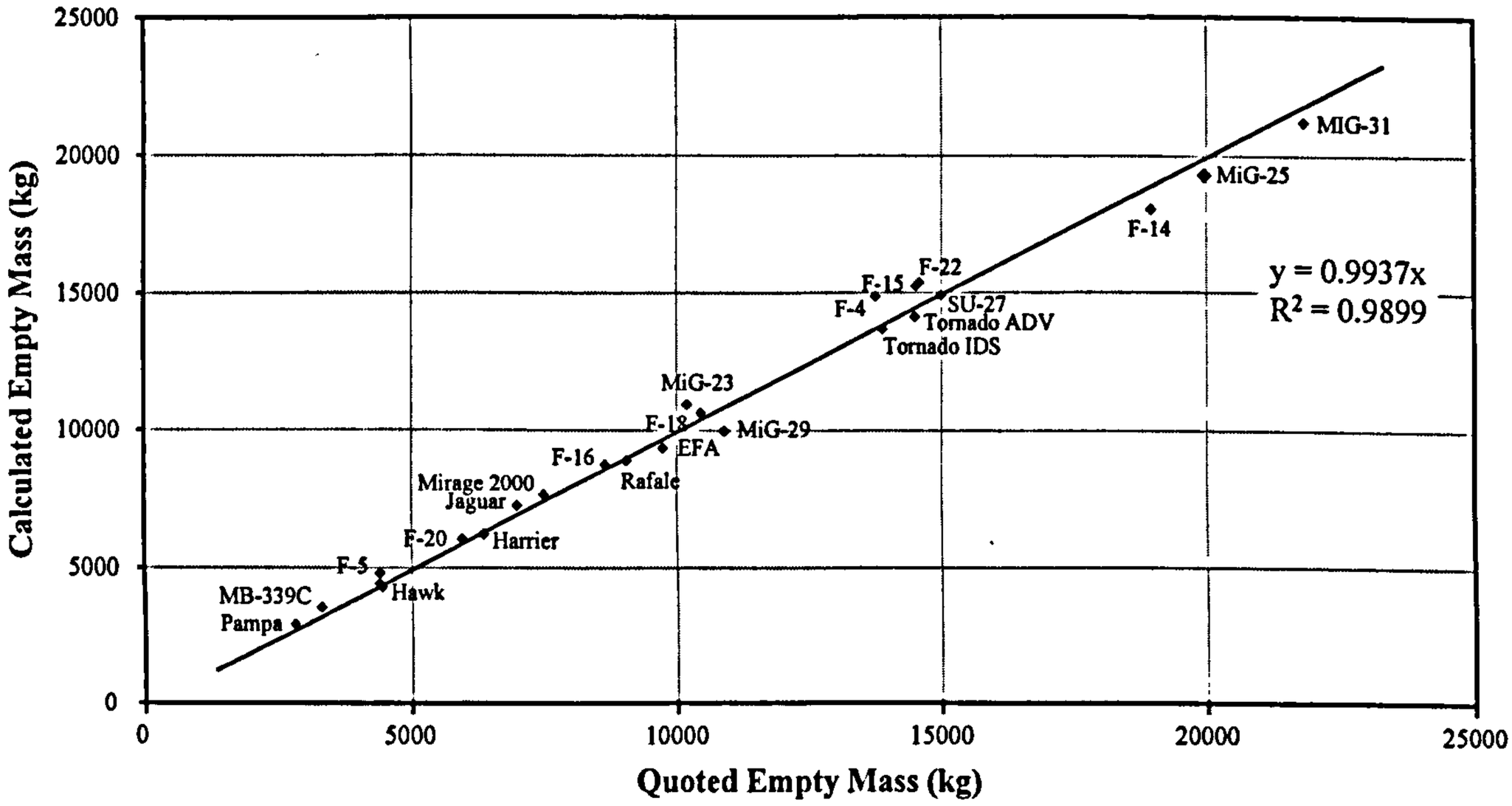


Figure 4-8. Mass Estimation Correlation.

It would obviously be better to validate individual components for a given all-up mass, but lack of published data and discrepancies between mass components and their sum to form all-up mass prevents direct comparison. Under such circumstances, the methods used would appear to be the only feasible solution, even though they may show the mass estimation method in a favourable light. Even with this caveat, the results are very good, with a best-fit line having a gradient of 0.9937, and a coefficient of determination of 0.9899.

4.3.4 Volume accounting

In order to ensure that the aircraft contains enough volume to enclose the structure, systems, crew, and all internal fuel, some kind of volume check is required. COMBAT uses a relatively simple method to estimate the total fuel volume available in the wing, and adds it to the available fuselage fuel volume. This last volume is calculated by estimating the total fuselage volume using simple geometric shapes, and then subtracting the volume of all of the individual systems, based either on known geometry, or on standardised system densities.

There are no discrete internal fuselage fuel tanks as such, but the fuel is allowed to occupy a fraction of the remaining fuselage volume. This results in a robust and fairly accurate method of ensuring adequate storage volume for internal fuel. Figure 4-9 gives a generalised view of the fuel storage system within the aircraft.

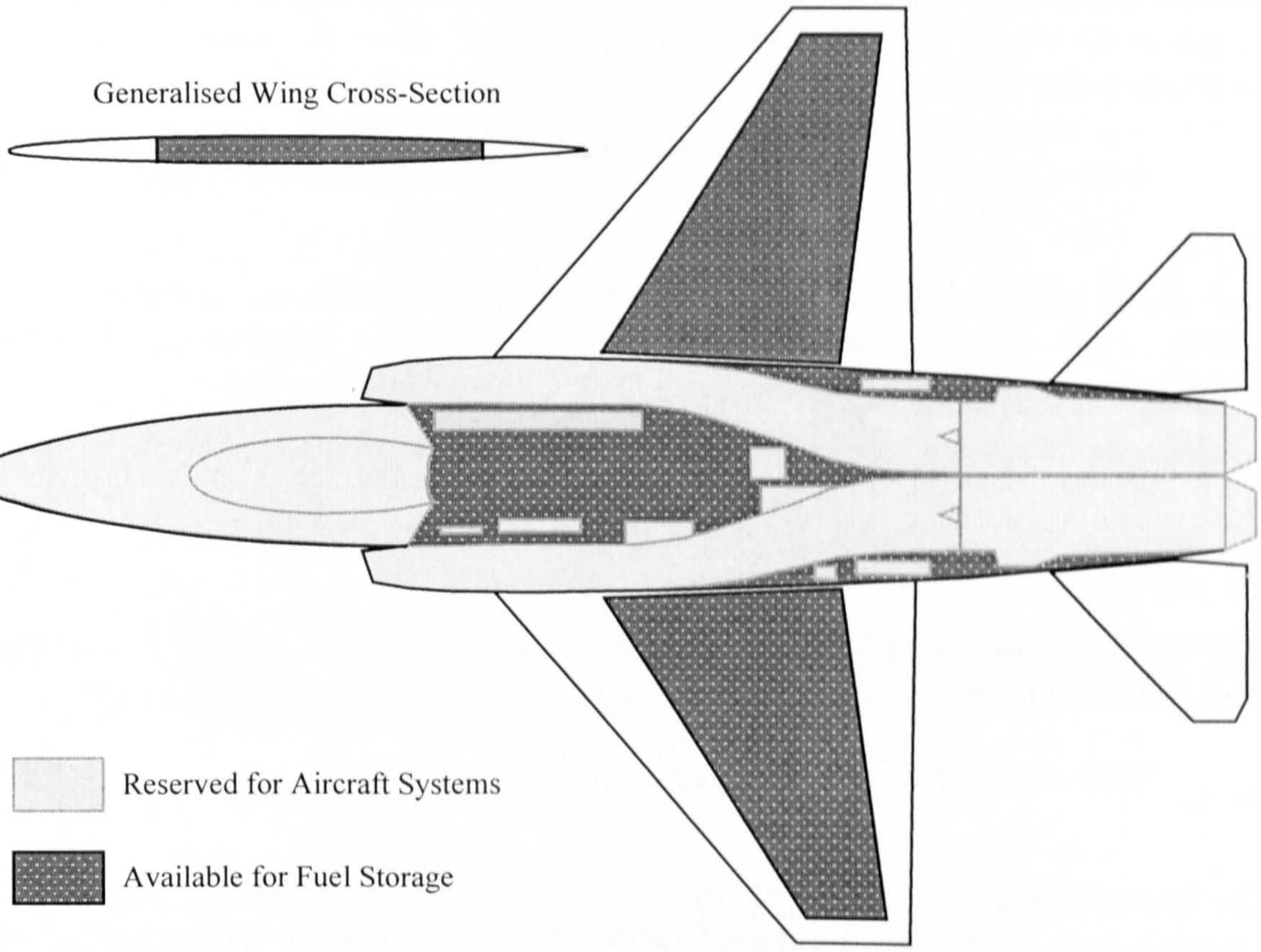


Figure 4-9. Aircraft Internal Fuel Storage Volumes.

The total volume available for fuel storage in the wing (VWFG) is calculated using the following assumptions:

- Fuel is stored in an integral tank between the front and rear main wing spars, limiting the storage ratio across the chord to approximately 50%.
- The tanks begin at the outer edge of the fuselage and continue almost to the wing tip.
- To allow for structure, fuel expansion, and the small volume lost because not all of the wing span is used, a factor of 80% is applied to the maximum available volume.

Using the equation for the volume of an obelisk, and converting into known parameters:

$$VWFG = \left(\frac{BW - BFH}{3}\right) \times \left[0.4 \times RTW \times CWCB^2\left(1 + UW + UW^2\right)\right] \ (m^3). \quad (4\text{-}40)$$

To estimate the total fuselage volume, the fuselage is split into two sections, the nose section, and the main body. The nose section volume, VFI, is treated as a tangential ogive with a base area equal to the maximum fuselage cross-sectional area less the area of the engine intakes, and of length 32.5% of the maximum fuselage length:

$$VFI = \frac{8}{15} \times 0.325 \times XFN \times \left(\frac{\pi \times DFH^2}{4} - OII\right) \ (m^3). \quad (4\text{-}41)$$

The main fuselage body volume is calculated from the average fuselage effective diameter, assuming that the effective volume is 85% of the maximum possible, and the length is 67.5% of the maximum fuselage length. Thus, the total fuselage volume (VFG) is found to be:

$$VFG = VFI + 0.85 \times \pi \times \frac{\left(0.5 \times (DFH + DFG)\right)^2}{4} \times 0.675 \times XFN \ (m^3). \quad (4\text{-}42)$$

One of the largest volumes that must be subtracted is the volume of the intake ducts, VIDG. Their volume is calculated from the average duct diameter and their length, which is found from the aft fuselage length and the total engine length:

$$VIDG = NENG \times \left[0.5 \times \left(\frac{OII}{NENG} + \frac{\pi \times DP1^2}{4}\right)\right] \times (0.675 \times XFN - LPG) \ (m^3). \quad (4\text{-}43)$$

The volume of the engine bay (VPB) is found by treating the engines as cylinders of diameter equal to the maximum engine diameter along the entire engine length:

$$VPB = NENG \times \frac{\pi \times DPH^2}{4} \times LPG \ (m^3). \quad (4\text{-}44)$$

The fuselage structure volume (VFW) is calculated from a standardised structure density obtained from Smith and Burnham[94], with a factor to allow for the percentage of the fuselage volume made from advanced materials, FMFG:

$$VFW = \frac{MFG}{2403 \times FMFG} \ (m^3). \quad (4\text{-}45)$$

All other system volumes are calculated similarly, the system densities being as follows:

System	Density (kg/m^3)	System	Density (kg/m^3)
Undercarriage Bays	1922	Air Services	480
Fuel System	480	Gun and Ammunition	800
Power & Flight Controls	600	Avionics	560
Hydraulics & Pneumatics	480	APU	600
Electrical	480		

Table 4-2. Aircraft System Densities

The density for the avionics system had to be averaged from the values in Smith & Burnham[94], as COMBAT does not treat the avionics and radar components as different systems, and assumes that the radar and antennae account for one quarter of the total avionics mass.

The volume required for the crew (VCI) is calculated separately, with a slight reduction in volume for a crew of two:

$$VCI = 1.69 \times NCP^{0.8} \quad (m^3). \tag{4-46}$$

Now that all the relevant system volumes have been calculated, they are subtracted from the total fuselage volume calculated using Equation 4-42, allowing a maximum of 50% of the total remaining space for fuel storage:

$$VTGFI = VWFG + 0.5\begin{pmatrix} VFG - VFW - VIDG - VUT - VBI - VPB - VGCA - \\ VSF - VSC - VSH - VSE - VSA - VCI - VAX - VPU \end{pmatrix} \quad (m^3). \tag{4-47}$$

It should be noted that the volume set aside for internal weapons carriage (VBI), which is a value specified in the aircraft input file, has been included in the above calculation.

The final volume calculation is to enable RQPMIN to check that the internal fuel volume (VTGFI) is adequate. This is done most efficiently by checking that the required mission fuel volume (VTGF) is less than the sum of the total internal volume and the volume of external fuel. Reversing the equation to correspond with the form required for an inequality constraint:

$$FUNC(7) = VTGF - VTGFI - \frac{MTGFX}{779} \quad (m^3). \tag{4-48}$$

4.4 Summary

Chapter 4 has:

- Outlined the operation of the whole synthesis code, including the interrelation of the aircraft synthesis, optimiser, and life cycle cost modules.
- Presented the method of aircraft geometry description, and given relevant algorithms as used in the synthesis code.
- Detailed the mass estimation equations and procedures, and given a correlation of the results for 24 combat aircraft, the results of which are very accurate for a conceptual design methodology.
- Described the method used to predict aircraft volume and fuel storage, and the equations used to ensure that this is met during the aircraft design procedure.

5 Aerodynamics and Propulsion

Each of the two main parts of this chapter describes the respective model; aerodynamics and propulsion. The level of detail given in parts is not as high as for the previous chapter, as two of the major models described below were already publicly available, and are well documented. Readers requiring greater detail are directed towards the references listed in this chapter for a full description of the methods and operation of the models outlined below.

The three aerodynamics models used are responsible for predicting the available lift, the angle of attack - based on the lift-curve slope-, and the drag during cruising and manoeuvring flight. They are all based on empirical, or semi-empirical methods, and are well matched to the level of detail generated by the geometry definition modules of previous chapters. The maximum (or available) lift model is based on a set of equations developed at Cranfield, and seems to give excellent results from simple algorithms. The angle-of-attack module is slightly more complex, having factors to account for the wing configuration, advanced sections, and a simple factor for the effect of vortex lift. The drag module is the most complex of the three aerodynamic models, making use of a large number of data tables that are interpolated to give a drag build-up approach.

The propulsion model is split into three major parts; thermodynamic design, thermodynamic analysis, and engine dimension and mass estimation. The first two sections are described, but the equations for the thermodynamic modelling and analysis are not given because of the complexity of those models, and the excellent references available. The engine dimension and mass estimation models were developed during the course of this study, and provide a level of accuracy not previously available at the conceptual design stage. They make use of parametric equations to predict inlet (fan) diameter, maximum diameter, bare length, maximum length, and engine dry mass. Results from each of the above calculations are given, together with an example of the installed performance of a modern military turbofan engine.

5.1 Aerodynamic Modelling

As stated previously, the aerodynamics module consists of three models. The first predicts available lift coefficient based on wing configuration and geometry, Mach number, and the presence of high-lift devices. The second calculates the angle of attack from the lift-curve slope, which is based on the clean wing geometry, and the flight Mach number, and contains a simple correction for the effects of vortex lift. The third model is the largest and most complex of the three, and calculates the drag of the aircraft based on its geometry, lift coefficient, configuration, and the presence of external stores and retractable components.

The three sections are not connected, except for the fact that they use many of the same parameters and are involved in the estimation of aerodynamic values. They are separate subroutines in the synthesis code, and operate independently of one another. In this way, any one of the routines could theoretically be replaced, and the overall synthesis should still work, provided the replacement subroutine is capable of passing all of the relevant variables.

5.1.1 Lift Model

This model is based on a method proposed by Howe[48], which, despite its simplicity, seems to give remarkably accurate results. The method requires the wing geometry parameters and aircraft configuration, and predicts the take-off, landing, high-speed (manoeuvre), and loiter (buffet limit) lift coefficients.

The method is applicable to all low-aspect ratio configurations, including variations such as cambered leading edge deltas, canard-deltas, and swept/delta wings with leading-edge extensions (LEX). The choice of the wing and tail configuration, specified during the aircraft definition phase, has a major influence on the lift factors applied to the basic model.

Take-off lift coefficient is based on the wing aspect ratio (AW), quarter-chord sweep (QW4), and a number of configuration-dependent factors, all of which are given in Table 5-1. For aft-tail aircraft:

$$CLTO = \left(\sqrt{\frac{AW}{2}} + DELTLE\right) \times \frac{\cos(QW4)}{1.15^2} + DELTET \quad \text{, and} \tag{5-1a}$$

$$CLTO = \frac{\cos(QW4) + DELTL}{1.15^2} \quad \text{for delta and canard-delta configurations.} \tag{5-1b}$$

Landing lift coefficient is calculated similarly, where, for aft-tail aircraft:

$$CLLA = \left(\sqrt{\frac{AW}{2}} + DELTLE\right) \times \cos(QW4) + DELTEL \quad \text{, and} \tag{5-2a}$$

$$CLLA = \cos(QW4) + DELTL \quad \text{for delta and canard-delta configurations.} \tag{5-2b}$$

Trailing-Edge Device Lift Contribution - Conventional Configurations	Take-Off DELTET	Landing DELTEL
Single-slotted flap	0.24	0.54
Double-slotted or Fowler flap	0.29	0.66
Configuration Increments	**Low-Speed DELTL**	**High-Speed DELTH**
Aft-tail		0.6
Delta	0.3	0.6
Aft-tail with LEX and leading-edge devices		0.8
Canard-Delta	0.4	0.8
Delta with leading-edge device	0.5	0.8
Canard-Delta with leading-edge device	0.6	1.0

Table 5-1. High-Lift Device and Configuration Lift Coefficient Increments.

Aircraft having aft-tail configurations have the extra factor DELTLE, which is zero for aircraft with plain leading edges, and 0.25 for those having leading-edge devices.

Cruise lift coefficient is the maximum lift that the aircraft should fly at for a cruise, loiter or CAP phases, and is normally the C_L boundary of aerodynamic buffet. For aft-tail aircraft:

$$CLLO = 0.28 \times \cos(QW4) \quad \text{, and} \tag{5-3a}$$

$$CLLO = 0.4 \times CLLA \quad \text{for delta and canard-delta configurations.} \tag{5-3b}$$

Maximum lift coefficient is a very difficult parameter to estimate during high-speed or manoeuvring flight. This is because of the highly non-linear lift conditions that occur, and the poor correlation between wind-tunnel and flight-test performance data at the extremes of the aerodynamic envelope. The current model builds on Howe's method, and was developed for COMBAT using attained turn data for the F-16C published in Jane's All the World's Aircraft 1995. The model contains additional factors to account for Mach number and the loss of total available lift at high load factors, due to the effect of the aircraft trim. This latter factor is included in the attained turn equations described at section 6.1.3, for ease of implementation.

For all configurations except conventional aft-tail aircraft without LEX, the calculation of the maximum available manoeuvre lift coefficient (CLHS) follows the following procedure. For most configurations, CLHS drops linearly from its theoretical maximum at M = 0, to level off at 85% of the maximum value from Mach numbers from M = 0.6 up to M = 0.95, where shock formation will start to dominate the aerodynamic characteristics of the aircraft. For conventional aft-tail configurations without LEX, CLHS drops from the theoretical maximum to level off at 80% of the maximum between M = 0.8 and M = 1.0.

The value drops off sharply once the wing becomes mostly supersonic, so that:

$$DELTCL = -0.3448 \times MACH^3 + 1.9805 \times MACH^2 - 3.8288 \times MACH + 2.9939 \ , \tag{5-4}$$

where DELTCL is the actual lift coefficient divided by the theoretical maximum, i.e. the value by which the maximum is multiplied to get the real value of the lift coefficient. Thus, the manoeuvre lift coefficient is:

$$CLHS = \left(\sqrt{\frac{AW}{2}} \times \cos(QW4) + DELTH \right) \times DELTCL \ . \tag{5-5}$$

There is a final reduction in the available C_L when the wing leading-edge becomes sonic, i.e. the shock wave generated from the wing root lies along the leading edge. In reality, the effect occurs when the shock is just ahead of the wing, probably caused by the increased local flow velocity due to acceleration around the leading edge. An estimate is made of the Mach number (MACHLE) for which the Mach angle would be equal to the leading edge sweep (QWL):

$$MACHLE = \frac{1}{\cos(QWL)} \times \cos(ALPHA) - 2 \times RTW \ . \tag{5-6}$$

If the flight Mach number is greater than MACHLE, the whole wing is in supersonic flow, and DELTCL should follow Prandtl's theoretical calculation:

$$\beta = \frac{1}{\sqrt{M^2 - 1}} \ .$$

In reality, the lift coefficient cannot reach infinity at M = 1, so a six-order polynomial approximation to this was found, using data from the F-16C:

$$CLHS = CLHS \times DELTCL \times \begin{pmatrix} 7.3272 \times DELTM^6 - 28.387 \times DELTM^5 + \\ 42.277 \times DELTM^4 - 30.313 \times DELTM^3 + \\ 10.774 \times DELTM^2 - 2.1764 \times DELTM + 1 \end{pmatrix} \ . \tag{5-7}$$

As MACHLE relies on the value of the angle of attack, ALPHA, the method is iterative, but converges quickly to return an estimation of the true lift coefficient available at a given Mach number for a known geometry and configuration.

A graphical representation of the model is shown in Figure 5-1. The two lines show the values predicted by the model, and the other data points are the aircraft lift coefficient values calculated from published F-16C data for increasing load factor. It should be noted that increasing g results in lower available lift coefficient, because of the increasing effect of the tail trim, as previously stated. The low values of available C_L for g = 9 at subsonic speeds, and for g = 1 at supersonic speeds are probably due to a limit being applied by the flight control system at the bounds of the flight envelope. No attempt has been made to include these limits, as they will almost certainly be aircraft-specific.

Due to the military sensitivity towards such information, lift coefficient data is very rarely published for combat aircraft, as it can be used to predict aircraft performance. This prevents checks against actual lift coefficients, but from the results of previous comparisons with real aircraft, the accuracy seems very good. Obviously, small changes in aircraft dimensions can have quite a large effect on the aerodynamic performance, and this should be considered when modelling or comparing with data for in-service aircraft. The general trends should be well defined, though, making the model a very significant contributor to the overall capability of the aircraft design synthesis.

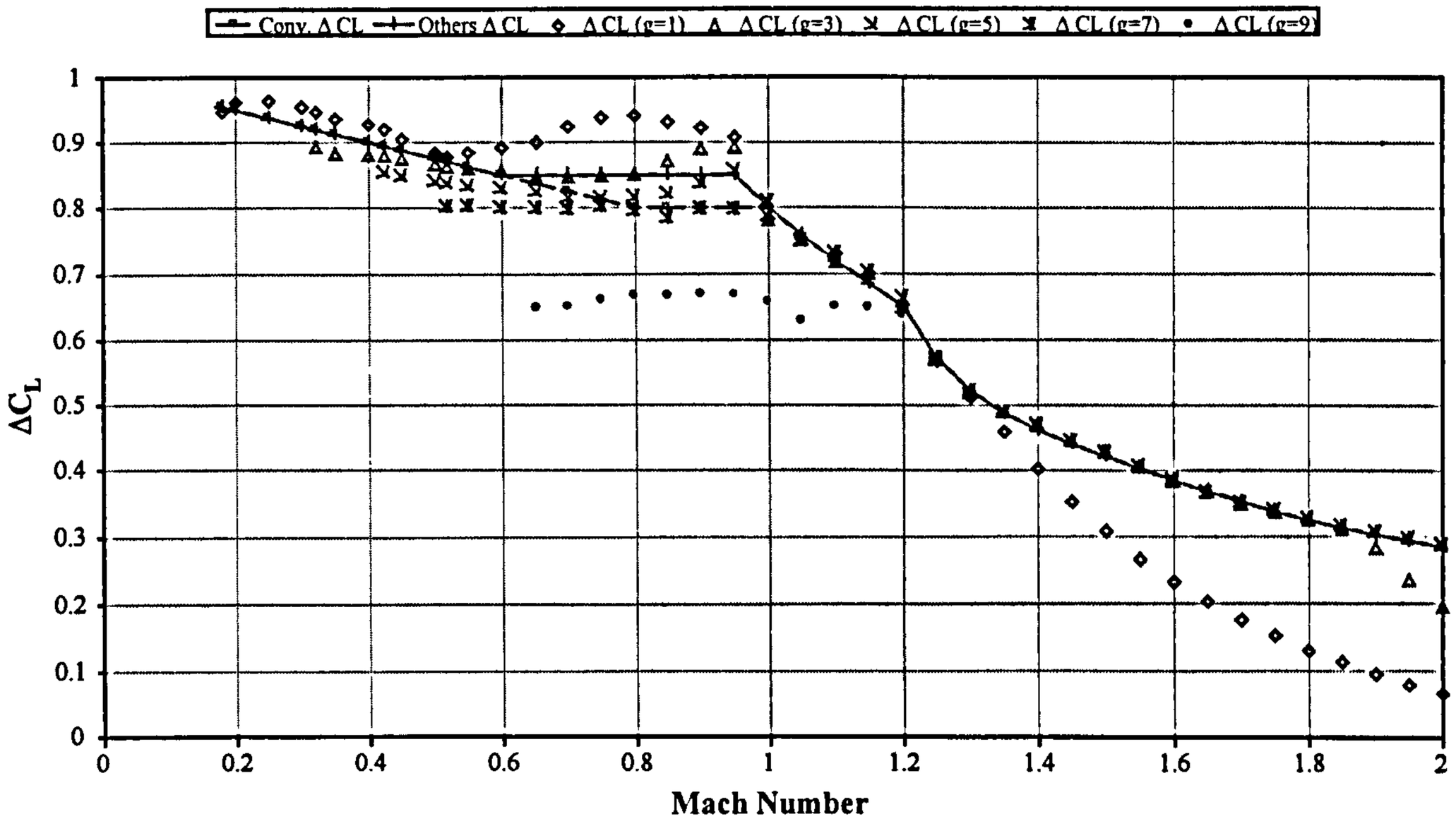


Figure 5-1. Example Lift Correction with Mach Number.

As stated above, actual lift coefficient data is rarely published, making model validation extremely difficult. The calibration of the individual lift models is impossible with the data available, but the comparison of the lift coefficient values when used in other performance models has proven slightly easier. The reader is directed to section 6.1, which details the aircraft point performance calculation methodologies. Several of the coefficients estimated using the above equations are utilised in the performance equations of that chapter, and the comparison of results can be seen to be very good. Since it is the overall aircraft performance that is the main concern of a study such as this, the accuracy of the individual models, whilst important, is of lesser concern than the prediction ability of the synthesis as a whole.

5.1.2 Angle-of-Attack Model

The angle-of-attack model is based on the work of Schemensky[84], and predicts the angle of attack (AoA) based on the Mach number and the lift-curve slope, which is a function of the aircraft geometry. The value of the lift-curve slope is predicted using several rather involved semi-empirical equations that result in a continuous expression for the subsonic, transonic, and supersonic regions. The model is also able to predict lift-curve slope values for aircraft having supercritical aerofoils. A modification has been made from the original method to take account of the extra lift incurred by highly swept delta wings and conventional wings with leading-edge extensions (LEX).

The value of the lift-curve slope is given by:

$$CLA = CLAB \times KT \times KB \tag{5-8}$$

where CLAB is the wing-alone lift-curve slope with no thickness effects. The factors KT and KB account for the effect of aerofoil thickness and fuselage interference, respectively. The method is complicated, and as the Schemensky report is very thorough, only the major equations are given below.

The basic wing lift-curve slope (CLAB) is calculated from a modified form of the equation for the lift-curve slope for trapezoidal wings, with corrections for use with non-trapezoidal shapes:

$$CLAB = \frac{1}{\frac{\left(\frac{1}{RDN}\right)\left(\frac{M^*}{M}\right)^z}{CLA0} + \frac{BETAP}{4}} \quad \text{for } M > M^*, \text{ and} \tag{5-9}$$

$$CLAB = \frac{\pi \times RDN \times TWOD \times AW}{\left(TWOD + \sqrt{TWOD + \left(1 - COSZ^{4/3}\right)\left(\frac{M}{M^*}\right)^{8/3} \times \left(\frac{AW}{2 \times COSZ}\right)^2}\right)} \quad M<M^* \ . \tag{5-10}$$

Where: $RDN = \frac{\pi}{180}$,

$$M^* = \frac{10 + 0.91 \times AW^3}{10 + AW^3} + \left(1 - \frac{10 + 0.91 \times AW^3}{10 + AW^3}\right) \times (1 - COSZ)^2 \ , \tag{5-11}$$

COSZ = cos(QW2) , where QW2 is the effective mid-chord sweep, CLA0 is defined as the wing-alone lift-curve slope at $M = M^*$, in radians,

$$BETAP = (M - M^*) \times \left[1 + \left(\frac{M^*}{M}\right)^Y\right]^2 \ , \text{ and}$$

$$Y = \frac{1 + \pi \times AW}{3 + \pi \times AW} \times \left(2 + \frac{2}{3} \times \sqrt{UW - UW^2}\right) \ .$$

The two-dimensional lift curve slope (TWOD) is normally set to 1.0, but is altered slightly for supercritical aerofoils.

For thin wings, experimental values of the lift-curve slope normally reach a peak at speeds somewhat lower than M^*. With low sweep and moderate-to-high aspect ratio, the peak occurs below M = 1, whilst with high sweep and/or low aspect ratio, the peak usually occurs at, or above, sonic speeds. At speeds well above sonic, the lift-curve slope decreases with increasing Mach number, and when the leading edge becomes sonic, the level approaches the two-dimensional theoretical value:

$$CLAB = \frac{4\pi}{180 \times \sqrt{M^2 - 1}} \qquad (5\text{-}12)$$

Wings having thick aerofoils undergo a degradation in lift-curve slope beginning at $M > M^*$, and lift-curve slope dips to reach a minimum below $M < 1$, and then recovering to a second peak at $M > 1$. To account for this phenomenon, the thickness correction factor, KT, is used. The calculations for KT are lengthy and complex, and are not detailed here, but Figure 5-2 gives a good representation of the phenomena explained above.

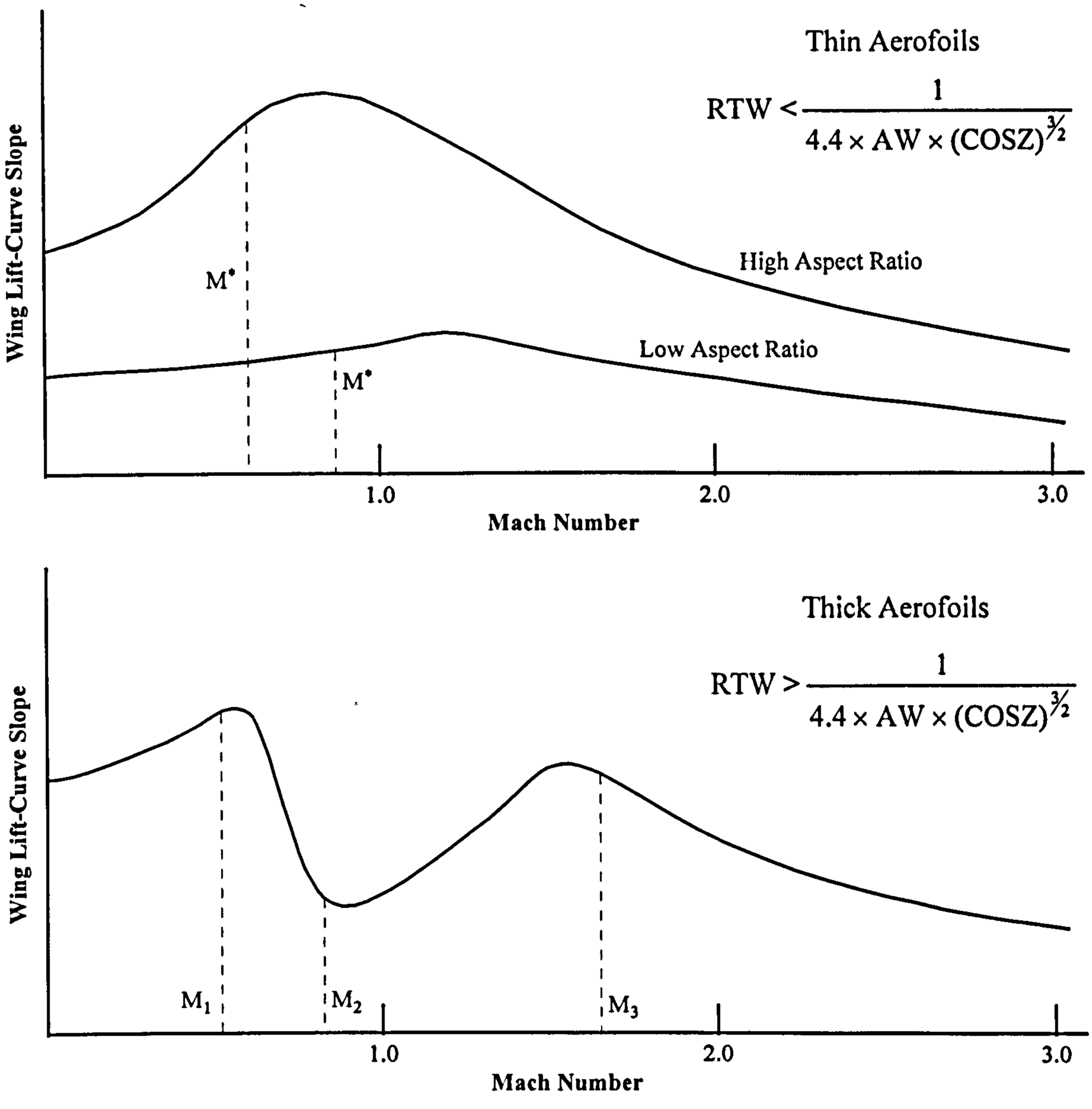


Figure 5-2. Typical Lift-Curve Slopes.

Another factor in the lift-curve slope prediction equation is the fuselage interference factor (KB), defined as follows:

$$KB = \left(1 + \frac{BFH}{BW}\right) \times \left(1 - \frac{BFH}{BW}\right)^{F} \tag{5-13}$$

Where:

$F = \frac{16 + 3 \times AW^2}{8 + 5 \times AW^2}$, and BFH and BW are the fuselage maximum width and the wing span, respectively.

The factor KB accounts for the change in the wing lift due to the part of the body that encloses the wing at that point, and due to the wing-induced lift on that part of the fuselage. It can be seen that (to a first order) KB is independent of Mach number.

The final factor in the estimation of the wing lift-curve slope is the vortex-lift factor, KV. This is an attempt to include the extra lift produced by highly-swept wings, which results in a slight increase in the lift-curve slope, and a corresponding reduction in the angle of attack for a given C_L. The vortex lift factor, KV, is estimated as follows:

$$KV = 2.075 \times \sin(QWL) \times \sin^2(ALPHA) \times \cos(ALPHA) \tag{5-14}$$

Where:

QWL is the leading-edge sweep, and ALPHA is the aircraft angle of attack.

The true angle of attack is then estimated from the values calculated previously:

$$CLCALC = CLA \times \cos(ALPHA) \times \sin(ALPHA) + KV \tag{5-15}$$

Because the vortex lift correction is dependent on the angle-of-attack, the model iterates until the estimated and resulting angle-of-attack converge. At this point, the calculation of the angle of attack is complete, and the value is returned to the aircraft performance modules.

The value of the lift-curve slope calculated above is for the wing only, as this was felt to be sufficiently accurate for the requirements of the COMBAT aircraft design and optimisation tool. The Schemensky model does include the calculation of tail lift-curve slope, which would ordinarily be added to the basic wing value calculated above, to produce the total aircraft lift-curve slope. However, the Schemensky model does not make it clear how the presence of a canard would be modelled, and since a correction for aircraft trim is included in the aircraft performance algorithms, it was felt that a more consistent set of calculations could be achieved using the wing-only lift curve slope.

5.1.3 Drag Model

The DELTA Drag method was developed by Feagin and Morrison[34], of the Lockheed-California Company, to enable the simple and rapid estimation of aerodynamic drag characteristics of future aircraft designs. The model uses an empirical total drag technique developed from data for 19 supersonic and subsonic military aircraft, and 15 advanced or supercritical aerofoil concepts. It was coded at Cranfield and updated over a period of time to include estimates for flap, undercarriage, stores, and external fuel drag. It was modified during the course of this study to act as a subroutine for the main aircraft synthesis model, and corrections and improvements were made to several of the estimation algorithms.

The method can be used to estimate the total configuration drag polar within a speed range of $0.1 < M < 2$ (approx.) and a C_L range of 0.0 to about 0.60, or buffet onset. For lift coefficients above this value, results are extrapolated from the last known data points, with a corresponding drop in accuracy. Results obtained using the method to predict known aircraft characteristics are good, and the degree of accuracy is sufficient for establishing the trends and function gradients required by the COMBAT design synthesis.

To provide the basic data for this study, lift and drag data, buffet boundary, and geometric configurations, information was supplied by NASA Ames on 18 subject aircraft. These aircraft were the T-2B, T-37B, KA-3B, A-4F, TA-4F, RA-5C, A-6A, A-7A, F-4E, F-5A, F-8C, F-11E, F-100, F-101, F-104G, F-105B, F-106A, and XB-70. Data for the S-3A and 15 concepts were supplied by Lockheed.

The data on the subject aircraft included:

- Wing geometry (aspect ratio, sweep, thickness, and aerofoil section)
- Cross-sectional area distribution (area progression curves)
- Drag coefficient variation with lift coefficient and Mach number (polars)
- Lift coefficient buffet limits versus Mach number

Items correlated for each aircraft include:

- Design lift coefficient
- Design Mach number
- Drag divergence Mach number
- Compressibility drag versus Mach number
- Pressure drag versus Mach number
- Lift coefficient for buffet onset versus Mach number

DELTA is most applicable near the cruise condition, although modifications have been made to the original method to enable turning performance to be modelled, with a small decrease in accuracy. The technique is particularly suited to preliminary design procedures, where design details are minimal and where it is usually desirable to investigate the trade-off of such design parameters as sweep, aspect ratio, thickness, or body geometry, on performance. The following sections present the final method resulting from this empirical drag correlation technique, which also make up the aircraft drag prediction code.

The method relies on a series of curves that are stored in a tabular format in a large data file. These tables will be referred to throughout the following text, but the values contained in them are not essential for the understanding of the methodology. The code uses the data from the tables with a conic-cubic spline procedure, to interpolate between the specific data points, obtaining a smooth representation of the original data. This avoids any step changes in the drag polars, which is a requirement for RQPMIN. All data required for the model is provided from the aircraft geometry and sizing modules.

As with most empirically derived procedures, the range of applicability of this method is most accurate within the range of data from which it was derived. In order to keep the aircraft design parameters within the historical data ranges, a number of constraint function calculations have been added. If the design exceeds the data range, the inequality constraint is violated, and RQPMIN takes the appropriate remedial action.

During the first call to the drag subroutines, the drag data values are read and assigned to a number of arrays, labelled to correspond with the chart number of the original report. The drag data file is closed and not re-opened, as the drag data values are saved by the subroutine for future use. Variables used throughout the drag methodology are set (including fuselage drag parameters and wetted areas), before the first of the five main parts of the subroutine is entered.

Part 1 - Design Lift Coefficient and Mach Number (CLDES & MD)

The design lift coefficient (CLDES) of the configuration depends on whether the wing has a subsonic (RTW > 0.065), or supersonic (RTW < 0.065) aerofoil thickness/chord ratio. CLDES for subsonic wings, which include the advanced/supercritical sections, is a function of wing sweep, aspect ratio, effective thickness, and camber, and is obtained from interpolation of the tabular data. The data for CLDES of supersonic wings is somewhat easier to model, and is determined from the following expressions:

$$CLDES = -0.06 + 0.475\,X - 0.125\,X^2 \quad \text{, if X is less than 1.2, or} \tag{5-16a}$$

$$CLDES = 0.145 + 0.1625\,X - 0.006944\,X^2 \quad . \tag{5-16b}$$

Where:

$$X = AW \times RTW^{1/3} \quad . \tag{5-17}$$

The lift coefficient upon which the majority of the following drag computations are based is the increment from the design lift coefficient, CLCASE - CLDES, or ΔC_L.

The two-dimensional wing drag divergence Mach number (MD2) is next obtained at the design lift coefficient by interpolation of the relevant tables. Again, different methods are applied for the three different wing section types (subsonic, supercritical, and supersonic). The two-dimensional drag divergence Mach number is converted to a three-dimensional value, by making Mach corrections for sweep (MQW4) and aspect ratio effects (MAR) from the relevant tables. This results in the configuration's design Mach number (MD), and the incremental Mach number (MDD), upon which much of the following drag computation is based:

$$MD = MD2 + MQW4 + MAR \quad , \tag{5-18}$$

$$MDD = M - MD \quad . \tag{5-19}$$

Part 2 - Turbulent Skin Friction (CDF)

The code calculates those component drag values that are independent of lift, the first of which is skin friction. Using data from the aircraft synthesis model, the reference lengths of the fuselage, wing, and tail surfaces are calculated. The reference length for the fuselage is its overall length, whilst for the wing and tail surfaces, it is their individual exposed mean aerodynamic chords. The inlet capture area (from the engine modelling routines) is included in this data to ensure a correct physical representation for the calculation of wetted area and fuselage fineness ratio.

The Reynolds number per metre, flight velocity, and reference length, are used to compute the flat plate skin friction coefficient, CF, for each major airframe component, based on two simple quadratic equations for Reynolds numbers greater and less than 10 million, respectively:

$$CF = 0.004075 - 0.001303 \times 10^{(RN)} + 0.000153 \times 10^{(RN)^2} \quad , \tag{5-20a}$$

$$CF = 0.00445 - 0.002115 \times 10^{(RN)} + 0.000635 \times 10^{(RN)^2} \quad , \tag{5-20b}$$

where RN is the Reynolds number (in millions) of the airframe component.

The fuselage form factor, RCDF, is calculated from the fuselage fineness ratio (RDFL), and is a measure of the ratio of the drag on the fuselage body, relative to the drag on a flat plate. Form factors for the wing (RCDW), and the tail and fin (RCDET & RCDEF) are found from quadratic equation based on the component thickness/chord ratio. A separate equation is used for advanced wing aerofoil sections.

The value of CF previously obtained is corrected for Mach effects by the inclusion of the term CFCOR, which is the ratio of compressible to incompressible friction drag obtained at the desired Mach number:

$$CFCOR = 1.0 - 0.03 \times M - 0.043 \times M^2 \quad . \tag{5-21}$$

For most combat aircraft there is an additional level of miscellaneous drag associated with items such as gun ports, gutters, antennae, blisters, and canopy shape. A realistic overall drag level can successfully be estimated by including a variable percentage allowance for 'excrescence' drag. This factor depends on the overall size (wing area) of the aircraft as, for larger aircraft, it will be a much smaller percentage increase than for a small aircraft. The method for estimating this factor is an up-date of a previous Cranfield model[47]:

$$FACTOR = 1.1778 + \left(0.1 \times \sqrt{\frac{20}{SW}} \right) \quad . \tag{5-22}$$

The turbulent skin friction drag coefficient is then calculated for each component, using the values calculated, where the subscript $_I$ indicates the value for each component:

$$CD0_I = CF_I \times \frac{SWET_I}{SW} \times RCD_I \times CFCOR \quad , \tag{5-23}$$

The total turbulent skin friction drag for the entire aircraft is calculated:

$$CDF = (CD0F + CD0W + CD0ET + CD0EF) \times FACTOR \quad . \tag{5-24}$$

Part 3 - Compressibility Drag (CDC)

The compressibility drag increment due to volume (CDC) is assumed to be composed of form drag - which is a function of shape and volume effects on viscous pressure levels due to increases in local Mach number - and compressibility effects, as local flows become sonic and shock waves form. In the supersonic speed range, form compressibility and zero-lift wave drag are assumed to be synonymous.

In the model, the configuration zero lift compressibility drag is assumed to be composed of three components - that due to wing, fuselage, and wing/body interference effects.

$$CDC = CDCW + CDCF + CDCINT \ . \qquad (5\text{-}25)$$

Wing compressibility drag (CDCW) is represented as a function of thickness/chord ratio, camber, and the Mach number increment, MDD, from the design Mach number. One set of tabular data is used for conventional, advanced, and supersonic aerofoil shapes in the subsonic regime, and another correlation method is used for supersonic flight. This change in correlation parameter was causing discontinuities in wing compressibility drag between the two speed ranges in the original method, but this was corrected during the development of the COMBAT aircraft synthesis code.

Fuselage compressibility drag (CDCF) is estimated using an approximation to the full aircraft area distribution, shown in Figure 5-3, which is not available at the conceptual design stage. Here, in contrast to the friction calculation, the inlet capture areas are removed for internally mounted engines, and fuselage compressibility drag is interpolated from the data tables. It is a function of cross-sectional area excluding the intakes (OPI), the fineness ratio using OPI, and the base drag area (OFGB), which is calculated from the maximum required nozzle exit area, as shown in section 4.2.2.

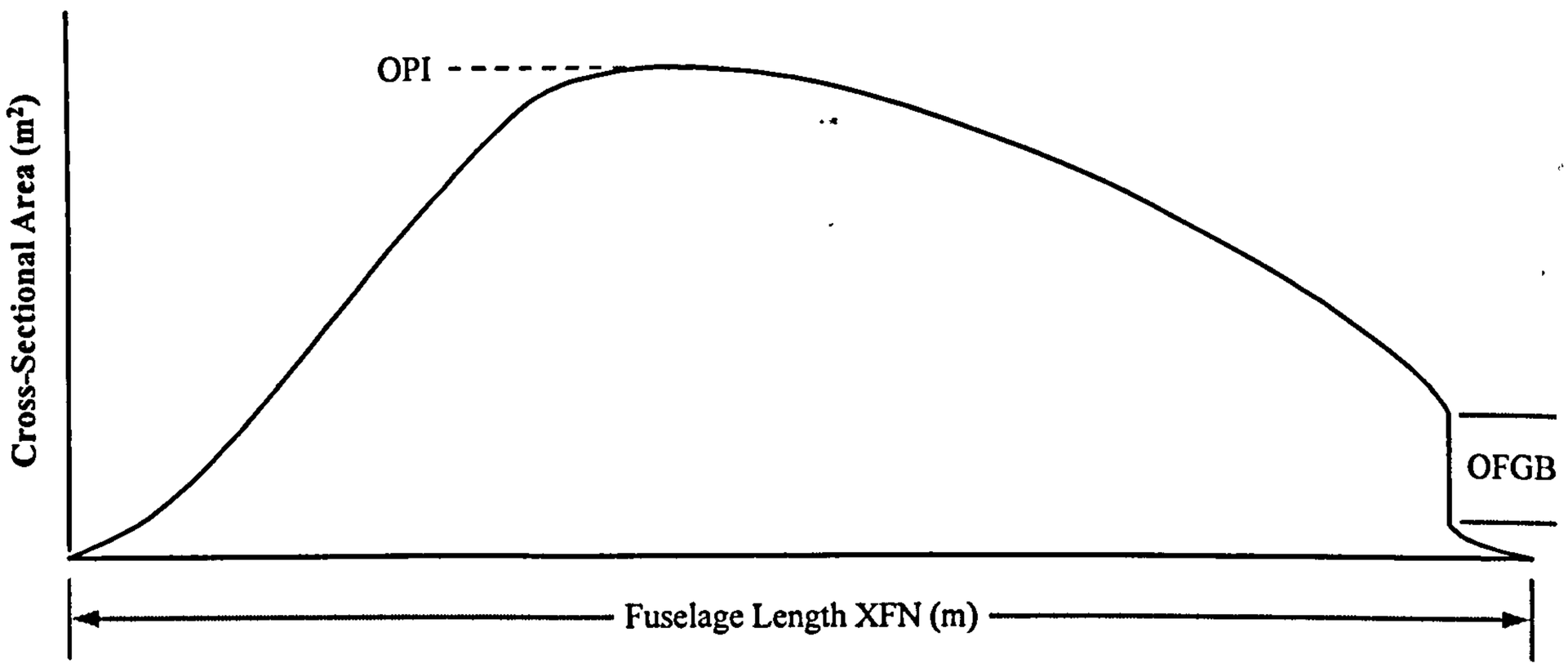


Figure 5-3. Compressibility Drag Fuselage Geometry Representation.

Wing/fuselage interference drag (CDCINT), as derived for this method, is tabulated as a function of body diameter to wing span ratio, taper ratio, and wing sweep. The assumption is made that interference drag is zero below M = 1.0.

The values of CDCW, CDCF, and CDCINT are combined to produce the total configuration compressibility drag, using Equation 5-25. The resulting values, when added to the previously computed level of friction drag and any other zero-lift drag data, will produce a configuration drag level that is independent of lift throughout the design Mach range.

Figure 5-4 shows an example of the relative contributions to the total compressibility drag coefficient (CDC) of the fuselage (CDCF), wing (CDCW), and interference (CDCINT).

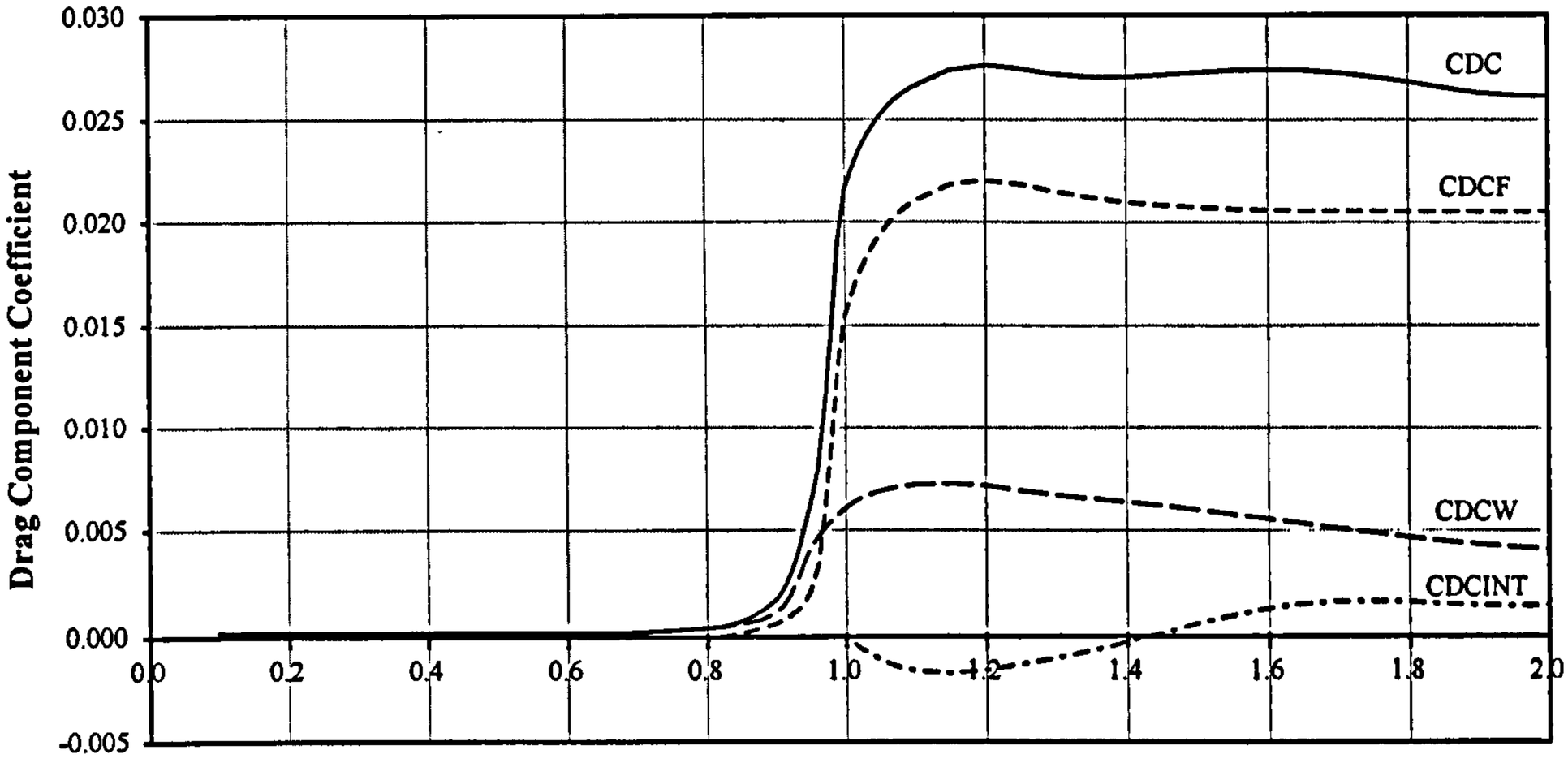


Figure 5-4. Example Compressibility Drag Build-Up.

Miscellaneous Drag

Before the calculation of the total zero-lift drag coefficient, the miscellaneous zero-lift drag factors are calculated. These include the drag due to the deployment of the landing gear and flaps, and due to the presence of external stores and fuel. The landing gear drag (CDADDL) is calculated from the aircraft all-up mass and wing area, whilst the zero-lift flap drag (CDADDF) is a function of the flap dimension parameters FBWFI, FBWFO & RCWF, and the thickness/chord ratio (RTW) and flap angle (QFLA).

The external weapons and fuel drag estimates (CDADDS) were added during the development of COMBAT, and are based on the mass of external stores, and the flight Mach number. The drag coefficient data were taken from Raymer[78] and Nicolai[71], and are shown in Figure 5-5. It should be noted that drag data for external fuel carriage at supersonic speeds was not available; it is therefore not feasible to accurately model supersonic mission legs with external fuel carriage.

The method compares the external fuel or weapons mass with the default values (175 kg for AIM 9 & pylon, 441.7 kg for 150 gallon & 3533.5 kg for 2 × 600 gallon), and scales the drag coefficient. The reference area used by both Raymer and Nicolai was 26 m^2, which has also been accounted for in the model.

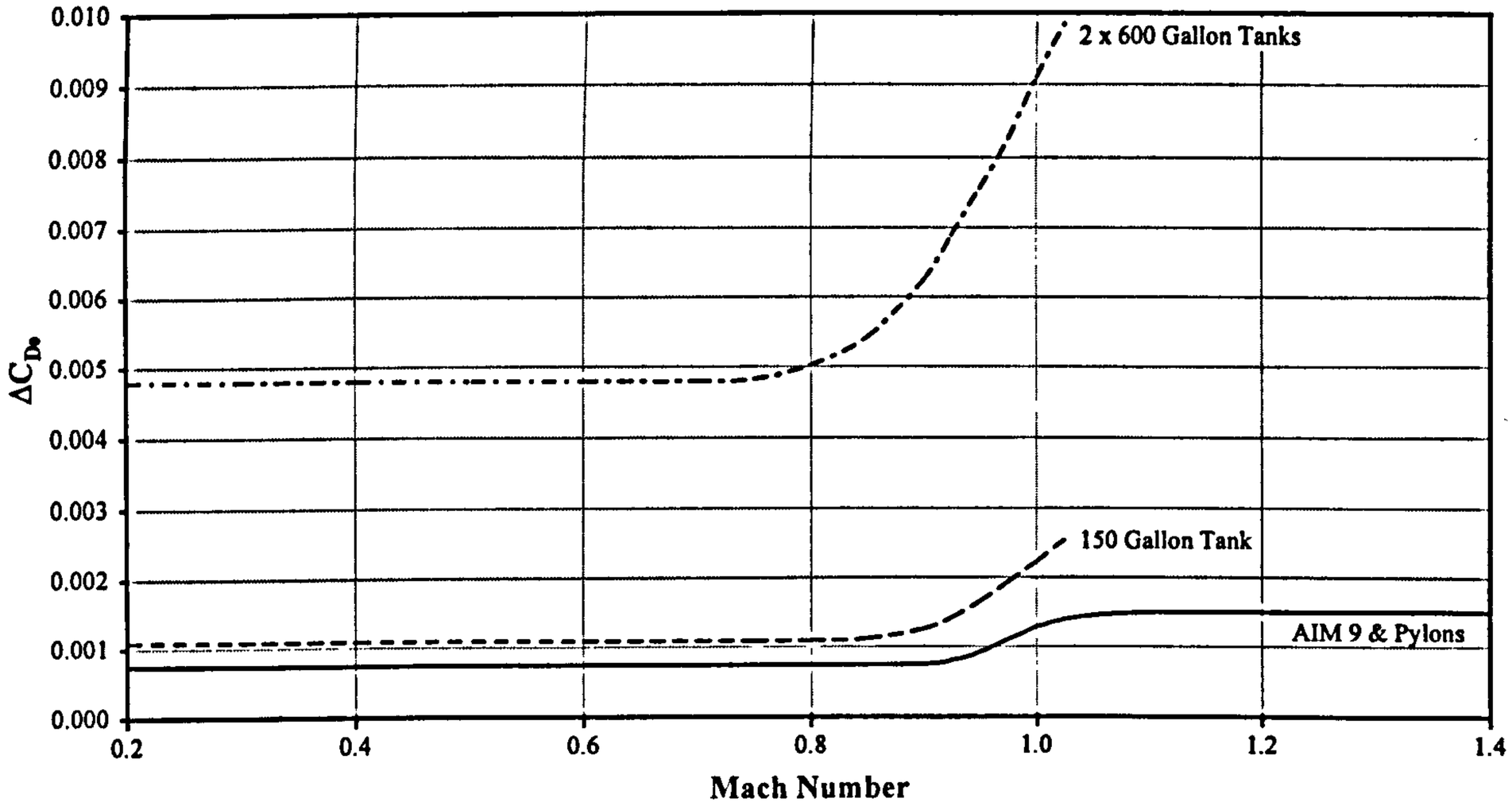


Figure 5-5. External Store Zero-Lift Drag Coefficients.

The total additional drag is the sum of the three components:

$$CDADD = CDADDL + CDADDF + CDADDS \tag{5-26}$$

The total zero-lift drag is simply the sum of the friction, compressibility, and additional drag values:

$$CD0 = CDF + CDC + CDADD \tag{5-27}$$

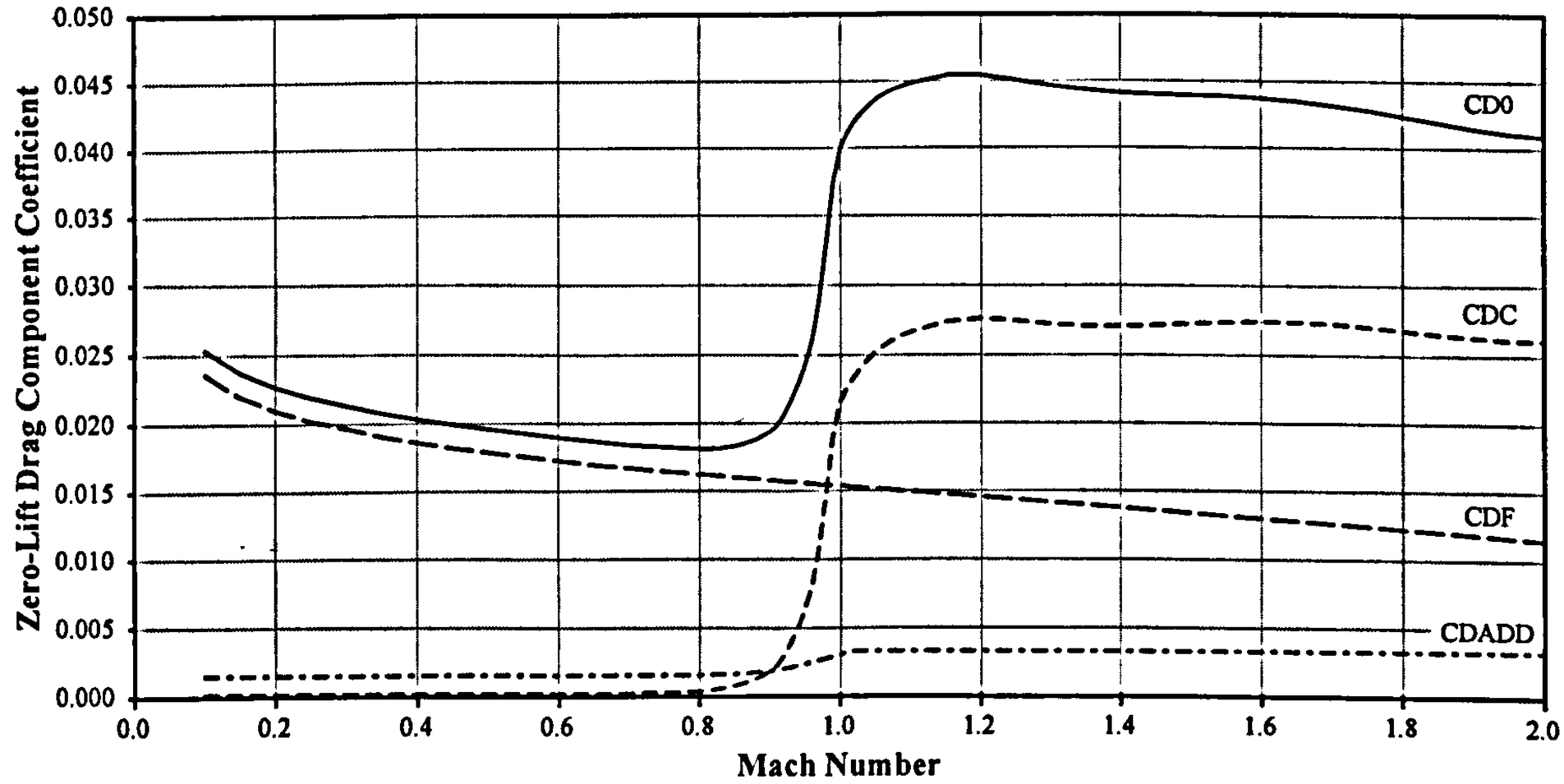


Figure 5-6. Example Zero-Lift Drag Build-Up.

The lack of data for external fuel tanks above M = 1.025 gives rise to the sudden flattening of the additional drag coefficient values at supersonic speeds in Figure 5-6.

Part 4 - Induced and Wing Pressure Drag (CDP)

Lift-dependent drag is predicted as the sum of the theoretical induced drag ($C_L^2/\pi AW$) and a derived term called the wing pressure drag. The theoretical value is used to define the primary variation of the induced drag with lift coefficient, and assumes that Oswald's lift efficiency factor is equal to unity. The wing pressure term is, therefore, composed of all lift-dependent drag items over and above that of theory.

The wing pressure term (CDP) is the induced drag increment above that produced by the theoretical level resulting from $C_L^2/\pi AW$. It is primarily a function of the wing shape, and includes a combination of many effects (separation, lift-dependent compressibility drag, span-wise flow, body effects, etc.), none of which are easily computed with the limited aircraft data available.

Several tables contain the subsonic pressure drag data as a function of aspect ratio, thickness/chord ratio, camber, MDD, and CLCASE-CLDES. The tables are interpolated twice, once using MDD and ΔC_L to predict CDP, and these values are then used for a second interpolation based on $AW\times(RTW)^{1/3}$, as in Equation 5-17.

Part 5 - Total Aircraft Drag (CDCASE)

The drag items computed under the previous four parts are now combined with the theoretical induced drag levels, using the relationship:

$$CDCASE = CD0 + CDP + \frac{CLCASE^2}{\pi \times AW} \quad . \tag{5-28}$$

An example of the total clean lift coefficient is shown in Figure 5-7, based on the design data for the F-16C, together with data that was available for the YF-16. The results can be seen to agree fairly well, but differences between the prototype and in-service aircraft prevent any firm conclusions being drawn about the accuracy of the code.

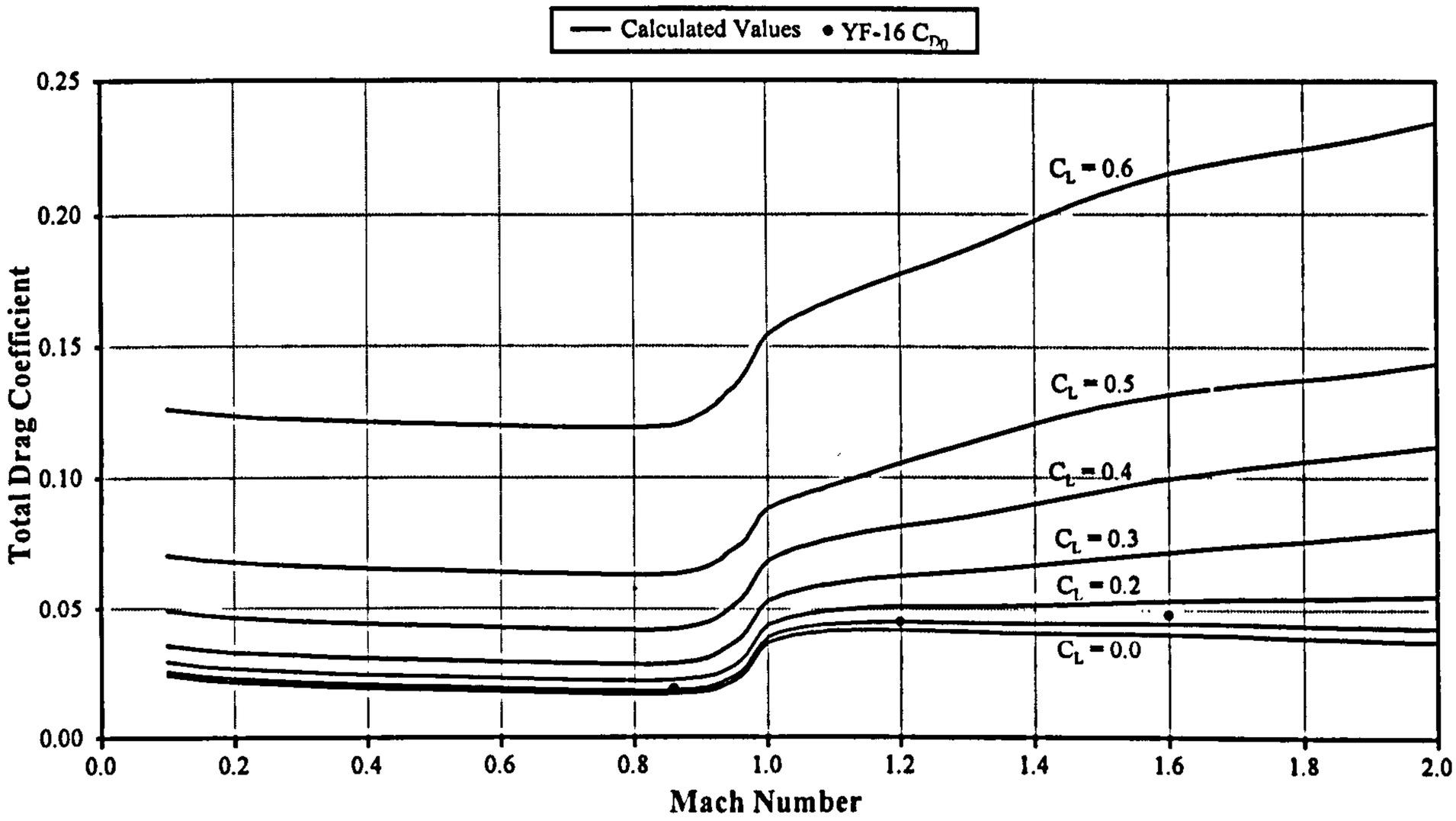


Figure 5-7. Example Total Drag Coefficient.

5.2 Engine Modelling Routines

The ONX and OFFX codes were originally written by Dr. Jack Mattingly to perform gas turbine machinery cycle analysis, and were presented as both source and executable code with the first edition of Mattingly's first book[65]. However, due to the commercial exploitation of the source code, only the program executable was supplied with future printings. The source code was kindly made available on a limited basis, and several modifications and improvements to the accuracy of the code have been made.

The object of cycle analysis is to obtain estimates of the *performance parameters* (thrust and specific fuel consumption) in terms of *design limitations* (such as turbine inlet temperature and component efficiencies), the *flight conditions*, and *design choices* (such as compressor pressure ratio, bypass ratio, etc.)

Engine design starts with the on-design (or design-point) analysis, which presumes that all design choices are still under control and that the size of the engine is yet to be fixed. This is usually known as the "rubber" engine-sizing process, where the performance parameters are given as "specific" values that are normalised or factored with the engine size. Thus, each complete set of design choices will result in an engine with its own operating and performance characteristics. This section of the engine design is performed by the subroutine ONX, and from relatively simple starting values, the nature of the engine cycle is determined.

Once the engine cycle, design point, and engine limitations have been set, the engine is analysed away from that design point by the off-design analysis program, OFFX. This program returns all of the major performance parameters for a particular engine and flight condition. From these values, the fuel burn for each mission leg is estimated and this, in turn, leads to the calculation of the total mission fuel required.

The ONX/OFFX models deal will all types of two-shaft gas turbine machinery, including turboprops, turbofans (with and without reheat), and ramjets, but the method has been limited for the purposes of this study, to turbojets and turbofans, with and without reheat. The generalised engine shown in Figure 5-8 gives an impression of the level of thermodynamic modelling capability, and Table 5-2 describes the locations and station numbering used in the standardised engine nomenclature.

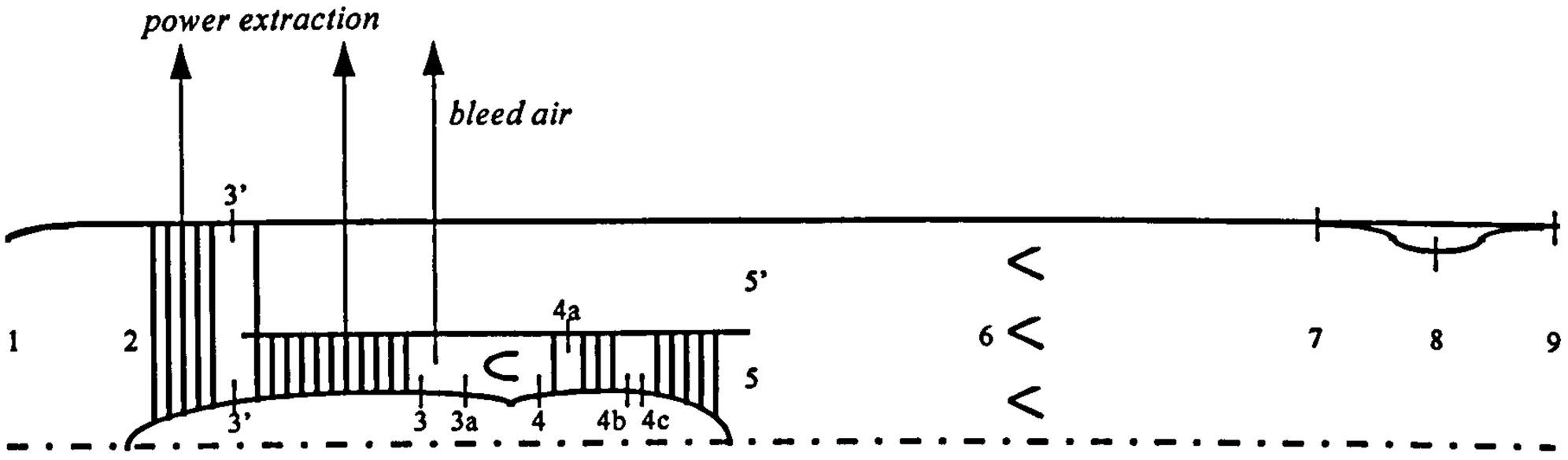


Figure 5-8. Reference Stations - Mixed-Stream Turbofan Engine[65].

Station	Location	Station	Location
0	Far upstream or freestream	4b	High pressure turbine exit Modelled coolant mixer 2 entry
1	Inlet or diffuser entry	4c	Coolant mixer 2 exit Low pressure turbine entry
2	Inlet or diffuser exit Fan entry	5	Low pressure turbine exit Core stream mixer entry
3′	Fan exit High pressure compressor entry	5′	Fan bypass stream mixer entry
3	High pressure compressor exit	6	Mixer exit Afterburner entry
3a	Burner entry	7	Afterburner exit Exhaust nozzle entry
4	Burner exit Nozzle vanes entry Modelled coolant mixer 1 entry High pressure turbine entry for total pressure ratio definition	8	Exhaust nozzle throat
4a	Nozzle vanes exit Coolant mixer 1 exit High pressure turbine entry for τ_{tH} definition	9	Exhaust nozzle exit

Table 5-2. Engine Reference Station Definitions.

Because of the complexity of the engine cycle analysis procedures, it is not the intention of this section to detail the whole methodology. Instead, the user is directed to "Aircraft Engine Design"[65], Chapters 4 and 5, which contain a full description of the design and cycle analysis procedure. This section will detail those areas not specifically included in the book, the changes made to the original method, and the overall engine control subroutines, which were written specifically for the COMBAT design synthesis program.

Many different engine design parameters can be altered, including the design flight conditions, low and high-speed shaft bleed & power off-take values, component pressure ratios, polytropic efficiencies, mechanical efficiencies, and off-design engine limits. Some of these are clarified below, although where explanations are valid for several variables, only the first is explained. Number subscripts refer to the engine stations shown in Figure 5-8.

Altitude	Assumed engine design flight altitude. (m)
Mach Number	Assumed engine design flight Mach number.
Pi Compressor	Total compressor pressure ratio. (P_3/P_2)
Pi Fan	Fan pressure ratio. (P_3'/P_2)

Bypass Ratio	Ratio of bypass to core air mass flow rate at design conditions.
TT4 Max. (TET)	Maximum turbine entry temperature (TET). (°R)
TT7 Max. (A/B)	Maximum afterburner jet-pipe temperature, if applicable. (°R)
Fuel Heating Value	Fuel heating value. (Btu/lb.m)
Bleed Air	Percentage of core (HP) airflow removed for a/c systems. (%)
Cooling #1 or #2	Cooling airflow for HPT & HPT nozzle vanes. (%)
Power T-O Low/High	Mechanical power take-off coefficient for low & high-speed shafts.
Pressure Ratio	Ratio of total pressure leaving / total pressure entering component.
Polytropic Efficiency	Efficiency of rotating machinery components, representing level of technology, rather than behaviour of a given device. (%)
Combustion Efficiency	Degree to which chemical reactions are completed. (%)
Spool T-O Efficiency	Ratio of mechanical power output / mechanical power input to account for losses due to windage, friction & seal drag. (%)
P0/P9	Ratio of freestream pressure to exit pressure. 1.0 for max. thrust.
Mach @ Stn #5	Core stream mixer entry Mach number.

The following text describes the off-design engine limits, which can be used to ensure that the off-design performance parameters of the engine components remain within, or close to, the design values. If any of the limits appearing below "Max. Comp. Ratio" are set to zero, the limits are ignored, but this may over-predict thermodynamic performance for the designed engine.

TT4 / TT7 Max.	max. off-design turbine entry / afterburner temperature. (°R)
P0/P9 Max.	max. off-design freestream-to-exit pressure ratio.
Max. Comp. Ratio	max. compressor pressure ratio.
Max. P @ Stn 3	max. temperature at HP compressor exit (material limit).
Max. T @ Stn 3	max. temperature at HP compressor exit (material limit). (°R)
Max. % RPM	max. ratio of low or high-pressure shaft speed to design value.

All variables required for the engine design model are described in detail in "Aircraft Engine Design", but suggested values for some that vary with different technological eras can be found in Table 5-3. The technology levels 1 to 4 roughly correspond to the technology available in the time frames; pre-1965, late 1970's, 1990's, and the next generation of aircraft engines likely to be designed post-2000.

An example of the state of current technology is the French SNECMA M88-2 engine for the Dassault Rafale, which has the highest published turbine inlet temperature (T_4), of 1850K (3330R). This value is approximately one third of the way between technology levels 3 and 4, and this is the suggested start point for any engines entering service in the late 1990's or early next century. Therefore, to design an engine for an aircraft entering service in the years just post-2000, values slightly higher than those under 'Level of Technology' 3 in Table 5-3 would be used.

Component	Figure of Merit	Type	Level of Technology			
			1	2	3	4
Diffuser	$\pi_{d\,\max.}$	*A* *B* *C*	0.90 0.88 0.85	0.95 0.93 0.90	0.98 0.96 0.94	0.995 0.98 0.96
Fan	e_f		0.78	0.82	0.86	0.89
Compressor	e_c		0.80	0.84	0.88	0.90
Burner	π_b η_b		0.90 0.85	0.92 0.91	0.94 0.98	0.95 0.99
Turbine	e_t	Uncooled Cooled	0.80	0.85 0.83	0.89 0.87	0.90 0.89
Afterburner	π_{ab} η_{ab}		0.90 0.85	0.92 0.91	0.94 0.96	0.95 0.99
Nozzle	π_n	*D* *E* *F*	0.95 0.93 0.90	0.97 0.96 0.93	0.98 0.97 0.95	0.995 0.98 0.97
Maximum T_3		(°R)	1200	1450	1600	1700
Maximum T_4		(°R)	2000	2500	3200	3600
Maximum T_7		(°R)	2500	3000	3600	4000

A = subsonic aircraft with engines in nacelles

B = subsonic aircraft with engine(s) in airframe

C = supersonic aircraft with engine(s) in airframe

D = fixed-area convergent nozzle

E = variable-area convergent nozzle

F = variable-area convergent-divergent nozzle

Table 5-3. Component Figures of Merit for Different Technological Levels.

5.2.1 Code Modifications

The original executable code was a set of menu-driven programs that could perform cycle analysis calculations for a single design/analysis point, or for varying design limitations, flight conditions, or design choices, to produce tabular data. As the menu-driven system adopted by the original ONX/OFFX code was not compatible with current requirements, the menus were removed and the data input was changed to a data file system, similar to that for the aircraft design program described in Chapter 4.

The fuel burn calculations in the original code were simple and accuracy was only fairly good, as the changes in gas properties of the working fluid passing through the engine were required as fixed input variables. Thus, the change in temperature and air/fuel ratio for different cycles was not correctly accounted for, as the user had to make an estimate of how the gas property values c_p and γ would change through the engine cycle. This simplified calculation was replaced with the more accurate, iterative, method suggested in Mattingly's second book[66].

The gas properties c_p and γ are calculated from the fluid temperature, and the air/fuel ratio for the three major changes that occur through the engine; i.e. ambient conditions, in the burner, and at the afterburner. Gas properties are assumed to remain constant across components; c_p and γ do not change within the compressors/turbines, etc.

Because the naming convention used by Mattingly has been retained for the ONX/OFFX engine modelling routines, the following symbols are defined:

α = engine bypass ratio
A_n = area at station 'n'
A_n^* = area corresponding to $M = 1$, or ideal area, at station 'n'
β = bleed air fraction
C_{pM} = specific heat at constant pressure in the mixer
C_{pc} = specific heat at constant pressure upstream of main burner
C_{pt} = specific heat at constant pressure downstream of main burner
ε_1 = cooling air #1 mass flow rate
ε_2 = cooling air #2 mass flow rate
f = burner fuel/air ratio
f_{AB} = afterburner fuel/air ratio
η_{AB} = efficiency of afterburner
η_b = efficiency of burner
h_{PR} = heating value of fuel
h_{tn} = total enthalpy at station 'n'
P_{TO} = power off power take-off shaft
T_0 = temperature at station 0 (inlet)
$\tau_{c'}$ = total temperature ratio of fan
τ_{cH} = total temperature ratio of high pressure compressor
τ_λ = enthalpy ratio of burner
$\tau_{\lambda AB}$ = enthalpy ratio of afterburner
τ_M = total temperature ratio of mixer
τ_{m1} = total temperature ratio of station 4a to 4
τ_{m2} = total temperature ratio of station 4c to 4b
τ_{tH} = total temperature ratio of high pressure turbine
τ_{tL} = total temperature ratio of low pressure turbine

The following major algebraic changes to the code were made, concerning the calculation of the fuel/air ratios at the burner and afterburner:

Equation 4.15 from "Aircraft Engine Design"[65], which reads

$$f = \frac{\tau_\lambda - \tau_r \tau_{c'} \tau_{cH}}{h_{PR}\,\eta_b \,/\, (C_{pc}\, T_0) - \tau_\lambda}$$

was removed and replaced with Equation 6-36 from "Elements of Gas Turbine Propulsion"[66]:

$$f = \frac{h_{t4} - h_{t3}}{\eta_b \, h_{PR} - h_{t4}} \quad . \tag{5-29}$$

In this last case, the value of h_{t4} is a function of the air/fuel ratio, f, and thus the solution is iterative.

A similar procedure is used to calculate the afterburner air/fuel ratio, where Equation 4.16 from "Aircraft Engine Design", which reads

$$f_{AB} = \left(1 + f\frac{1-\beta-\varepsilon_1-\varepsilon_2}{1+\alpha-\beta}\right)\left\{\frac{\tau_{\lambda AB} - \tau_\lambda \tau_{m1} \tau_{tH} \tau_{m2} \tau_{tL} \tau_M \left(C_{pM}/C_{pt}\right)}{h_{PR}\eta_{AB}/\left(C_{pc}T_0\right) - \tau_{\lambda AB}}\right\}$$

is removed, and replaced with Equation 7-118 from "Elements of Gas Turbine Propulsion":

$$f_{AB} = \frac{h_{t7} - h_{t6}}{\eta_{AB} \, h_{PR} - h_{t7}} \quad , \tag{5-30}$$

and again the solution of f_{AB} is iterative.

The enthalpy values of the calculations are returned by the subroutine that is also responsible for calculating the c_p and γ values for the gas properties. This makes for a very efficient and elegant solution to the problem, and improves the accuracy of the fuel burn calculations. However, a small run-time penalty is incurred because of the iterative nature of the calculations.

5.2.2 Engine Design and Installation

Before the engine can be 'flown' in the aircraft, its thermodynamic cycle must first be decided, and the inlet and nozzle parameters must be set to ensure the correct calculation of installation losses. These processes are controlled by a subroutine that is called during the initialisation of the aircraft sizing process.

The data for the ONX engine design model are read from the engine input file. Together with the air parameters from the atmosphere model, they are passed to ONX, where the basic engine calculations are performed for a datum air mass flow. The engine cycle design is now fixed, and most other changes to the engine are given by changes in the air mass flow rate. The engine performance is analysed at all given point and mission flight conditions to determine feasible sizing parameters for the inlet and nozzle.

The size of the inlet area (A_1) for a fixed area inlet is required to estimate the inlet losses (ϕ_{inlet}) for sub- and supersonic inlets. The assumption of a fixed-area inlet simplifies the problem of finding the off-design condition(s) corresponding to the largest inlet area. Since the absolute engine size is not yet fixed, the required inlet area is referenced to the freestream area of the engine design air flow at its design point (A_{0ref}), giving a required value of $\frac{A_1}{A_{0ref}}$. When the engine is re-sized (new air mass flow and A_{0ref}), then the new required inlet capture area can be determined directly, because both the flight condition and the above area ratio are constant.

During subsonic flight, the engine airflow is usually accelerated from the freestream Mach number (M_0) to the inlet capture Mach number (M_1). In order to prevent choking of the inlet, M_1 must be less than unity, and usually less than 0.8; i.e. the inlet capture area (A_1) must be slightly larger than the area that would be required to choke the engine flow, $A_1^* = A_0^*$. For subsonic flight, the flight condition with the largest A_0^* determines the inlet capture area, normally corresponding to low-speed, high altitude flight.

However, low-speed flight at any altitude will cause large inlet capture areas, leading to corresponding thrust losses and increased drag during high-speed flight. To prevent overly large intakes, flight conditions causing an A_1 larger than that required at $M_0 = 0.9$ @ 30000 ft (9144 m) for supersonic aircraft, and $M_0 = 0.8$ @ 30000 ft for subsonic aircraft, are ignored. This is a typical compromise struck in practical inlet design, and inlets are often seen to have auxiliary inlet doors to compensate for this during low-speed, high-thrust flight modes. Allowing for a small safety margin, the inlet sizing equation becomes:

$$\frac{A_1}{A_{0ref}} = 1.04 \times \left(\frac{A_1}{A_1^*}\right)_{M_1=0.9} \times \frac{A_0^*}{A_{0ref}} \quad . \tag{5-31}$$

During supersonic flight, no freestream deceleration or streamtube contraction is expected, so the fixed inlet capture area (A_1) must simply exceed the largest required engine airflow area at freestream conditions (A_0) by the minimum amount needed for boundary layer bleed and a margin of safety. The amount of boundary layer bleed will be about 4% for a Mach 2 external compression inlet, and this has been used in the calculations. This again leads to the sizing of the inlet:

$$\frac{A_1}{A_{0ref}} = 1.04 \times \frac{A_0}{A_{0ref}} \quad . \tag{5-32}$$

In order to evaluate the nozzle loss (ϕ_{nozzle}), the nozzle exit area (A_9), the maximum diameter ahead of the nozzle exit (A_{10}), and the nozzle length (L) is required. The nozzle exit area (A_9) for any flight condition is obtained directly from OFFX. A_{10} may not be smaller than A_9 for any flight condition, and therefore it is set to be just larger than the largest A_9 that is required by the mission specification; A_{9max} normally occurs at maximum throttle at maximum speed.

The nozzle length (L) is more difficult to establish, particularly for subsonic aircraft where there is no clear break between the aft fuselage and the nozzle itself. Currently, the nozzle length is expressed as a fraction of the D_{10} diameter - $L = 1.7 \times D_{10}$ for subsonic aircraft, and $L = 1.1 \times D_{10}$ for supersonic aircraft.

For maximum thrust at all flight conditions, except maximum Mach number, the maximum pre-nozzle exit area is calculated:

$$\frac{A_{10}}{A_{0ref}} = 1.1 \times \frac{A_9}{A_0} \times \frac{A_0}{A_{0ref}} \quad . \tag{5-33}$$

This completes the engine thermodynamic cycle design, inlet area ratio, and maximum nozzle area, which are also used to define the inlet and fuselage base areas.

5.2.3 Off-Design Analysis & Engine Losses

Thus far, only the uninstalled thrust (F) that would be produced by an isolated engine has been considered. On installing the engine in the airframe, there will be losses in the intake and nozzle that cannot be ignored. This methodology treats these internal losses as the "self-drag" of the engine, which must be overcome through extra thrust from the engine itself. The thrust losses must be calculated from the delivered thrust, which is not yet known, leading to an iterative solution for each value of delivered thrust.

When the OFFX code is called, the flight conditions (altitude & Mach number) and the engine operating mode (Maximum/Military or required thrust) are specified. According to the flight phase and conditions, the following values are also set:

1. Altitude (m)
2. Mach number
3. Atmospheric temperature (°C)
4. Atmospheric pressure (Pa)
5. Engine mode, or Thrust (N)
6. Reheat mode (0 - off, 1 - on)

As with the ONX program, all values are converted from Metric to Imperial Units before passing them to the OFFX engine subroutine.

The inlet and nozzle loss (ϕ_{inlet} and ϕ_{nozzle}) equations and methodology are again taken directly from "Aircraft Engine Design", Chapter 6. Only the inlet and nozzle external losses are considered, and are represented in their most convenient form - as a fraction of the uninstalled thrust, so that:

$$\phi_{inlet} = \frac{D_{inlet}}{F} \quad , \tag{5-34}$$

$$\phi_{nozzle} = \frac{D_{nozzle}}{F} \quad \text{, and} \tag{5-35}$$

$$T = F - (\phi_{inlet} F) - (\phi_{nozzle} F) \quad \text{, or}$$

$$F = \frac{T}{1 - \phi_{inlet} - \phi_{nozzle}} \quad . \tag{5-36}$$

Where:

D_{inlet} = drag of the inlet system
D_{nozzle} = drag of the nozzle system
F = total uninstalled propulsive force available from the engine
T = total installed propulsive force available from the engine

These losses are dependent on the flight conditions, the ratio of the relevant areas, and the current engine performance conditions. Thus, for each mission leg, the engine internal losses will change, requiring an iterative solution before the analysis is complete. Figure 5-9 shows a graphical representation of the engine performance analysis procedure.

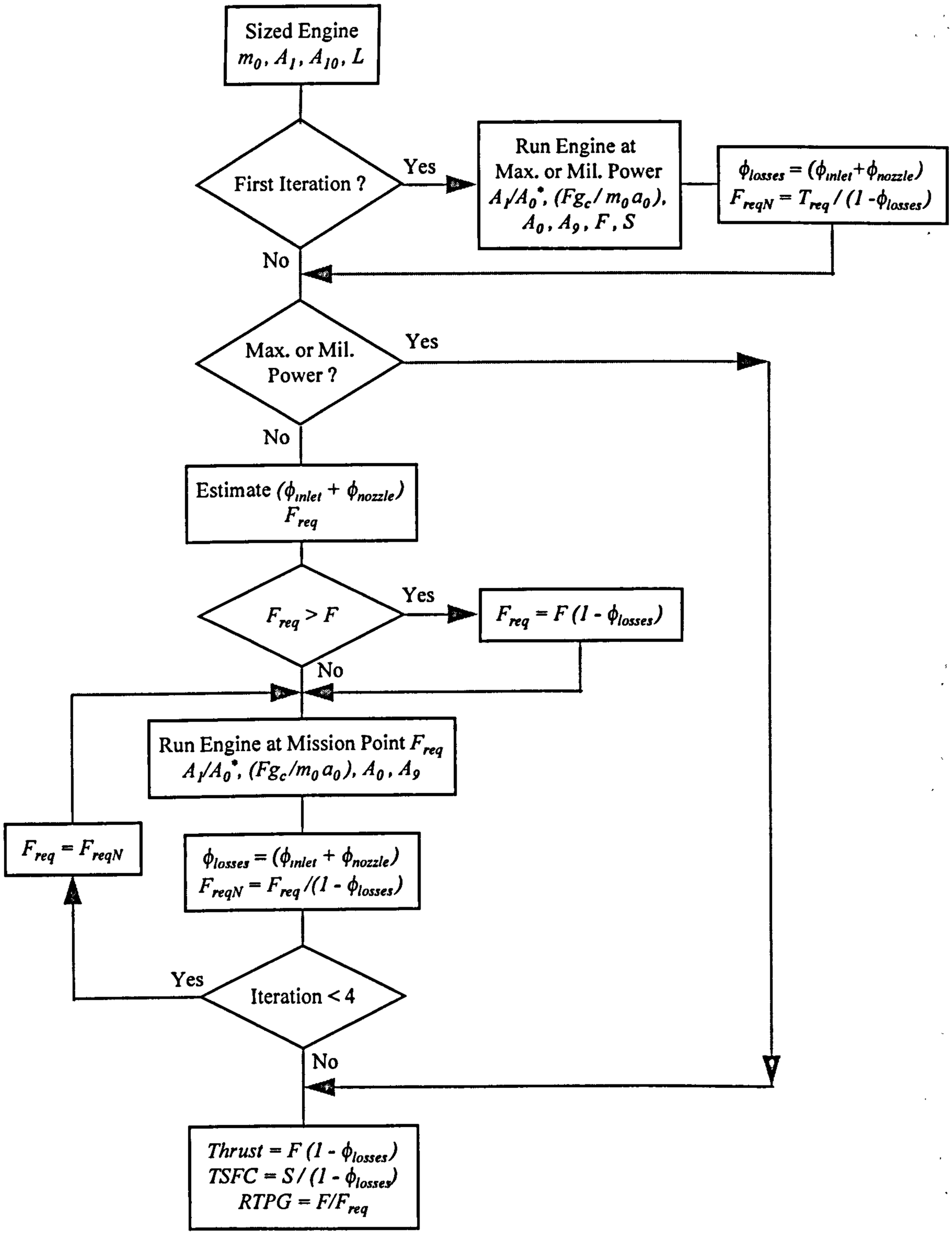


Figure 5-9. Engine Mission Analysis Procedure Flow Chart.

It should be noted that the final calculation in the above analysis is the ratio of available to required thrust, which becomes the optimiser constraint function, RTPG. At every call where the engine is called with a required thrust value, this ratio is calculated and compared with the previous value in the aircraft synthesis code, to ensure that all flight thrust requirements are met. This is very useful, as it allows the thrust at every mission phase to be checked using only a single variable, improving the robustness of the code.

The theory behind the calculation of inlet losses is quite complex, and only the major equations used are given here:

Subsonic flight: $M_0 < 1$

$$\phi_{inlet} = \frac{\frac{M_0}{M_1}\sqrt{\frac{T_1}{T_0}}\left(1+\gamma M_1^2\right)-\left(\frac{A_1}{A_0}+\gamma M_0^2\right)}{\left(\frac{Fg_c}{m_0}\right)\left(\frac{\gamma M_0}{a_0}\right)} \quad . \tag{5-37}$$

Supersonic flight: $M_0 \geq 1$

$$\phi_{inlet} = \frac{\left(\frac{A_1}{A_0}-1\right)\left\{M_0-\sqrt{\frac{2}{\gamma+1}+\frac{\gamma-1}{\gamma+1}M_0^2}\right\}}{\left(\frac{Fg_c}{m_0}\right)\left(\frac{\gamma M_0}{a_0}\right)} \quad . \tag{5-38}$$

Where:

$$M_1 = f\left(\frac{A_1}{A_0^*}\right) \quad , \quad \left(\frac{A}{A^*}\right)_1 = \frac{A_1}{A_0^*} \quad , \quad \frac{T_1}{T_0} = \frac{1+\frac{\gamma-1}{2}M_0^2}{1+\frac{\gamma-1}{2}M_1^2} \quad , \text{and} \quad \frac{A_1}{A_0} = \frac{\frac{A_1}{A_0^*}}{\left(\frac{A}{A^*}\right)_{M_0}} \quad .$$

The estimation of nozzle losses (ϕ_{nozzle}) is more complicated than the estimation of ϕ_{inlet}, and once again the full description of the methodology can be found in "Aircraft Engine Design", Chapter 6. The method has been developed to predict the exhaust nozzle pressure drag in subsonic and supersonic flows, based on the "integral mean slope" (*IMS*) as defined by the equation:

$$IMS = \frac{1}{\left(1-\frac{A_9}{A_{10}}\right)}\int_{\frac{A_9}{A_{10}}}^{} \frac{d\left(\frac{A}{A_{10}}\right)}{d\left(\frac{x}{R_{10}-R_9}\right)} d\left(\frac{A}{A_{10}}\right) \quad , \tag{5-39}$$

where the geometrical areas are the same as already defined, x is the axial distance along the nozzle, from nozzle start to nozzle exit, and R is the radius at the points already mentioned.

It immediately becomes clear that this is a complex analysis, although it may be simplified by assuming a 'generic' nozzle shape; i.e. the exact shape of the nozzle is ignored. In practice, this is acceptable, because the detailed nozzle shape does not too severely affect the outcome of the calculations. The calculation of *IMS* is thus reduced to:

$$IMS \approx 1.8\left(\frac{D_{10}-D_9}{L}\right)\left(1-\frac{D_9}{D_{10}}\right) \quad . \tag{5-40}$$

The drag coefficients are then obtained from Figures 6.6, 6.7, and 6.8 of "Aircraft Engine Design". The data is represented in a tabular format, and fed into the same interpolation routines as were used in the drag estimation procedure. The equations below are then used to calculate the nozzle loss for each flight condition.

Subsonic Flight: $M_0 < 0.8$

$$\phi_{nozzle} = \frac{M_0 \frac{C_D}{2}\left(\frac{A_{10} - A_9}{A_0}\right)}{\left(\frac{Fg_c}{m_0 a_0}\right)}, \tag{5-41}$$

where the nozzle drag coefficient, C_D, is a function of the nozzle properties interpolated from the charts using *IMS*, as calculated above.

Transonic Flight: $0.8 \le M_0 \le 1.2$

$$\phi_{nozzle} = \frac{M_0 \frac{C_{DP}}{2}\left(\frac{A_{10}}{A_0}\right)}{\left(\frac{Fg_c}{m_0 a_0}\right)}, \tag{5-42}$$

where C_{DP} is a function of the freestream Mach number, M_0.

Supersonic Flight: $M_0 > 1.2$

A similar equation is used as for subsonic flight, but now C_D at $M_0 = 1.2$ is a function of *IMS*, and C_D at $M_0 > 1.2$ is found from the empirical relation

$$C_D(M_0) = C_D(1.2)\,\frac{1 - 1.4\,e^{-M_0^2}}{\sqrt{M_0^2 - 1}}\,. \tag{5-43}$$

At the end of the call to the engine-control subroutine, the following values are returned to the aircraft synthesis modules to complete the engine thermodynamic analysis:

Thrust, (N)	Design jet-pipe temperature T_7, (°R)
TSFC, (mg/N s)	Design low-speed power take-off coeff.
Area A_0, (m^2)	Design hi-speed power take0off coeff.
Upstream choked area, A_0^*, (m^2)	Inlet loss, ϕ_{inlet}
Design freestream area, A_{0ref}, (m^2)	Nozzle loss, ϕ_{nozzle}
Upstream choked design area, $A_0^*{}_{ref}$, (m^2)	Bypass ratio, α
Area ratio, A_9/A_0	Design bypass ratio, α_{ref}
Mass flow rate, (kg/s)	Compressor pressure ratio, π_c
Design mass flow rate, (kg/s)	Turbine inlet temperature off-design limit
Design turbine inlet temperature T_4, (°R)	Jet-pipe temperature off-design limit

The figures below show the relevant thermodynamic properties of an engine similar to the Pratt and Whitney F100-PW-220, which powers the F-16C. The example engine has the following properties at sea level static operating conditions when installed in an aircraft similar to the F-16C:

Air Mass Flow Rate	103.4 kg/s	Turbine Inlet Temperature	3010 °R
Bypass Ratio	0.7	Jet-Pipe Temperature	4000 °R
Compressor Pressure Ratio	24.8	Intake System	Single-shock

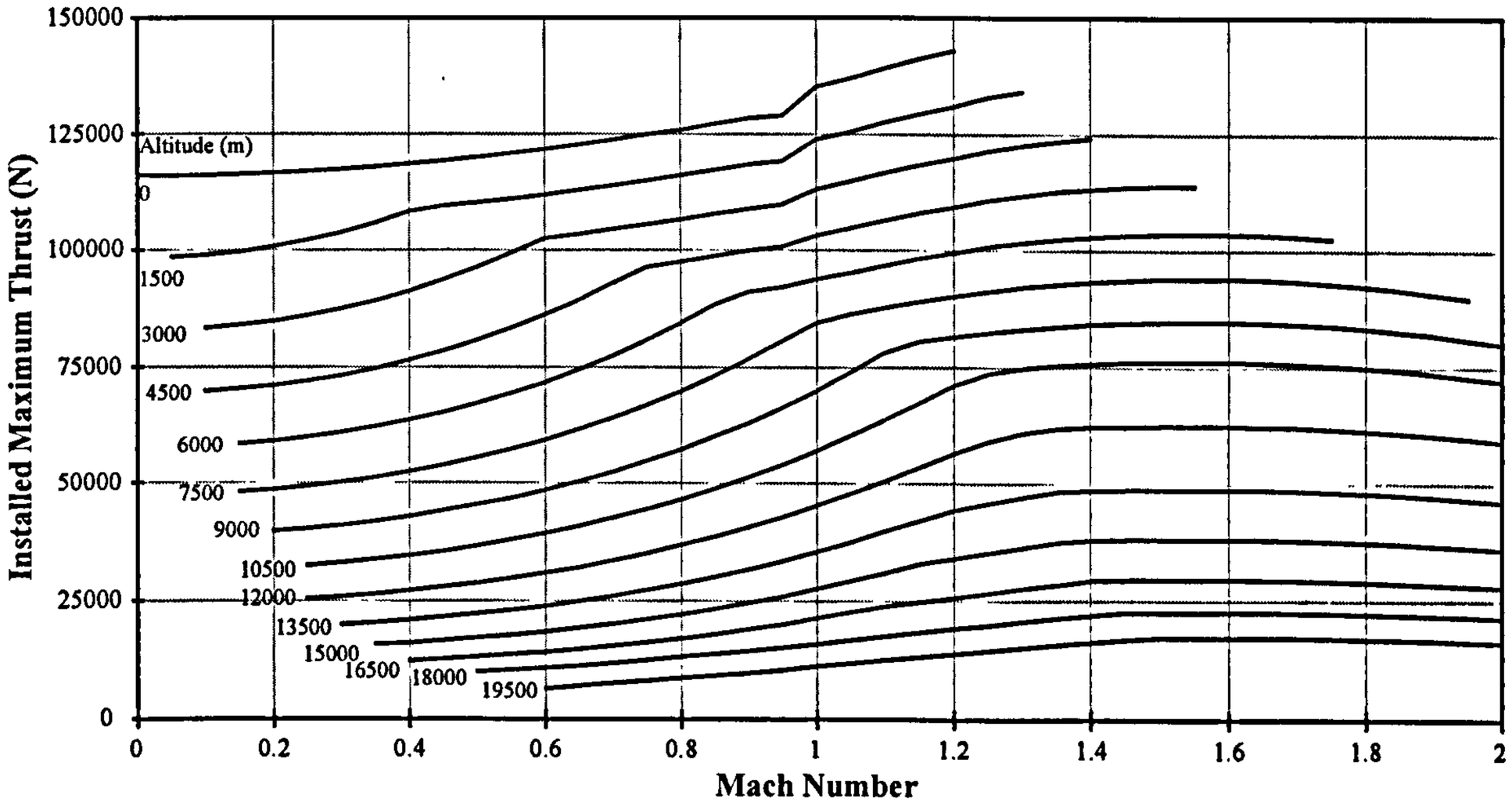


Figure 5-10. Example Installed Maximum Thrust.

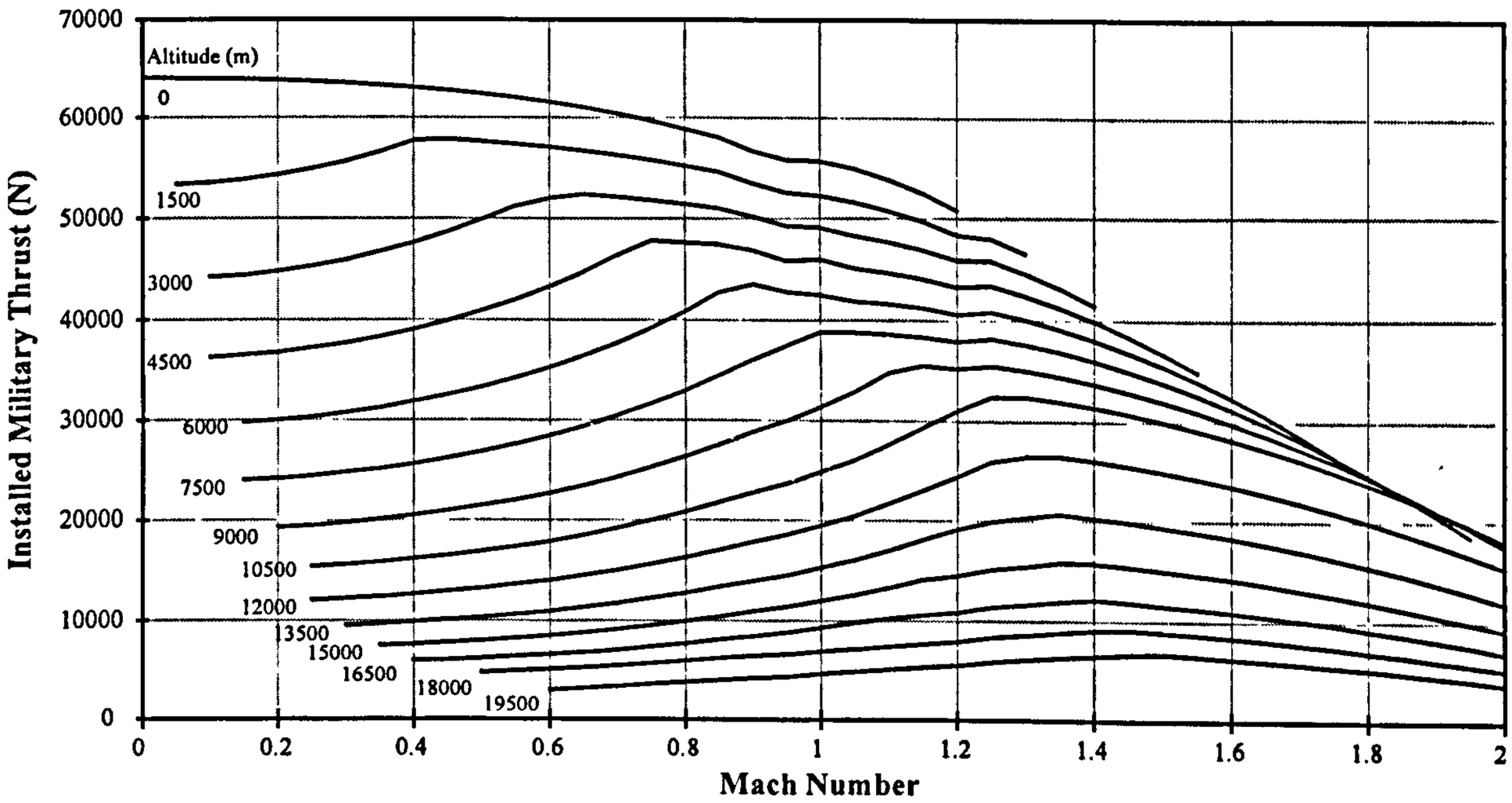
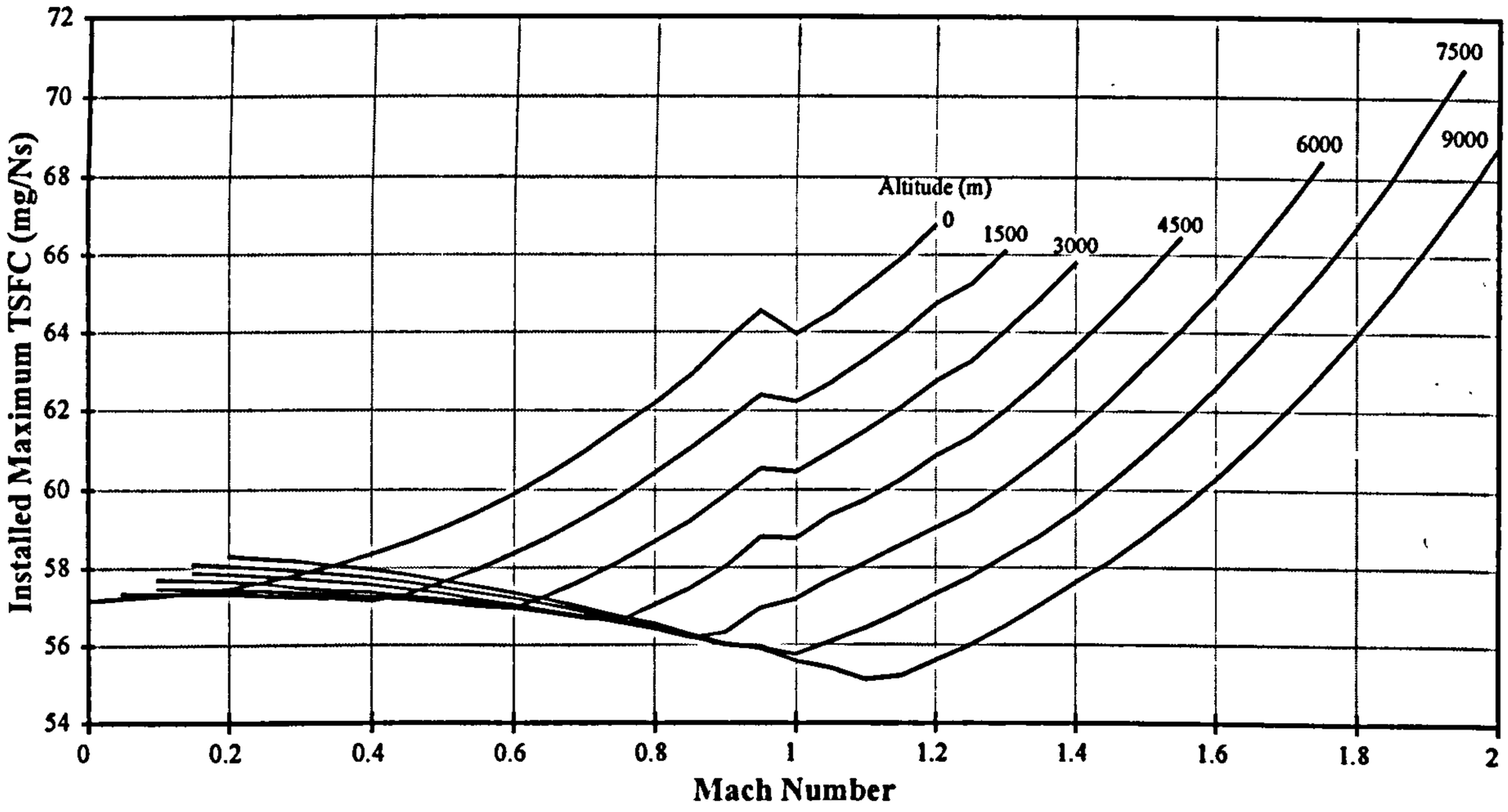


Figure 5-11. Example Installed Military Thrust.



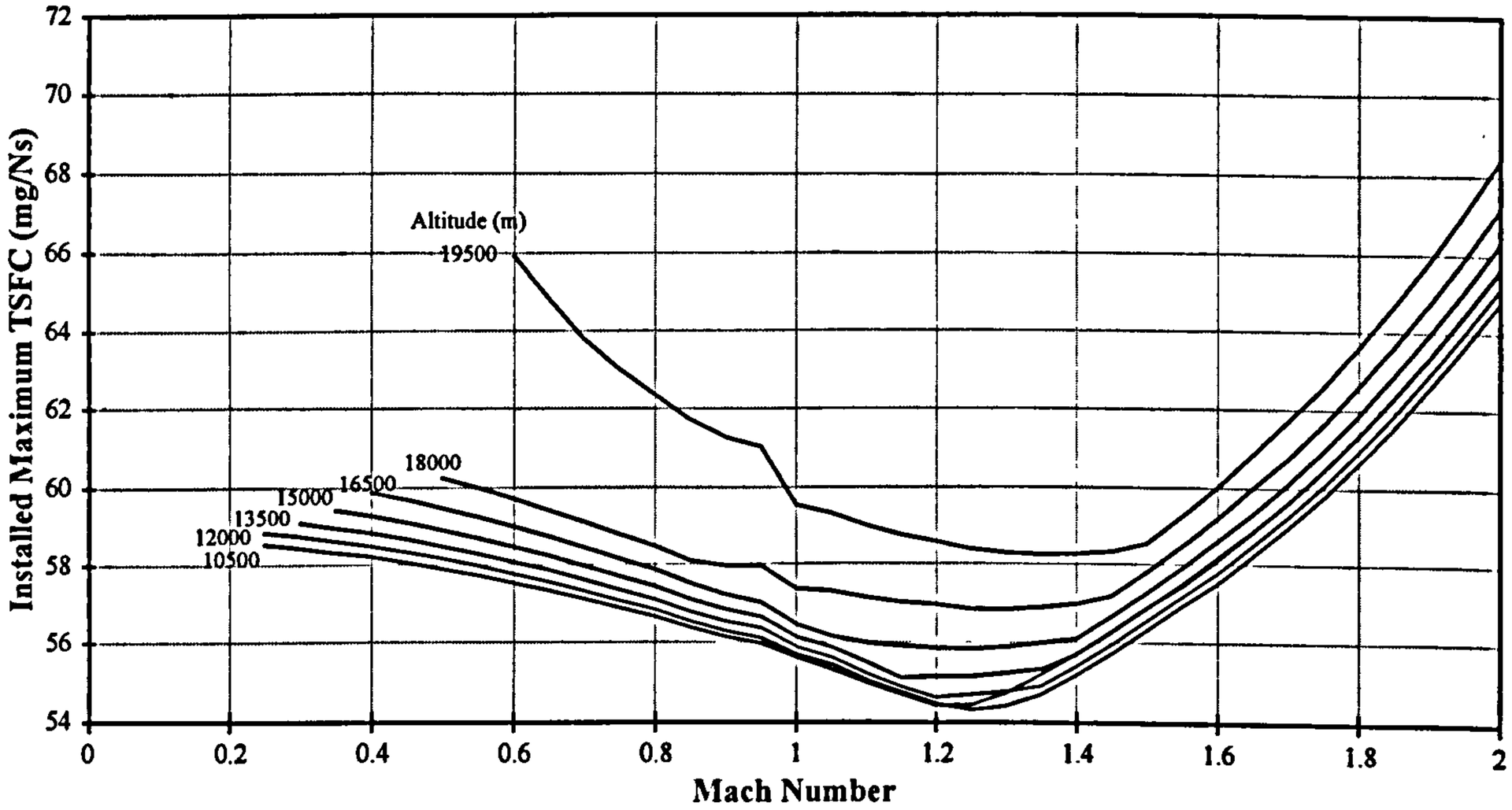


Figure 5-12. Example Installed TSFC - Maximum Thrust.

As can be seen from the figures, the combination of the thermodynamic and internal loss equations produces some complicated performance curves. These would be extremely difficult to produce using simpler methods, and prove the worth of including the ONX and OFFX thermodynamic models.

There are sudden jumps in values at, or near, M = 1 in may of the curves. These are caused by slight discrepancies between the internal loss models, and the fact that the intake characteristics will change significantly from subsonic to supersonic operation. The curves could have been smoothed slightly, but this was felt unnecessary, as it is unlikely that any aircraft would operate for significant lengths of time at this flight condition, i.e. M ≈ 1.0, at sea level.

5.2.4 Engine Dimensions and Mass Estimation

Once the engine air mass flow rate has been established for a particular application, and the design choices and limitations have been set, the mass and physical dimensions of the powerplant are calculated. Several existing methods available for aircraft engine sizing[4,36] were investigated, but it was discovered that the results were not at all consistent, and errors of more than 100% were experienced with new engines.

Continuing on a theme suggested by Whittle[103], a new engine dimension and mass estimation model has been successfully developed. The new methodology is based on the major engine design drivers; air mass flow rate, bypass ratio, compressor pressure ratio, number of shafts (although ONX/OFFX only deal with two-shaft engines), and reheat thrust increase. The database consists entirely of engines occurring on military aircraft, and a wide range of air mass flow rates and bypass ratios were captured to ensure that most engine types for future combat aircraft will lie within the extremes of the data. Many of the models investigated initially use engine thrust as a primary driver, which is affected by turbine inlet temperature and other technology factors. As these thermodynamic values have already been accounted for, the mass and external dimensions of the engine are determined by the engine design parameters.

Inlet diameter (DP1) is predicted from the mass flow rate, and is used by the volume estimation equations for the calculation of the intake duct volumes. Most engines will only provide optimum performance when the air flow at the compressor face is provided at approximately Mach 0.45, and leads to the fact that the engine inlet diameter (frontal area) is proportional to the sea-level static air mass flow rate (MPAD):

$$DP1 = \sqrt{\frac{MPAD}{131.5}} \text{ (m).} \tag{5-44}$$

Results for the 33 engines for which data was available, are shown in Figure 5-13. The best-fit line has a gradient of 1.0022 and a coefficient of determination of $R^2 = 0.9827$.

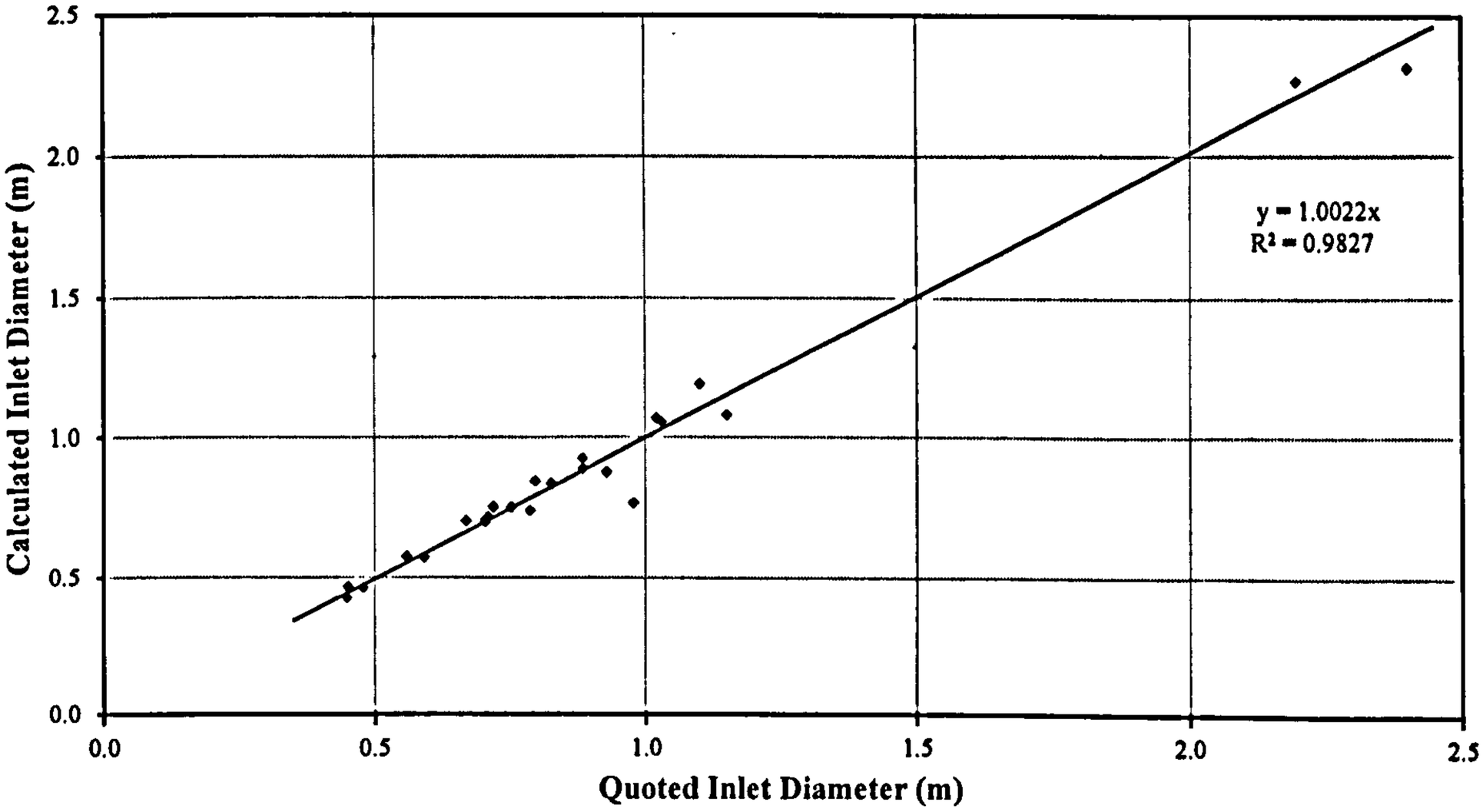


Figure 5-13. Engine Inlet Diameter Correlation Results.

The maximum diameter (DPH) is used in the engine bay volume and fuselage sizing calculations, and is a function of the air mass flow rate, bypass ratio (RPBY), and the number of spools in the engine (N_{SHAFT}):

$$DPH = \frac{0.1372 \times MPAD^{0.44} \times (1+RPBY)^{0.0025}}{\left(\frac{N_{SHAFT}}{2.9784}\right)^{0.072}} \text{ (m).} \qquad (5\text{-}45)$$

The expression for bypass ratio (1 + RPBY) is used so that the method is also applicable to pure turbojets, for which the bypass ratio is zero. The maximum diameter prediction equation was derived using data from 52 engines, as shown in Figure 5-14. The results are very good, with the best-fit line having a gradient of 0.9995, and a coefficient of determination of 0.9562.

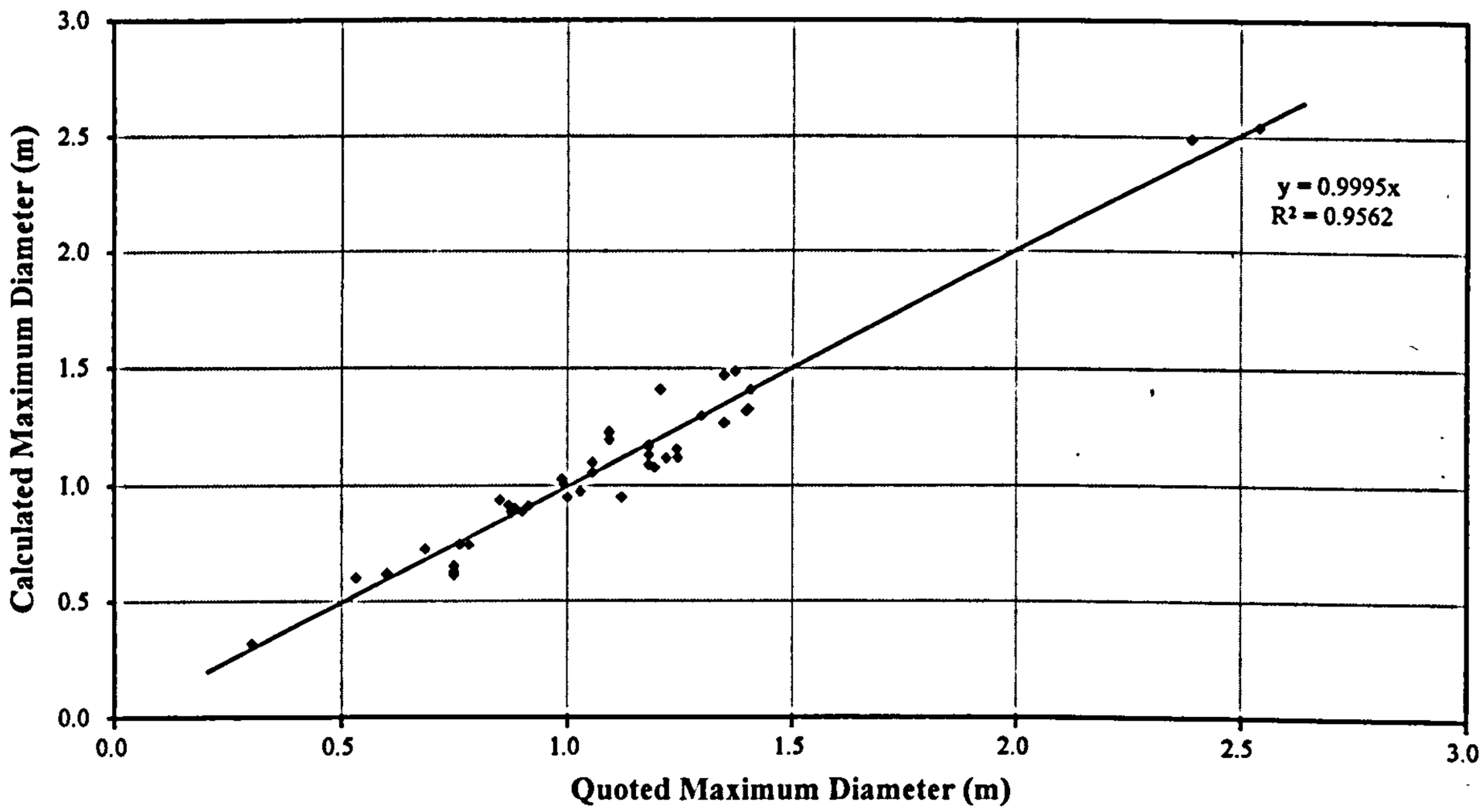


Figure 5-14. Engine Maximum Diameter Correlation Results.

Engine length (LPG) is calculated in two parts - one for the length of the bare engine, or for engines without reheat, and one for the length of the afterburner sections of engines having reheat capability. The bare engine length (LP12) is a function of the mass flow rate, bypass ratio, compressor pressure ratio (RPCH), and the number of shafts:

$$LP12 = \frac{0.4651 \times MPAD^{0.5272} \times 0.8764^{RPBY}}{RPCH^{0.1506}} \times 0.9223^{N_{SHAFT}} \text{ (m).} \qquad (5\text{-}46a)$$

For those engines having an afterburner, the length of the jet-pipe must be added to the bare engine length to produce the maximum engine length:

$$LPG = LP12 + \frac{0.2095 \times \left(\frac{TPGH}{TPGD}\right)^{0.0538} \times MPAD^{0.7284} \times 0.7588^{RPBY}}{RPCH^{0.2919}}, \qquad (5\text{-}46b)$$

where TPGH and TPGD are maximum and military sea-level static thrust, respectively.

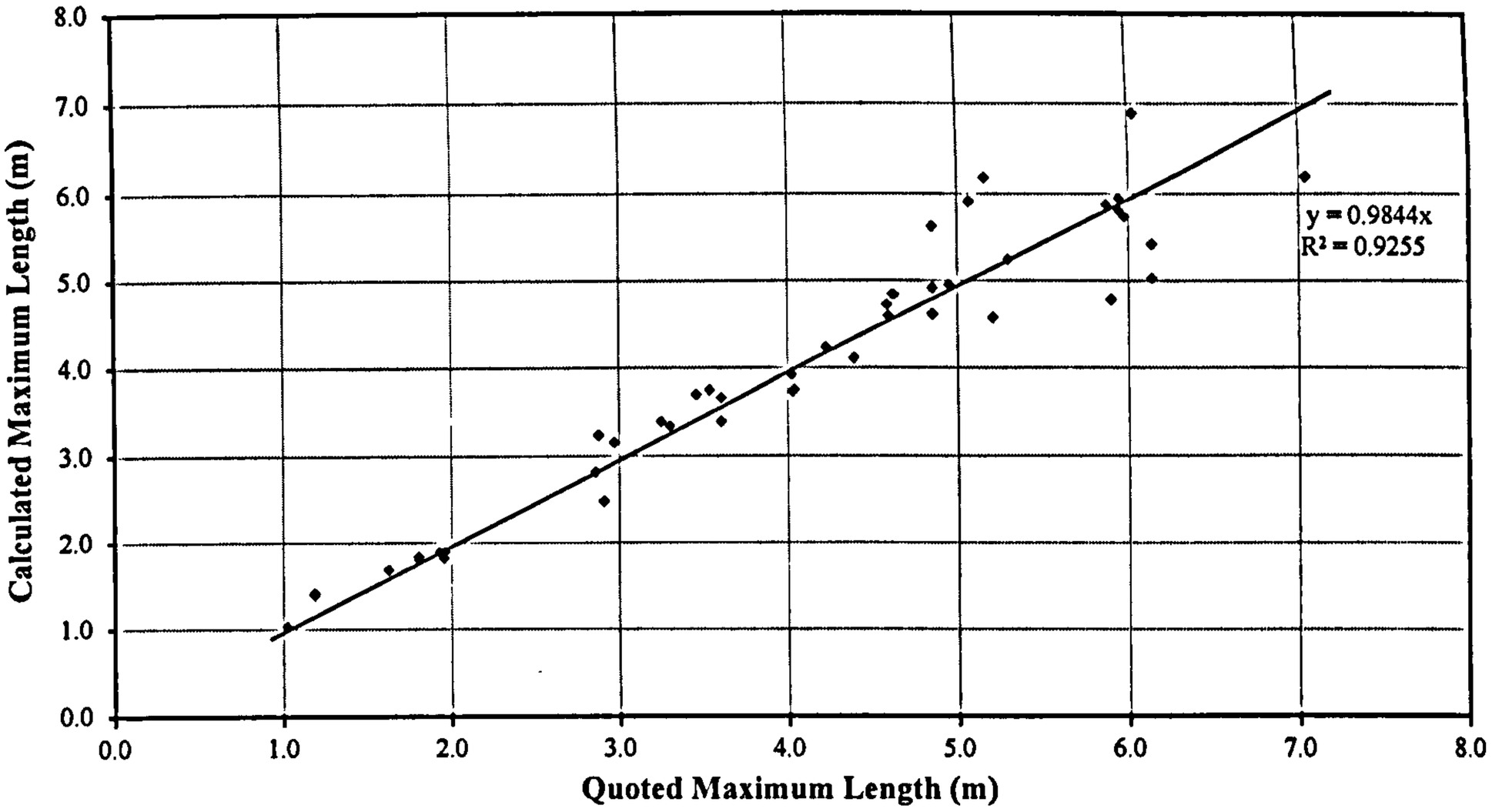


Figure 5-15. Engine Length Correlation Results.

The best-fit line of this correlation has a gradient of 0.9844, and a coefficient of determination of 0.9255, as shown in Figure 5-15. There is more scatter in this chart than the others, mainly as a result of the difference between the lengths of derivative engines, and the shortage of bare engine length data for engines with reheat. The equations were derived from data for 47 engines.

Engine mass (MPG) is generally taken to mean the mass of the engine including all engine-mounted ancillary and power off-take gearboxes, but without allowances made for the fuel and oil that will be required for the engine. Generally, this is so small as to be insignificant, and is taken as being included in the allowance for the powerplant installation mass in the overall mass calculations. As with the length calculations, the mass estimation is split into two parts; one that accounts for the mass of the bare engine, and another that calculates the mass of the reheat system, where applicable:

$$MPG = \frac{NENG \times 17.7638 \times MPAD^{0.9739} \times N_{SHAFT}^{0.2709}}{RPCH^{0.1385}\,(1+RPBY)^{0.4053}} \text{ (kg)}, \qquad (5\text{-}47a)$$

whilst there is an extra mass for the reheat system, making the total mass

$$MPG = MPG + \frac{NENG \times 0.3628 \times MPAD^{1.9796}}{\left(\frac{TPGH}{TPGD}\right)^{0.0021} \times (1+RPBY)^{0.8468} \times RPCH^{0.5264}} \text{ (kg)}. \qquad (5\text{-}47b)$$

These calculations have already appeared in section 4.3.2, but appear here as they were developed as part of the engine modelling procedure. The equations were developed using data for 53 engines; the results from the correlation are extremely good; the best-fit line has a gradient of 0.9801, and a coefficient of determination of 0.9741, as shown in Figure 5-16.

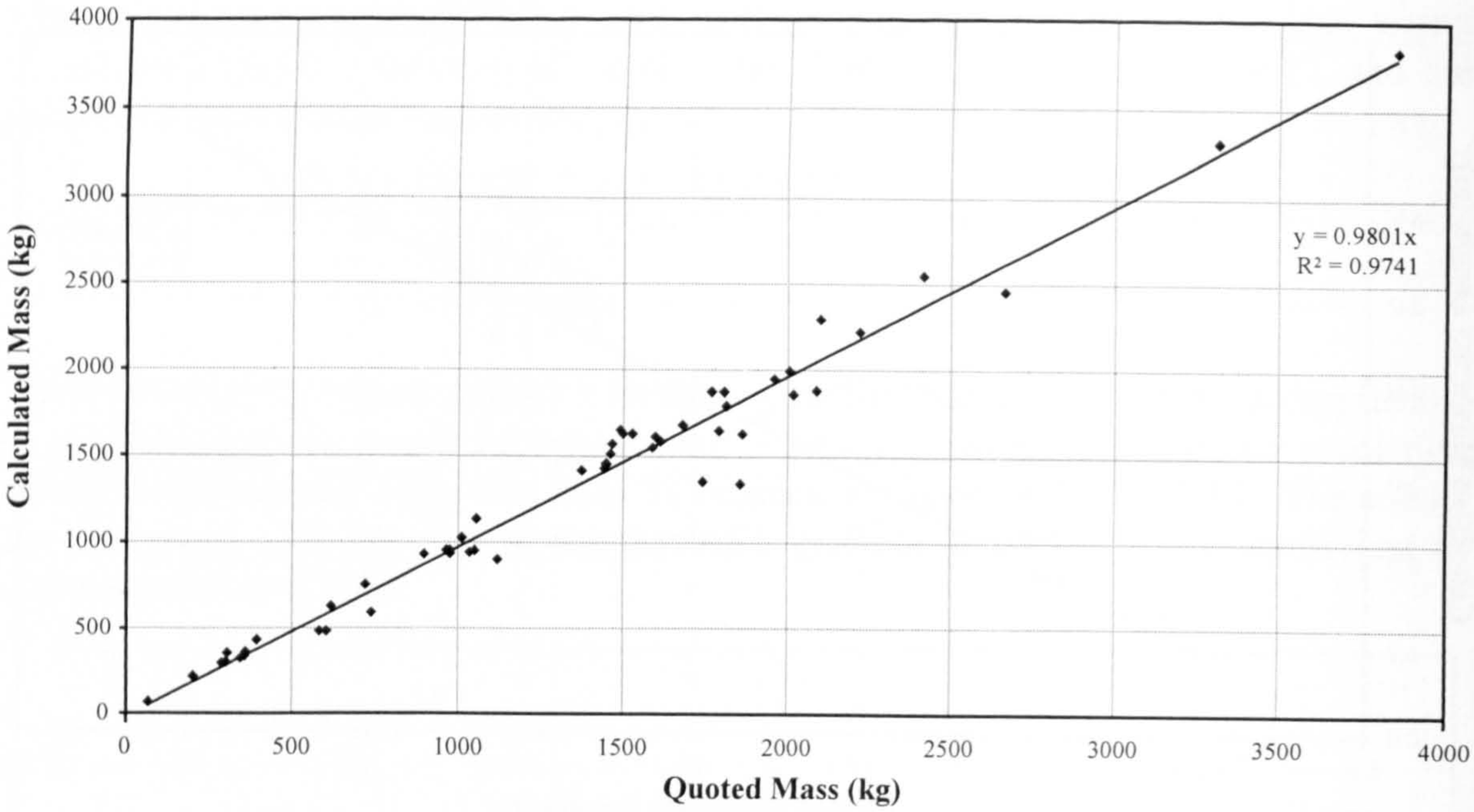


Figure 5-16. Engine Mass Correlation Results.

The results of the engine dimension and mass estimation equations have been combined to show the accuracy of the models. As can be seen, all the equations have very good prediction capabilities, with 50% or more of all predicted values within a 5% error band. The new engine parameter prediction equations considerably improve the accuracy of the engine modelling procedure, and allow the implementation of the ONX/OFFX thermodynamic models. To this end, they have proven to be an invaluable addition to the COMBAT aircraft design synthesis and optimisation model.

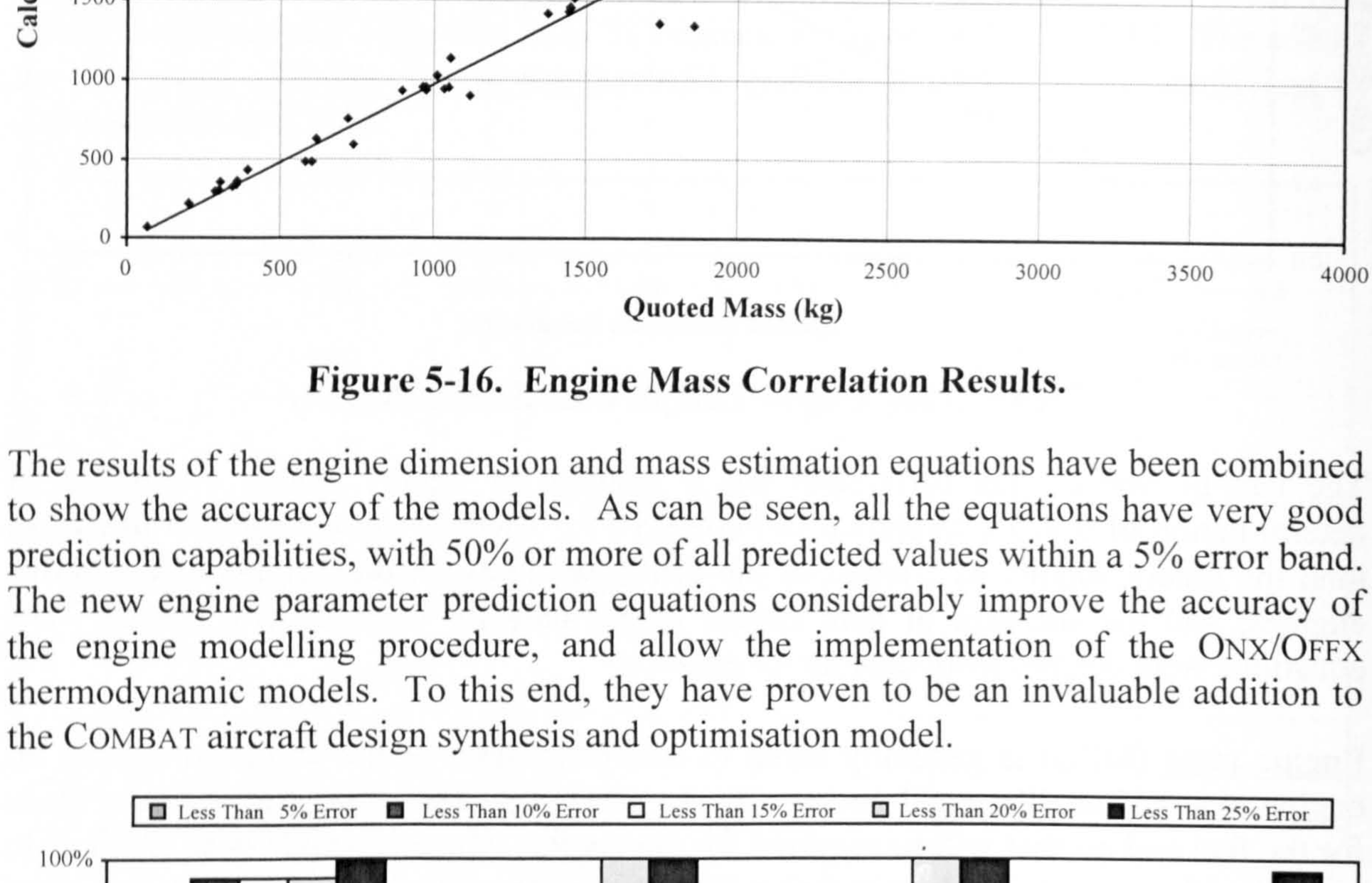


Figure 5-17. Percentage Error Bands for Engine Parameter Prediction Equations.

5.3 Summary

The Aerodynamics and Propulsion chapter has:

- Given a broad overview of the aerodynamic and propulsion modelling methods used within COMBAT.
- Explained the operation and theory behind the available lift model, together with all of the relevant equations and factors for most aircraft configurations.
- Given a thorough explanation of the angle-of-attack model, including most of the major equations. Where methods were complex and well documented in the original references, it has provided a clear description of the overall operation of the models.
- Illustrated the operation of drag model, including charts of the component build-up technique and a correlation of data for the YF-16.
- Explained the thermodynamic engine models in basic terms, but directed the reader to the references provided for greater detail. The modifications to the original models were detailed, and the method for the calculation of installation losses was also described.
- Provided details of the engine dimension and mass estimation algorithms developed for this synthesis model. Results of the predictions were shown individually, and in a group, to allow easy comparison of the individual models.

This page is intentionally blank.

6 Aircraft Point and Mission Performance

This chapter describes the methods and algorithms used to determine the performance characteristics of the designed aircraft. The performance calculations are split into two parts - those that deal with the specific performance requirements and help to size the aircraft, and those that are used in the estimation of mission performance and fuel burn. A wide range of performance calculations is available, so that a detailed set of requirements can be drawn up for any particular design study. The emphasis is mainly on flexibility, so that the constraint functions can be added in any order, in many ways, to fully describe the aircraft role.

The performance algorithms and equations used draw heavily on classical aircraft performance techniques, although some of the models for the point performance constraint analysis have been developed specifically for this study. The implementation of some of the traditional methods has been updated within iterative schemes to improve the accuracy of the calculations, and several of the distance and time legs have been split into smaller segments for the same reason. The final part of the performance calculation description gives a comparison of the instantaneous and sustained turn load factors for the F-16C.

The final part of the chapter gives a record of all of the constraints used in the COMBAT aircraft design and optimisation tool. These include design constraints, airframe sizing functions, aerodynamic limits, and the point performance calculations detailed earlier in the chapter. Where relevant, the reader is directed to the preceding chapters for a detailed description of the constraints and corresponding equations.

6.1 Point Performance Calculations

Point performance calculations are used to compare the delivered performance of the designed aircraft with the required performance figures. They play a crucial role in the sizing of the aircraft, as the performance constraints often determine the aircraft wing size and engine thrust levels. The sizes of the wing and engine have a major impact on the overall design of the aircraft, as seen in the previous chapters, and therefore the point performance calculations must be accurate, if a realistic design is to be produced.

COMBAT is able to consider up to ten different point performance constraints, using a system of codes for maximum flexibility. The requirements can be entered in any order, and the amount of fuel, payload, engine operation (maximum or military thrust), and the individual point performance level can be specified. The constraint codes and variable definitions are given in Table 6-1 and Table 6-2, respectively.

Phase	Code	Altitude	Mach	Data 1	Data 2	Data 3	Data 4	Data 5
Take-off	1	Altitude	0.0	0.0	Fuel	Payload	Reheat	S_{TOG}
Landing	2	Altitude	0.0	μ_B	Fuel	Payload	Arrest	S_{LAND}
Attained Turn	3	Altitude	Mach	1./2.	Fuel	Payload	0.0	N g's / °
Sustained Turn	4	Altitude	Mach	1./2.	Fuel	Payload	Reheat	N g's / °
Mach No:	5	Altitude	Mach	0.0	Fuel	Payload	Reheat	0.0
Accel./Climb	6	$Altitude_I$	$Mach_I$	Alt_F	$Mach_F$	Payload	Reheat	Time
Excess Power	7	Altitude	Mach	N g's	Fuel	Payload	Reheat	SEP
End/Max. Alt.	9	0.0	Mach	0.0	Fuel	Payload	Reheat	0.0

Table 6-1. COMBAT Point Performance Constraint Codes.

Variable	Explanation
Altitude	flight altitude for this constraint comparison. (m)
Mach	flight Mach number for performance comparison.
μ_B	braking coefficient of friction for landing phase.
N g's / °	load factor (g's) or turn rate (°/sec) for manoeuvre. (g's)
1. / 2.	option for either g's (1) or °/sec. (2) for attained and sustained turn calc's.
Fuel, Payload	proportion of original mission fuel or payload remaining during manoeuvre.
Reheat	allow use of reheat for specified manoeuvre. (0 - no, 1 - yes)
Arrest	efficiency of arresting device. (0.0 for none, 1.0 for high-efficiency reverser)
S_{TOG}, S_{LAND}	take-off or landing ground roll distance. (m)
Time	time in which climb/acceleration phase must be completed. (minutes)
SEP	amount of specific excess power required during manoeuvre. (m/s)

Table 6-2. Point Performance Constraint Variable Definitions.

The final calculation is an estimation of the maximum altitude of the aircraft for a given flight Mach number. The mass values at this condition are also used for the attained and sustained performance calculations at the end of the optimisation run, if required.

6.1.1 Take-Off Performance

The performance of an aircraft during take-off can be difficult to model, mainly because of the non-linearity in engine thrust and internal losses at very low speeds. To prevent this causing program instability, and to speed the process by avoiding an iterative calculation method, an existing empirical Cranfield method[46] was updated with new data to perform the take-off ground run calculation.

The method makes use of the 'un-stick', or take-off lift coefficient (CLTO), which is generated by the aerodynamics modules, the aircraft wing loading, and the uninstalled thrust/weight ratio. Take-off altitude, percentages of fuel and payload on board for the take-off run comparison, and whether the aircraft is allowed to use reheat (if applicable), can all be specified, as shown in Table 6-1. These are all taken into account in the calculation of the equation parameters. After simplifying, the equation becomes:

$$STOG = \frac{0.10437 \times MASS \times G}{SW \times CLTO \times \left(\frac{TPGH}{MASS \times G}\right)^{1.35}} \text{ (m).} \qquad (6\text{-}1)$$

where SW is wing area, TPGH is maximum uninstalled thrust, and STOG is the take-off ground roll, or the distance the aircraft travels before it achieves 'wheels-up'.

As with other empirical models, correlations between quoted and calculated values were generated with existing aircraft data, as shown in Figure 6-1.

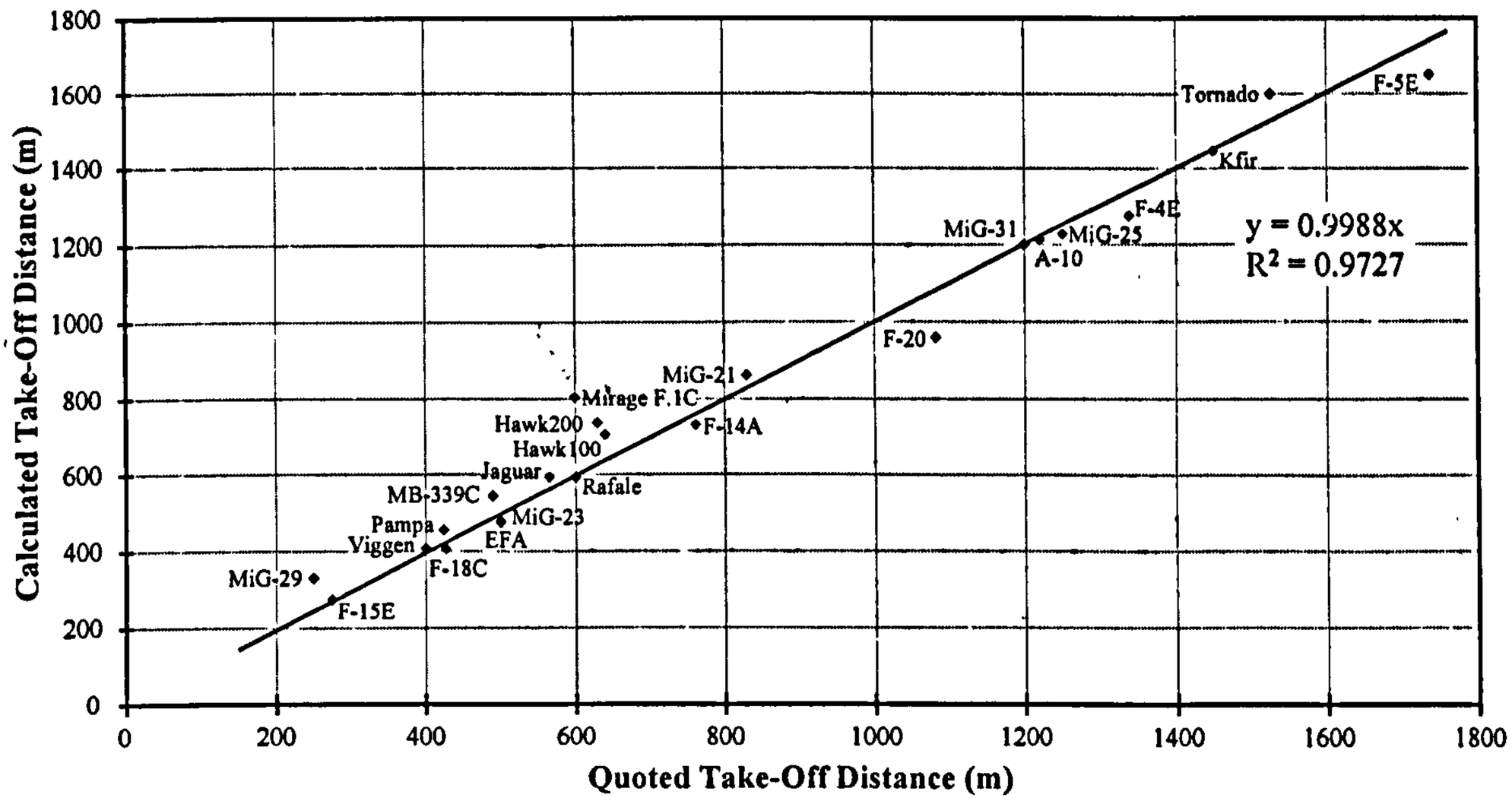


Figure 6-1. Take-Off Distance Correlation Results.

The accuracy of the method is very high, with a coefficient of determination of 0.9727, and a gradient of approximately 1. The additional benefit of the one-line algorithm used is that it circumvents problems with low-speed thrust non-linearity, resulting in a computationally efficient and highly accurate result for aircraft take-off run prediction.

RQPMIN compares the required (CONSTRAINT) and calculated take-off run (STOG) using an inequality constraint:

$$\text{FUNC} = \text{STOG} - \text{CONSTRAINT} \quad \text{(m)}. \tag{6-2}$$

6.1.2 Landing Performance

The landing performance method follows a similar method to that for take-off, being based on an up-dated Cranfield method, and using historical data to directly predict landing ground run. Landing altitude, percentage of fuel and payload on board during landing, braking coefficient, μ_B, and the arrestor efficiency can all be specified.

The braking coefficient depends mainly on the conditions on the runway surface, and is usually between 0.2 for wet runways, up to a maximum of 0.4 for modern anti-skid braking mechanisms on dry asphalt runways. Arrestor efficiency varies from 0.0 for aircraft having braking systems only, 0.3 - 0.4 should be used for aircraft with a drag-chute arresting system, to 1.0 for aircraft with efficient clamshell-type thrust reversers.

The modified equation for prediction of landing distance is:

$$\text{SLA} = \left(\frac{0.8521 \times \text{MASS}}{\rho \times \text{SW} \times \text{CLLA}}\right)\left(\frac{1.6}{\mu_B + \text{Arrest}} + 1\right) \quad \text{(m)}, \tag{6-3}$$

where μ_B and Arrest are the braking coefficient and the arrestor efficiencies, and CLLA is the landing lift coefficient. The RQPMIN constraint function is an inequality constraint and is calculated in the same way as shown in Equation 6-2. The results of the landing distance correlation are shown in Figure 6-2.

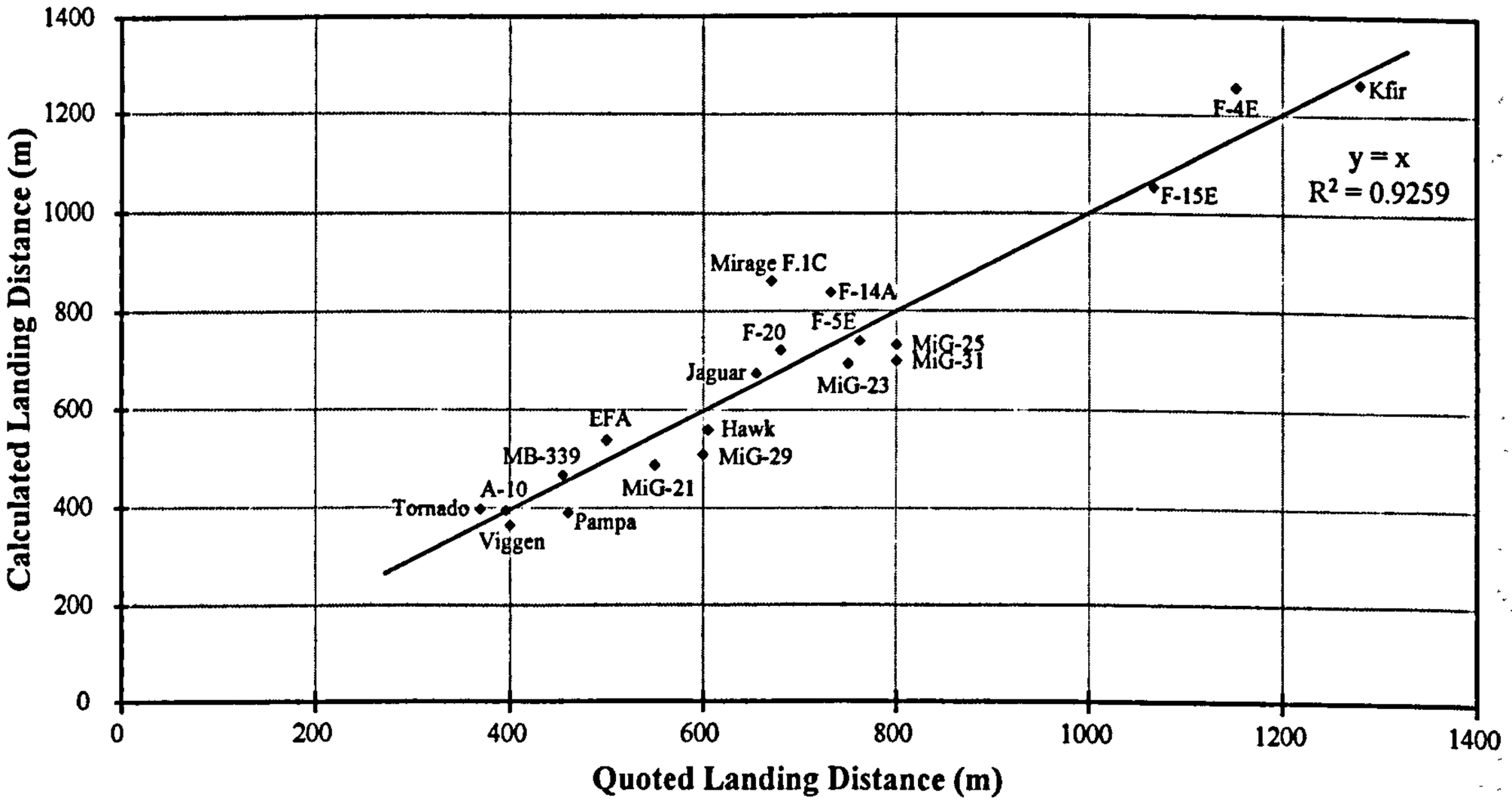


Figure 6-2. Landing Distance Correlation Results.

The results of the landing correlation are not as good as most of the others seen thus far, mainly due to the difficulties in capturing the braking coefficient and arrestor efficiency values. Obviously, the results could be improved by careful choice of both of these parameters for each aircraft, but this would not necessarily improve the method. The results are acceptable, and the gradient is exactly unity in this case, although the coefficient of determination (0.9252) is not as high as one would like.

6.1.3 Attained Turn Performance

Attained turn rate is also known as instantaneous turn rate, and is a measure of the aircraft turn performance based only on the flight condition, wing area, and the maximum lift coefficient available. The manoeuvre is a very important measure of the aircraft's overall survivability and lethality, even though it can result in the loss of a large amount of energy. The methods used are based on classical methods, which can be found in any aircraft performance or design text, but a full description of the models is given for completeness.

Once the air properties have been determined from the specified flight condition, and the mass calculated from the percentage fuel and payload inputs, the aerodynamics module is called. This returns the maximum available aircraft lift coefficient (CLHS), based on the methods detailed in section 5.1.1. If the required turn rate (GITR) has been given in °/second, it is converted to g's, as follows:

$$GITR = \sqrt{\left(\frac{CONSTRAINT \times RDN \times V}{G}\right)^2 + 1} \quad . \tag{6-4}$$

where V is flight velocity, G is gravitational acceleration, and RDN is the conversion to radians.

The attained turn rate is then calculated according to the aircraft configuration and Mach number. For Canard-Delta Aircraft at all Mach Numbers:

$$GITRC = \frac{\rho \times V^2 \times (SW + SETN) \times CLHS}{2 \times MASS \times G} \quad , \tag{6-5a}$$

where SW and SETN are the gross wing and exposed tail areas, respectively.

For all other aircraft up to and including Mach = 1.0:

$$GITRC = \frac{\rho \times V^2 \times SW \times CLHS}{(2 \times MASS \times G) \times \left(1.05^{(GITR-1)} + \sqrt{Mach} \times \left(1 - 1.05^{(GITR-1)}\right)\right)} \quad , \text{and} \tag{6-5b}$$

$$GITRC = \frac{\rho \times V^2 \times SW \times CLHS}{2 \times MASS \times G} \quad \text{above Mach} = 1.0. \tag{6-5c}$$

The extra factor, including the (GITR-1) and Mach terms, in Equation 6-5b is part of the lift model, but is added to the attained turn equation for ease of implementation. At lower load factors and speeds, less tail downforce is required to hold the aircraft at high incidence. This is caused by the increase in tail loads required to achieve high angles-of-attack, and therefore high lift coefficients, and the increase in aircraft stability (nose-down pitching moment) with increasing speed, due to the rearward shift in the centre of pressure.

It should also be noted that Equation 6-5a uses the sum of the gross wing area and the horizontal stabiliser (canard) areas, as canard-delta aircraft normally require an up-load on the canard to maintain the turn rate, which contributes to the overall aircraft lift.

At the end of the calculation, the values predicted must be compared with those specified to ensure sufficient turn performance. If the turn performance was specified in °/sec, the calculated g value (GITRC) must be converted back for the calculation of the RQPMIN inequality function:

$$\text{FUNC} = \text{CONSTRAINT} - \left[\frac{G \times \sqrt{GITRC^2 - 1}}{V \times RDN}\right] . \qquad (6\text{-}6)$$

The basic equations for the manoeuvre should be correct, as they are based on classical performance methods, and the factor above would seem to be an effective way of accounting for the tail down-loads in aft-tailed aircraft. Unfortunately, because of the significance of the turn rate values, real aircraft data for such manoeuvres is closely guarded, and a correlation of real and calculated turn rates for a range of aircraft cannot be presented.

There is one aircraft for which a significant amount of data has been released; the Lockheed (General Dynamics) F-16C. A comparison between the actual and calculated turning flight envelope for the F-16C is shown in section 6.2.10.

6.1.4 Sustained Turn Performance

Sustained turn rate is a measure of the ability of the aircraft to manoeuvre without losing either speed or altitude, i.e. without a drop in energy-height. It depends on many more parameters, including the drag coefficient at relatively high lift coefficient values, and on the maximum thrust of the engine(s). Consequently, there is greater room for errors to appear, and sources of those errors are harder to trace.

The air properties are determined from the specified flight condition, and the mass is calculated from the percentage fuel and payload inputs. If the turn rate has been given in °/second, the sustained turn rate is converted to g's using Equation 6-4. The maximum available engine thrust, and the total lift coefficient (CLCASE) are calculated by the relevant models. The method also calculates the wing-only lift coefficient (C_L), depending on the turn rate (GSTR) and the aircraft configuration.

For aircraft with conventional aft-situated tails, the total aircraft lift coefficient is calculated:

$$\text{CLCASE} = \frac{GSTR \times 2 \times MASS \times G}{\rho \times V^2 \times SW} . \qquad (6\text{-}7a)$$

For delta-wing aircraft, the total required lift coefficient is increased by the deflection of the elevons to maintain the turn, increasing the required wing-only lift coefficient due to the location of the elevons on the wing trailing edge:

$$\text{CLCASE} = \frac{1.1 \times GSTR \times 2 \times MASS \times G}{\rho \times V^2 \times SW} . \qquad (6\text{-}7b)$$

For delta-canard configurations, the sum of the gross wing area and exposed canard area is used for the calculation of the total aircraft lift coefficient:

$$CLCASE = \frac{GSTR \times 2 \times MASS \times G}{\rho \times V^2 \times (SW + SETN)} \quad . \tag{6-7c}$$

For conventional aircraft with leading-edge extensions (LEX), the wing lift coefficient is reduced slightly because of the strong effects of vortex lift on the relatively low-sweep wing. The wing C_L value is reduced so that:

$$CL = CLCASE - 2.075 \times SIN(QWL) \times SIN^2(ALPHA) \quad , \tag{6-8}$$

where QWL is the leading-edge sweep angle, and ALPHA is the angle of attack returned by the second aerodynamic module, detailed in section 5.1.2. This correction is not applied to ordinary conventional wings because they do not have the very high sweep angles necessary for the generation of useful vortex lift. The deflection of the elevons on delta-wing aircraft, and the favourable interference of the canard on the wing of the canard-delta are deemed to preclude the correction for these configurations.

The total drag coefficient is returned, based on the wing-alone lift coefficient, but the ratio of the total lift coefficient over the drag coefficient is used in the calculation of sustained turn rate:

$$GSTRC = \left(\frac{NENG \times Thrust}{MASS \times G}\right)\left(\frac{CLCASE}{CDCASE}\right) \quad . \tag{6-9}$$

As with all performance constraints, this is converted into an inequality constraint directly, or using a similar method to that in Equation 6-6 for sustained turn values given in °/second.

As stated earlier, the results of the sustained turn performance models will not be as accurate as the instantaneous turn rate output, because the number of models and subroutines called here is much larger. If the errors are assumed to be additive for each model, the sustained turn model will never be as good as the attained turn performance methods, because there are more models involved here.

As with the attained turn performance calculations, comparison of the models with known aircraft is difficult because the sensitivity of the data prevents it from entering the public domain. The exception to this is the F-16C, for which the sustained turn rate throughout the flight envelope, for a known mass, has been obtained. A comparison is shown in section 6.2.10.

6.1.5 Mach/Speed Performance

To calculate the maximum level flight speed of an aircraft at a given altitude and mass requires an iterative or graphical approach, because the thrust and drag models will be changing non-linearly with speed. This can lead to problems with the tolerance of the iteration convergence, especially in the constraint calculations of the optimisation procedure. In order to achieve an efficient and robust constraint, the actual constraint has been altered, so that it is not maximum speed, but rather excess thrust that becomes the optimiser inequality function.

The air properties are determined from the specified flight condition, and the mass is calculated from the % fuel and payload inputs, after which the total aircraft lift coefficient is determined, using the standard equation:

$$\text{CLCASE} = \frac{2 \times \text{MASS} \times \text{G}}{\rho \times \text{V}^2 \times \text{SW}} \quad . \tag{6-10}$$

This lift coefficient is supplied to the drag subroutine with the flight conditions and aircraft weapons loads, and the drag coefficient (CDCASE) is returned, from which the drag force (FDRAG) is calculated.

The maximum or military engine thrust is calculated, depending on the constraint, and the value is used with the drag force from above to form the RQPMIN inequality constraint:

$$\text{FUNC} = \frac{\text{FDRAG} - \text{NENG} \times \text{Thrust}}{1000} \quad \text{(kN)}. \tag{6-11}$$

6.1.6 Climb and Acceleration Performance

The climb and acceleration phases in COMBAT are always assumed to approximate a minimum-time climb/acceleration manoeuvre. The same method is also used in the mission performance calculations, as the fuel burn for a minimum-time climb/ acceleration and a minimum-fuel climb/acceleration are very similar. A climb or acceleration indicates that the aircraft is increasing its total energy-height from one value to another by generating more thrust than is required to overcome drag alone. The standard equation for the calculation of energy-height is:

$$\text{ZE} = \text{Altitude} + \frac{\text{V}^2}{2\,\text{G}} \quad \text{(m)}. \tag{6-12}$$

Thus, an increase in either speed, altitude, or both, without a corresponding drop in the other, constitutes a climb/acceleration phase. The constraint ensures that the aircraft can climb or accelerate from one altitude and Mach number to another within a specified amount of time.

The procedure used to approximate a minimum-time climb is split into three phases, the first being acceleration at constant altitude until maximum specific excess power is achieved. The second phase is a climb at constant equivalent airspeed until the final energy-height is achieved. At this point, the aircraft may be above and flying slower, or below and flying faster than the specified final flight condition. To reach that point, the aircraft performs a 'zoom' manoeuvre in the third phase, where it trades altitude with speed until its flight condition matches that specified.

During the constant acceleration phase, Phase 1, the thrust and the drag (FDRAG) are calculated as in previous sections. These are used in the calculation of the specific excess power (SEP), which is defined as:

$$\text{SEP} = \frac{\text{NENG} \times \text{Thrust} - \text{FDRAG}}{\text{FM} \times \text{MTT} \times \text{G}} \times \text{V} \quad \text{(m/s)}, \tag{6-13}$$

where FM is the current mass fraction, determined by the remaining fuel and payload.

This SEP is compared with the previous value, and if it is larger, the Mach number is incremented by Mach = 0.1 for all speeds, except 0.85 > Mach > 1.2, where the increment is Mach = 0.01. This process continues, with the values of thrust and drag being calculated at the mean of the start and end velocities, until the new value of SEP is not larger than the last one. For every increment, the time taken to perform the calculation is based on linear Newtonian motion:

$$\text{TIME} = \text{TIME} + \frac{\text{VF} - \text{VI}}{\text{ACCEL}} \text{ (seconds)}, \tag{6-14}$$

where the acceleration is also based on Newtonian motion:

$$\text{ACCEL} = \frac{\text{NENG} \times \text{Thrust} - \text{FDRAG}}{\text{FM} \times \text{MTT}} \text{ (m/s}^2\text{)}. \tag{6-15}$$

Once the maximum SEP has been reached, or the required energy height has been achieved, Phase 1 is terminated and Phase 2 begins.

Phase 2 is a climb at constant equivalent airspeed (constant dynamic pressure), until the final aircraft flight condition (energy-height) is reached. If the energy-height reached at the end of Phase 1 is the same as the final energy-height, this phase is skipped.

In order to determine the flight path the aircraft should follow, the dynamic pressure needs to be determined to ensure that the climb takes place at constant equivalent airspeed:

$$\text{VRO} = \text{VFinal} \times \sqrt{\frac{\rho}{1.225}} \text{ (m/s)}. \tag{6-16}$$

The altitude is incremented in steps (normally 500 m, although this can be easily changed), and at each increment the air parameters are updated with the mid-point values. Using a method suggested by Vinh[99], the aircraft climb angle (THETA), and the vertical component of the climb speed (VZ) are found, from which the time to climb can be calculated:

$$\text{VZ} = \left(\frac{\text{NENG} \times \text{Thrust} - \text{FDRAG}}{\text{FM} \times \text{MTT} \times \text{G}}\right) \times \frac{\text{V}}{\left(1 + \frac{1.225 \times \text{BETA} \times \text{VRO}^2}{2 \times \text{G} \times \rho}\right)} \text{ (m/s)}, \tag{6-17}$$

where FM is the mass fraction, V is the true flight speed, VRO is the speed for constant dynamic pressure at the new altitude, and BETA is a factor to account for the change in air density during the climb leg:

$$\text{BETA} = \frac{\rho_I - \rho_F}{1.225 \times \text{ALTINC}} \text{ (1/m)}, \tag{6-18}$$

and ρ_I, ρ and ρ_F are the air densities at the start, middle and end of the climb leg.

The aircraft climb angle is calculated from the ratio of the true flight speed and the vertical component of the flight speed:

$$\text{THETA} = \sin^{-1}\left(\frac{\text{VZ}}{\text{VM}}\right) \text{ (radians)}. \tag{6-19}$$

This angle is used to modify the values of available thrust, weight, and drag in Equation 6-17, upon which the true values of the climb angle and the vertical speed component will converge. VZ is used in the calculation of the time to climb the one leg:

$$\text{TIME} = \text{TIME} + \frac{\text{ALTF} - \text{ALTI}}{\text{VZ}} \quad \text{(seconds)}. \tag{6-20}$$

The process is repeated until the required energy-height has been achieved.

At every increment of energy-height, either speed, altitude, or both, the new mass fraction is calculated using a standard fuel-burn equation:

$$\text{FMF} = \text{FMI} \times e^{\frac{-\text{TSFC} \times \text{G} \times \text{DZE}}{\text{VM} \times \left(1 - \frac{\text{FDRAG}}{\text{Thrust} \times \text{NENG}}\right)}}, \tag{6-21}$$

where FMI and FMF are the initial and final mass fractions, TSFC is the thrust specific fuel consumption, DZE is the change in energy-height, and VM is the true airspeed at the mid-point of the climb/acceleration leg.

Because the mass has such a major impact on the performance of the aircraft, there is also an iteration to calculate the mass at the mid-point of the climb leg, and it is this value that is used in all the above calculations, except where stated otherwise.

Phase 3 is the final part of the overall climb manoeuvre, and models the 'zoom' to the final flight condition. To ensure that the drag is overcome during this final flight leg, ensuring that the final flight conditions are met, the mass at the end of Phase 2 is used to estimate the drag and thrust required at the end of the climb/acceleration. These values of thrust and TSFC are also used for the calculation of fuel burn during this final leg.

The same altitude increment as before is used, so that the zoom is done in small steps until the final conditions are reached. The time is calculated using a standard equation for non-linear motion:

$$\text{TIME} = \text{TIME} + \left(\frac{\text{ALTF} - \text{ALTI}}{\text{VF} - \text{VI}}\right) \times \text{LOG}\left(\frac{\text{VF}}{\text{VI}}\right) \quad \text{(seconds)}, \tag{6-22}$$

where VI and VF are the true airspeeds at the initial and final zoom step conditions.

The final mass fraction is calculated using the standard mass fraction calculation for a known thrust, TSFC, and time:

$$\text{FMF} = \text{FMI} - \left(\frac{\text{NENG} \times \text{Thrust} \times \text{TSFC} \times \Delta\text{TIME}}{\text{FMI} \times \text{MTT}}\right), \tag{6-23}$$

where all the parameters have their usual meanings, and ΔTIME is the time increment as calculated in the equation above.

The final operation in the climb/acceleration performance constraint analysis is the calculation of the RQPMIN inequality constraint. As all times in the input and output files are measured in minutes for ease of use, the final time is converted from seconds to minutes:

$$\text{FUNC} = \frac{\text{TIME}}{60} - \text{CONSTRAINT} \quad \text{(minutes)}. \tag{6-24}$$

As with the attained and sustained turn performance data, climb and acceleration data from one specified condition to another is very rarely seen. Consequently, it has not been possible to present a correlation of the accuracy of the method. However, because the methods used are based on theoretical performance procedures, and the climb/acceleration is broken down into many small legs, the accuracy should be good. Accuracy may be reduced slightly by the assumption of constant acceleration across the legs, and by the number of different models used, but these should be minor discrepancies compared to the overall times involved.

6.1.7 Specific Excess Power Constraint

Specific Excess Power represents the ability of an aircraft to accelerate and/or climb; when SEP is equal to zero, the thrust is equal to the drag, and the aircraft is therefore in a steady-state condition. This constraint allows the user to specify that the aircraft should have a level of performance over and above that required to achieve a certain performance constraint.

The constraint for specific excess power follows exactly the same format as that for sustained turn, with the constraint now being the specific excess power instead of the load•factor (g). After the air properties and masses have been determined, the lift coefficient is calculated using the methods presented in Equations 6-7a, b, and c. The drag force is calculated from this value, and the SEP is calculated using Equation 6-13. Finally, the inequality constraint is produced in the normal way.

6.1.8 Calculation of Maximum Altitude

The final section of the point performance calculations is the estimation of the aircraft maximum altitude. This has not been included as a performance constraint, as the value is required by the life cycle cost module, and constraints are not always specified by the user. Instead, the maximum altitude is used as a 'reality check', and to feed into to the maintenance effort prediction section of the Life Cycle Cost model.

The method is based on the calculation of SEP, and the altitude is altered at the Mach number and mass condition specified by the user at increasing altitudes, until the altitude at which zero SEP is achieved. After the initialisation of the air and mass parameters, the lift and drag coefficients are determined. The thrust and drag values are calculated, and the SEP is found from Equation 6-13. A second value of SEP is found at an altitude 100m above the first, and the altitude where SEP would be zero is estimated:

$$ALT(1) = \frac{-\left(PS(1) - \left(\frac{PS(2) - PS(1)}{100}\right) \times ALT(1)\right)}{\left(\frac{PS(2) - PS(1)}{100}\right)} \text{ (m)}, \qquad (6\text{-}25)$$

where ALT() and PS() are arrays for the two altitude and SEP values.

The new altitude value is used for the calculation of the new values of thrust and drag, and if the two SEP values agree within a small tolerance, the iteration stops and the maximum altitude is defined as the new value of ALT(1).

6.2 Mission Performance Calculations

The mission performance calculations work, for the most part, in a similar manner to the point performance constraint analysis methods, many of the equations being identical. The main difference in this section is that the major factor being calculated is the amount of fuel burnt for each mission leg. The sum of all of these masses, plus a user-defined reserve factor, gives the total mission fuel mass, one of the single most important values in the sizing of the aircraft, from both a mass and volume perspective.

The mission requirement variables operate in the same way as the point performance constraints, except that the mission codes now correspond with mission legs, rather than design constraint manoeuvres. Mission codes, and explanations differing from those for the point performance variables, are given in Table 6-3 and Table 6-4, respectively.

Phase	Code	Altitude	Mach	Data 1	Data 2	Reheat
Engine run	1	Altitude	0.0	Time	0.0	Reheat
Take Off	2	Altitude	0.0	μ_N	V_{TO}/V_{STALL}	Reheat
Climb/Accel/Desc	3	Altitude $_F$	Mach $_F$	0.0	0.0	Reheat
Cruise	4	Altitude	Mach	Distance	Optimise▲	Reheat
Manoeuvres	5	Altitude	Mach	Time / No: °	0.0 / N g's	Reheat
Weapons Drop	6	Altitude	Mach	Weapons	External★	Reheat
Loiter/CAP	7	Altitude	Mach	Time	0.0	Reheat
Landing	8	Altitude	0.0	0.0	0.0	0.0
End Mission	9	0.0	0.0	Fuel Reserve	0.0	0.0

Table 6-3. Mission Performance Codes

Variable	Explanation
Time	duration of mission segment. (minutes)
μ_N	rolling friction coefficient for take-off (brakes off).
Distance	distance travelled during cruise leg. (km)
Optimise▲	allow optimiser to control cruise Mach number. (0.0 - off, 1.0 - on)
Time	time for which combat should continue (minutes). Maximum of 20.0
No: °	total number of degrees to be turned through for manoeuvre leg. (°)
% Weapons	fraction of weaponry mass to be dropped / fired. e.g. 0.25
External★	weapons dropped from external pylons. (1.0 - External, 0.0 - Internal)
Fuel Reserve	the fraction of extra mission fuel to be carried, after loiter, etc. e.g. 0.05

Table 6-4. Mission Variable Explanations

▲ Optimisation of cruise Mach number can be used for main outbound & return legs.

★ External weapons release will result in reduced mass and stores drag, internal weapons release reduces aircraft mass only.

6.2.1 Engine Start, Warm-up, Taxi, etc.

This mission phase allows the simulation of a ground run of the engine at either military or maximum power for the purposes of warm-up, taxi, etc. Although the aircraft would rarely use full power on the ground, except for take-off, this is still a useful mission phase, provided the times are kept to reasonable values. A twenty-minute engine warm-up at engine idle may be equivalent to a one-minute run at military thrust.

After the air parameters have been initialised, either maximum or military thrust is requested from the engine modules, and the total aircraft mass at the end of the mission phase is found:

$$\text{Mass}_F = \text{Mass}_I - (\text{NENG} \times \text{Thrust} \times \text{TSFC} \times \text{Time} \times 60) \quad \text{(kg)}. \tag{6-26}$$

The results of this calculation should be very accurate, due to the simplicity of the algorithm used. The only errors that can be introduced come from the engine modules, which are relatively accurate models.

6.2.2 Take-Off

The take-off model in the mission performance modules is significantly different to that used for the point performance analysis. Whereas the point performance model is based on a single empirical expression, the fuel mass used is found by dividing the take-off run into ten equal velocity-increment sectors, and summing the fuel burned during each one.

The final velocity for take-off (VTO) is calculated from the mass prior to the start of the take-off run, and a factor for the ratio of the take-off velocity to the stall speed, typically in the region of 1.2:

$$\text{VTO} = (V_{TO} / V_{STALL}) \times \sqrt{\frac{2 \times \text{Mass}_I \times G}{\rho \times SW \times CLTO}} \quad \text{(m/s)}. \tag{6-27}$$

For each velocity increment, the mid-point is used for all calculations, and the thrust, drag, and mass are re-calculated every time. The aircraft drag is calculated in the normal way, except that there is now a small extra drag component to allow for the drag of the wheels and flaps:

$$\text{FDRAG} = 0.5 \times \rho \times V_M^{\,2} \times SW \times \text{CDCASE} + \mu_N \times \text{Mass}_M \times G \quad \text{(N)}. \tag{6-28}$$

where μ_N is the coefficient of friction, typically about 0.05, and the subscript M is used to depict the mid-point velocity/mass of the individual velocity increment.

The acceleration of the aircraft is treated as linear across each increment, and the time and fuel burn are calculated in the normal way. Once the aircraft has reached the take-off speed, there is said to be a three-second delay for rotation and wheels-up, by which time the aircraft is said to have reached an altitude of 15 m. The fuel burn for this small section is also calculated using the method shown in Equation 6-26. Thus, at the end of the take-off phase, the aircraft is said to be flying at a velocity of just over the user-specified stall speed ratio (due to the drop in fuel mass during the take-off run), at an altitude of 15 m.

The accuracy of the method should be quite high, but there will be slight discrepancies introduced by the highly non-linear variation of thrust with speed at the very low velocities near the start of take-off. This is unavoidable using this type of method, which is felt to be superior to the very simple methods of accounting for take-off fuel in references such as Raymer[78]. That said, the results should certainly be within a few percent of the true values, and should give the correct trends for changes in the aircraft design, which is most important. Design changes would not be reflected by the simple take-off fuel fraction values suggested by Raymer and others.

6.2.3 Change in Energy-Height

This model is almost identical to that used for the point performance module, that being one of the few models in the point performance section that calculates the fuel burn during the manoeuvre. One of the few differences with this model is that it must first determine whether any change in energy-height is a climb/acceleration or a descent/ deceleration. It does this by comparing the energy-height at the start of the manoeuvre with that of the final conditions and simply switching between the two options.

If the manoeuvre proves to be a descent/deceleration, the time taken is calculated using a single implementation of Equation 6-22, but the mass and distance covered are unchanged. If the manoeuvre is a climb/acceleration, the same 3-Phase method is used as for the constraint analysis. As the methods are so similar, the reader is directed to section 6.1.6 for a full description of the module and related equations.

The only other difference between the methods of the constraint and mission performance modules is the lack of the inequality constraint function, but this is replaced with the equations for the updating the mission time, aircraft mass, and distance covered (zero - no range credit).

6.2.4 Cruise

This module calculates the fuel burned during sustained straight-and-level flight at near-constant conditions. As this mission phase usually results in the largest amount of fuel burn, the methods used are based on an iterative implementation of the classical Breguet range equations. The cruise phase is broken down into a number of segments, typically 50 km in length, the fuel, time and distance values being summed to give the totals.

Once the air and mass values have been initialised by the atmosphere model, the lift, drag and TSFC are determined, and a first guess of the cruise leg fuel burn is made. After this first run, the mass, lift, drag, and engine parameters are determined at the mid-point of the range increment (ΔDIST), and the end mass fraction is re-calculated:

$$FMF = FMI \times e^{\left(\frac{-\Delta DIST \times 1000 \times TSFC \times G \times CDCASE}{V \times CLCASE}\right)} \quad . \tag{6-29}$$

This process is repeated twice, to ensure a good level of convergence without incurring severe run-time penalties, the leg fuel mass normally converging to within 2%. The method deliberately avoids placing a convergence tolerance on this iteration, as a small change in the aircraft design can alter the number of iterations that take place, leading to problems with step changes in the optimiser function analysis.

There is a procedure, performed during all relevant mission phases, to check the ratio of available to required thrust (RTPG). If there is insufficient thrust available to perform the specified mission phase, the program continues as if there were, but an RQPMIN inequality constraint would have a value greater than 0.0, violating the constraint. This is of particular interest in this mission phase, as a 'supercruise' can be specified at any time during the flight by disallowing the use of reheat, and specifying a supersonic cruise Mach number.

The user also has the option to let RQPMIN optimise the cruise Mach number during the sizing study. The method outlined above is unaffected, the cruise Mach number being set by the optimiser and not the input file. The final Mach number of the mission phase just prior to the cruise phase is also changed; the optimisation now becomes a pseudo-mission optimisation study, which is a very powerful option. For instance, if the previous mission phase is a climb/acceleration, the optimiser controls the energy height manoeuvre end Mach number and the cruise start Mach number, and determines the optimum point to achieve minimum fuel burn.

6.2.5 Manoeuvres

The manoeuvre phase allows the user to specify the altitude and Mach number at which the combat phase will take place. Either the time of combat, or the number of degrees to be turned through can be specified, using the current configuration and mass. If the total turn angle is set, the turn can be performed either at a specified load factor (g's), or at the maximum achievable g level. As with all mission phases, except landing, the use of reheat can be allowed or disallowed.

The air and mass properties are set, and if the turn rate is zero, it is set to an arbitrary value for the first iteration. The engine routines are called with a request for maximum or military thrust, and the total aircraft lift coefficient and wing lift coefficient are calculated using Equations 6-7a, b, and c. The wing lift coefficient is adjusted for conventional-LEX aircraft using Equation 6-8, the drag coefficients are calculated by the aerodynamics modules, and the sustained turn rate is calculated using Equation 6-9. If the difference between the calculated turn rate and the initial guess turn rate is greater than a nominal value, the guess is updated as the mean of the guess and calculated values, and the process is repeated.

The sustained turn rate cannot be larger than the design limit load; if this is found to be the case, the turn rate is set to the design limit load, GND, the lift and drag coefficients are determined, and the sustained turn rate is re-calculated. If it is found to still be larger than the design limit load, the turn rate is set to the limit, and the engine routines are called with a reduced thrust level, so as not to exceed the design limit load factor.

If the value of the turn rate has been specified, the procedure is somewhat simpler. The lift and drag coefficients are determined using the specified value, and the drag force for the turn is calculated. The engine routines are asked to supply a thrust equal to the drag force, and the corresponding TSFC. It is assumed that the available thrust is large enough to meet the requirement; the required/delivered thrust ratio, RTPG, is updated. This is a very powerful attribute, as the turn rate now acts as an extra constraint, one for which the exact fuel load is not known at the start of the analysis.

The final stage in the calculation of the manoeuvre fuel mass is to determine the time of the whole mission phase. If the total value of mission data value 1 is less than 20.0, this is taken to be a time for the mission phase, and the fuel mass is simply calculated using Equation 6-26. If the value is greater than 20.0, the value is treated as the total angle through which the aircraft must turn during the mission phase. The time is calculated from the total turn angle using the following standard equation:

$$\text{TIME} = \frac{\text{Angle} \times \text{RDN} \times \text{V}}{\text{G} \times \sqrt{\text{GSTR} - 1}} \quad \text{(seconds)}. \tag{6-30}$$

Finally, the mission time, mass, and distance values are updated, although distance is said not to have increased during the manoeuvre phase.

6.2.6 Weapons Drop

The weapons drop mission phase specifies the dropping/firing of a fraction of the total weapons mass, and whether external or internal weapons should be released. If internal weapons are specified, the mass of the aircraft will reduce, but the drag coefficient values remain unchanged. If the internal weapons drop specified is greater than the mass of internal weapons remaining, the difference is subtracted from the external weaponry, and the aircraft mass and drag coefficient are altered, as if an external weapon drop had been designated.

Time, mass, and distance values are incremented by 0.1 minutes, -Fraction×MB (kg), and 0 (km), respectively.

6.2.7 Loiter / Combat Air Patrol

This phase allows the user to specify the duration that the aircraft should hold at one particular altitude and Mach number, normally at the end of the mission, before landing, or as part of a Combat Air Patrol (CAP) sortie. The procedure is similar to that for the cruise phase, except that the iteration on final mass is only performed once for the entire duration of the loiter/CAP.

After the air and mass parameters have been set, the lift and drag forces are determined and the engine routines are called with a request for a thrust equal to the drag force. The thrust and TSFC are returned, and the loiter/CAP final mass fraction is calculated:

$$\text{FMF} = \text{FMI} \times e^{\left(\frac{-\text{TIME} \times \text{TSFC} \times \text{G} \times \text{CDCASE}}{\text{CLCASE}}\right)}. \tag{6-31}$$

The iteration is performed twice more, with all parameters being re-calculated from the mass at the mid-point of the mission phase. As with the cruise phase, this allows the accuracy of the calculation to be improved, without reducing the robustness of the optimiser function analysis.

The method should be as accurate as any other in the mission analysis module, as it uses classical performance methods with the drag and propulsion models described in sections 5.1.3 and 5.2. The model does not contain an error trap to prevent the aircraft from flying below the buffet speed; care must be taken not to specify loiter Mach numbers which may result in C_L values of about 0.4 or higher.

6.2.8 Landing

The value of the mass of fuel burned during the landing phase is difficult to estimate, as each aircraft will have a different landing sequence, and may or may not use reverse thrust. As the results do not change dramatically from one aircraft to the next, a fuel mass fraction has been applied to reduce aircraft mass by 0.5% during the landing phase.

The time taken for the landing sequence has been estimated by calculating the landing speed, assuming a velocity/stall speed ratio of 1.15, and the drag on the aircraft at that point, and assuming a linear deceleration to rest. This is not strictly accurate, but as this does not have a major impact on the rest of the design, it was felt adequate. The time, mass, and horizontal distance variables are updated, completing the mission analysis for the candidate aircraft design.

6.2.9 Mission Termination

To end the mission calculations, a mission code of 9 is used in the aircraft input file. The total mission fuel mass is calculated, using a supplied value for the fraction of reserve fuel to be carried, typically 5%. The total previous mass of fuel is incremented by the reserve factor, and the new total is compared with the total mission fuel fraction. The mission fuel volume is also calculated, assuming a standard density of 779 kg/m^3, which is compared with the total available fuel volume in both internal and external tanks, if available.

6.2.10 Attained & Sustained Turn Data

If specified by the user in the aircraft input file, two other files containing data for the turn performance and the drag coefficient for the designed aircraft will be produced. As the drag model was described in the previous chapter, it is not detailed here. The attained and sustained turn data file contains the data in two sections for attained and sustained turn load factor, from M = 0 to the maximum Mach number, MMAX.

The attained turn data is produced using the equations in section 6.1.3, and contains altitude and Mach number values for load factors from 1 to the design limit load factor (GND) in steps of two. The sea level Mach numbers at which the calculated load factor equals the next load factor to be found is calculated. The Mach number is incremented from the first value and, for each load factor in turn, the altitude at which the manoeuvre must be performed to deliver the specified load factor is calculated.

The method makes use of a modified Newton-Raphson search technique to calculate first the Mach number, and then the altitudes. Once the altitudes for all of the load factors have been calculated, the results are written to the file. A sample output file is shown in Figure 6-3, and shows the comparison of the calculated and quoted attained turn rates (from Jane's All the World's Aircraft[98] 1995-96) for the F-16C aircraft with half internal fuel and two wing-tip Sidewinder missiles.

The results of the comparison are seen to be excellent for the majority of the flight envelope, but this should be the case, as the lift models described in the previous section were calibrated with this data. The change in turn rate with increasing load factor is also very encouraging, and the discrepancy in turn rates at subsonic speeds for the g = 9 case

is probably due to angle-of-attack limiting in the aircraft flight control system at higher g levels. The over-prediction of altitude for the g = 1 case at supersonic speeds is also probably due to some kind of angle-of attack/speed limitation device in the flight control system. As these two last phenomena will be aircraft-specific, no attempt has been made to include them into the lift and/or attained turn models.

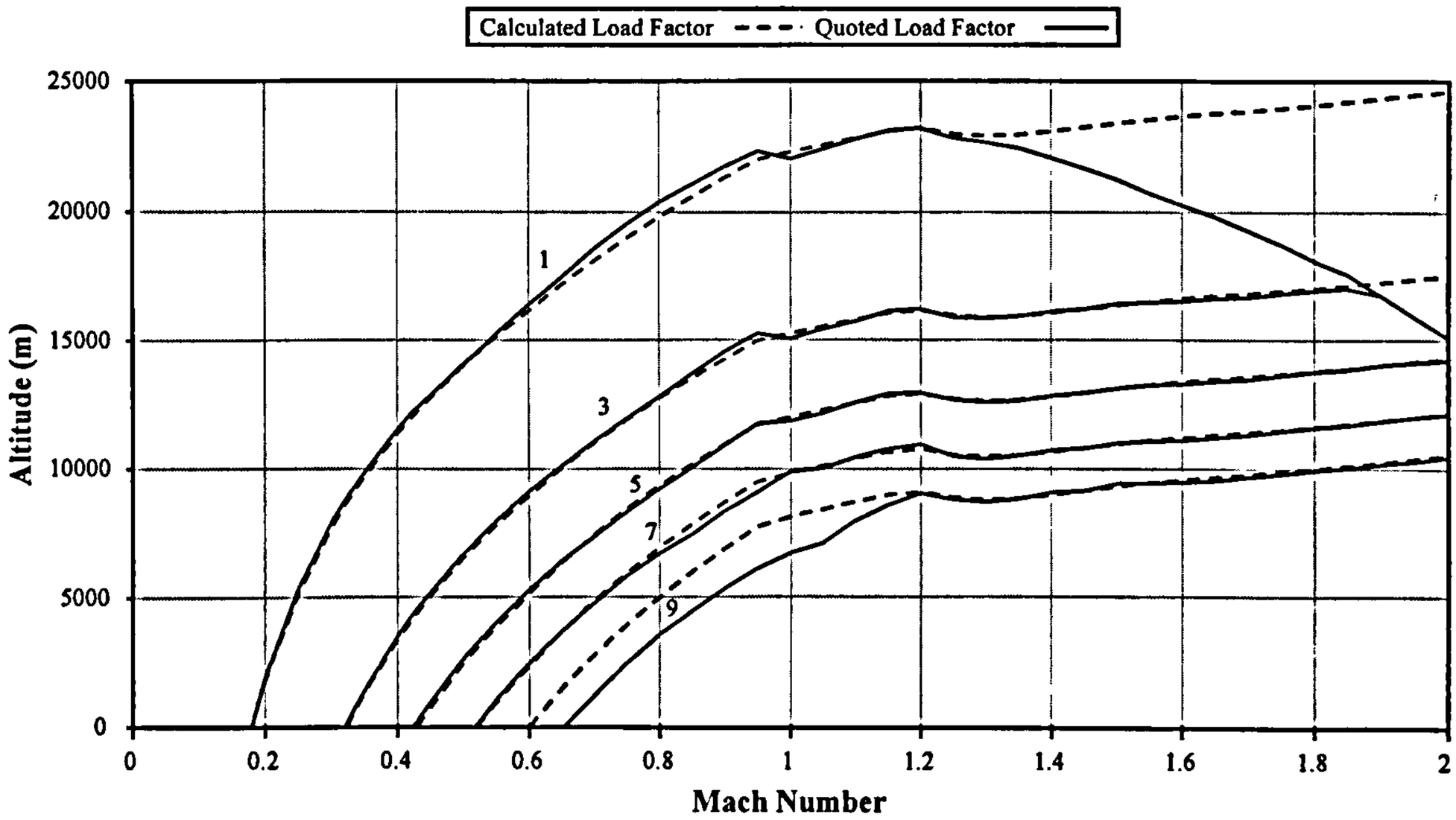


Figure 6-3. F-16C Attained Turn Load Factor Comparison Results.

The generation of sustained turn rate data file is much simpler, but relies on an external graphics program, such as GNUPLOT, to perform the interpolation of the data to produce constant load-factor contours. To write a code to perform such a task, or to calculate the sustained turn rate in a similar manner to the attained turn rate would have proven to be extremely difficult and time-consuming.

The sustained turn rate is calculated using the same methodology as that for the specified turn load factor mission phase, as described in section 6.2.5. Altitude is incremented from sea level to 110% of the maximum aircraft altitude, calculated in section 6.1.8, and a Mach number sweep from 0 to MMAX is performed. At each Mach/altitude combination the sustained load factor is calculated, and provided the value is not greater than the attained load factor, the value is written to the file. Because of the very high thrust levels available to combat aircraft, the theoretical sustained load factor can be found to be higher than the attained load factor, but the aircraft would almost certainly not be able to fly safely at the very low speeds where this occurs.

Calculated and quoted sustained turn rate data are shown in Figure 6-4. The results are not as good as for the attained turn rate, nor for many of the other comparisons, but this is to be expected. Large differences in performance can be caused by quite small discrepancies in any one of the models used for prediction of sustained turn rate, as stated in section 6.1.4. The majority of the difference is probably due to the engine model, because data for military engines is closely guarded, and validation of calculated data has proven impossible.

Although the accuracy of the absolute values could be improved on, the trends shown and the shapes of the contours are certainly similar. This should enable the comparison of competing conceptual designs using COMBAT, and the sustained supersonic load factors of real aircraft should perhaps be reduced slightly for comparative purposes.

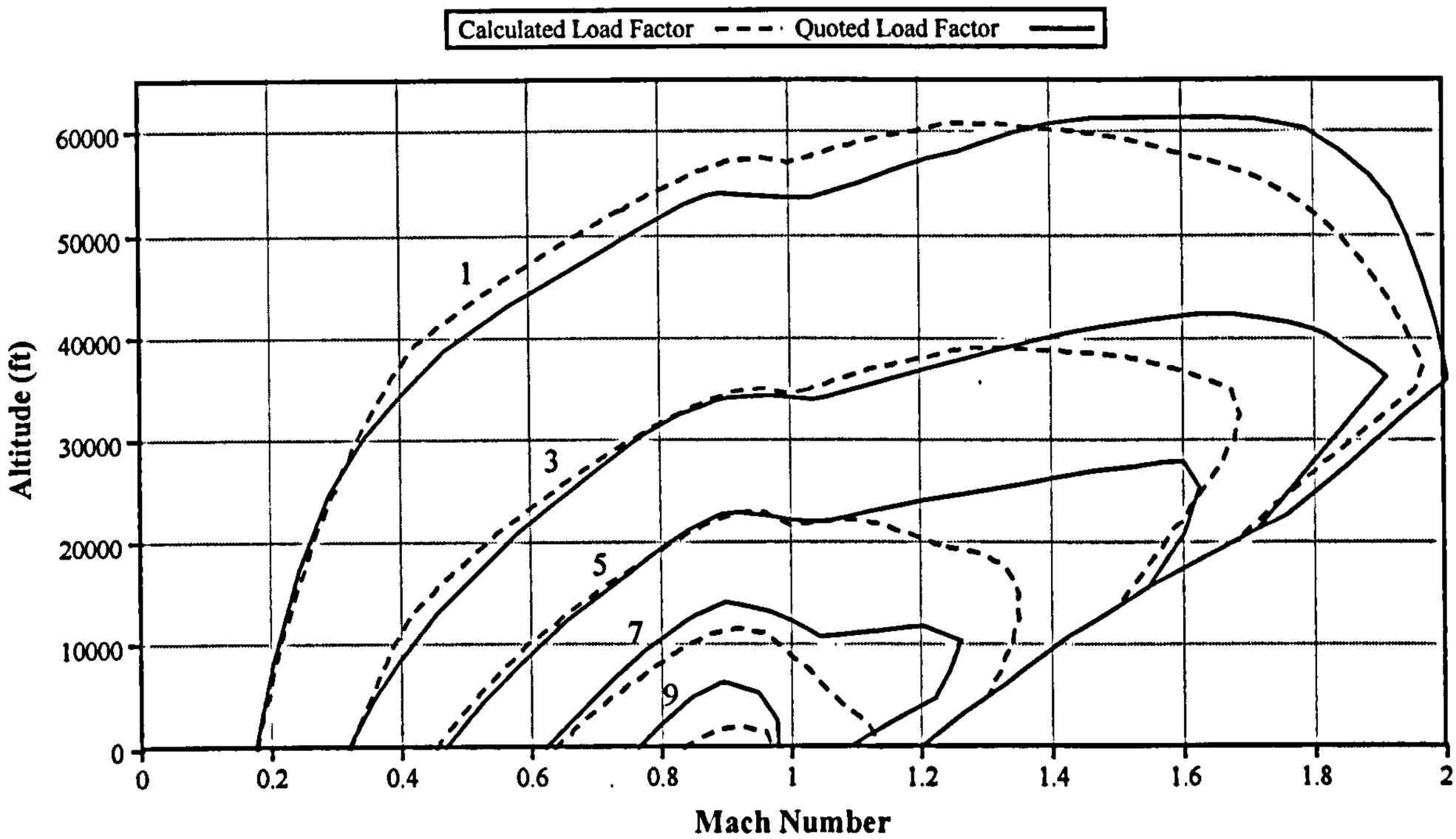


Figure 6-4. F-16C Sustained Turn Load Factor Comparison Results.

6.2.11 Post-Mission Calculations and Constraint Evaluation

Although many of the individual constraint functions have been stated in the previous chapters, a brief description of all of the constraints calculated in the sizing and performance analysis module is given here.

Constraint 1 is the objective function, and can be either aircraft gross mass, MTG, or a cost value, determined by the LCC model detailed in Chapter 7. The choice of objective function is set in the aircraft input file.

Constraint 2 checks that the available/required thrust ratio, RTPG is greater than 1.0. This ensures that there is always sufficient thrust to meet all of the flight conditions, and can be used as an extra constraint for supercruise and manoeuvring flight mission phases.

Constraint 3 checks that the total advanced materials structure mass is not larger than the structure mass itself, as described in section 4.3.1.

Constraint 4 is a check to ensure that the fuselage dimensions are adequate to accommodate the engine(s); see section 4.2.2.

Constraint 5 checks to ensure that the aircraft does not suffer from undue pitch-up at transonic speeds. It has been taken from the SWEPT aircraft synthesis model[63], and takes the form:

$$FUNC(5) = AW \times (TAN(QW4))^{0.8} - 3.2 \quad . \tag{6-32}$$

Constraint 6 checks that the total mission fuel mass is equal to the sum of the aircraft internal and external fuel masses, as described in above. This function is defined as an equality constraint in the RQPMIN input file.

Constraint 7 checks that the total mission fuel volume is equal to the sum of the aircraft internal and external fuel volumes, as described above. This function is defined as an equality constraint in the RQPMIN input file.

Constraints 11-24 are reserved for the DELTA drag estimation methodology, and ensure that the aircraft design parameters do not exceed the limits of the data interpolation included in that method. This should also help to ensure a feasible and realistic design, as the data in the DELTA method is determined from a large database of existing aircraft. All constraints have been implemented as inequality constraints and some values, which are constrained between upper and lower bounds, therefore appear twice.

Constraints 25-35 are the inequality constraints for the point performance analysis calculations as described in section 6.1.

6.3 Summary

The Aircraft Point and Mission Performance chapter has:

- Given an overview of the methods and logic behind the calculation of aircraft performance, and the role that the point performance plays in the aircraft sizing process.
- Explained the options available for both point and mission calculations, and the effects of some of the factors on the design as a whole.
- Detailed the equations and methods used to estimate aircraft point performance and, where possible, given correlations of the accuracy of those methods.
- Explained the operation of the mission performance models, with a particular emphasis on the fuel burned during each mission leg.
- Given a record of all of the constraint functions used in the COMBAT aircraft design and sizing methodology.

This page is intentionally blank.

7 Life Cycle Cost Model

This chapter describes the concept of Life Cycle Cost (LCC) and its application to this study. A description of the factors and typical breakdown is given, together with the algorithms and equations used in the prediction of LCC for peacetime operation of a combat aircraft.

The Life Cycle Cost model has been kept as a separate section to the rest of the COMBAT aircraft synthesis modules, as it is not strictly required for those routines. Indeed, if the objective function of the optimisation study is mass-based, then the LCC model is only called on the completion of the optimisation, for comparative purposes.

The LCC model is split into five main sections, each corresponding to a separate phase in the program life cycle: Research, Development, Test and Evaluation; Production; Ground Support Equipment and Initial Spares; Operation and Support; and Disposal. Each of these sections is further divided into appropriate sub-sections so that visibility of costs is maximised, and the use of Activity Based costing procedures is permitted in many of the models.

The structure of the models and equations used are given, together with the methods used to predict aircraft maintenance requirements, and economic discounting formulae. Correlations are presented where possible, although these are extremely difficult to produce for the majority of cost models, due to security restrictions, lack of data, and inconsistencies or inaccuracies in the data presentation.

7.1 Description and Model Structure

The LCC module is based on several models that have been acquired and developed from many different sources. Discounting of future costs is always used, and the model has been split into the areas most often quoted in the available literature:

7.1.1 Research, Development, Test and Evaluation (RDT&E)

This phase covers all areas of research and development prior to full-scale production of the aircraft. It includes; concept definition, design studies and integration, wind tunnel models and testing, laboratory testing, production of static and flight test airframes, avionics, software, propulsion development, flight testing, integrated logistics support, and programme oversight. RDT&E typically makes up about 15-20% of the LCC of modern, low-production (≈ 500) combat aircraft, and is obviously affected by the number of aircraft over which this cost can be amortised.

7.1.2 Production

This section includes; production engineering design, production investment (manufacturing facilities, tooling, jigs and fixtures), manufacturing labour, quality control, material and equipment, profit, overheads, administration, and purchasing of engines and avionics systems. These costs, divided by the number of aircraft, make up the Recurring Flyaway Cost, CRF.

$$CRF = \frac{CPROD}{NPROD} \ (\$). \tag{7-1}$$

The Procurement Cost would also include the cost of post-design services, initial support, and any other non-recurring costs that may be incurred. The production costs would typically account for approximately 35-40% of the total LCC of a modern combat aircraft, and the cost per aircraft decreases with number of aircraft built.

The total of the two major costs above (RDT&E & Production), divided by the total number of aircraft built, is called the Unit Acquisition Cost, CACQ, i.e.

$$CACQ = \frac{CRDTE + CPROD}{NRDTE + NPROD} \ (\$). \tag{7-2}$$

7.1.3 Operation and Support (O&S)

As the name suggests, these costs are incurred during the operation and support of the aircraft and associated weapons systems. They include; mission personnel, basing overheads and upkeep, planned maintenance, unplanned (arising) maintenance, replenishment spares, petroleum, oil and lubricants (POL), consumables, and training. The O&S phase typically contributes about 35-40% of the LCC, and it is in this area that the largest LCC savings can be made, by reducing the amount of support that an aircraft requires. This can be achieved by reducing one of many contributing factors, e.g. maintenance effort, fuel burned, aircrew, etc.

7.1.4 Disposal

This can amount to a wholly negative cost if the aircraft is sold intact at the end of its useful life. As this is very unlikely, the disposal cost model consists of the following contributors; disassembly labour, disposal of non-reusable material, and sale of scrap material (-ve). Depending on the relative values of these different components, the total disposal figure could be a cost, or a credit. The effects of discounting are most significant with disposal, and considerably reduce the effective cost of this activity.

7.1.5 Structure Definition

Although may of the models in the following sections existed in one form or another, a detailed structure of the whole LCC model was required before any CER development or validation could begin. Without a detailed structure of which costs were to be included in each section, the likely cost drivers and the interrelationships between them could not be investigated.

Many of the conceptual designs texts give 'Life Cycle Cost' estimating methods, but they are closer to basic acquisition and direct operating cost (DOC) calculations. The texts share a common theme, and it is more than likely that all of the models were derived from the same references. Certainly, the acquisition cost algorithms are based on early RAND models, of which the report by Hess and Romanoff[43] is a more recent update. The remainder of the cost models are simple expressions for the cost of fuel, salaries, etc., but no mention is made of spares costs, servicing, etc., which form the bulk of O&S costs.

Having discovered the Navy O&S cost model[5] and the US DoD CAIG report[9], a great deal of effort was expended in generating a set of cost components that could make use of the existing models where possible, but that could also be used to compare data with RAF Cost Of Support Spreadsheet (COSS) data[11]. This involved several iterations of the same cost breakdown structure, with modifications being suggested by several sources. One of the main difficulties with this section of the work was the difficulty in identifying suitable cost drivers for many of the models without having a model, against which the assumptions could be tested.

Another of the difficulties encountered is the lack of data available for activity-based costing (ABC) models, which have been described, and their benefits discussed. This was partly overcome using reports, such as that by Chaffey[22], to provide personnel data, and from numbers supplied by USAF VAMOSC staff. In this way, personnel numbers were correlated with the total aircraft maintenance figures, producing a visible and auditable figure for manning levels. Support and administration staff numbers were scaled from the numbers of direct personnel, giving a realistic method of calculating personnel and base support costs. In this way, a full O&S cost model structure was developed, making use of ABC methods wherever possible, and providing a modular format that can be built upon for future developments.

7.2 Parameter Initialisation

The Life Cycle Cost of the aircraft under consideration is determined from the simple parameters available at the conceptual design phase. Before the LCC can be calculated, several global variables must be defined, as they are used in many separate models.

7.2.1 Advanced Technology Factor

The advanced technology factor (FATF) is used in the estimation of the engineering and development support costs of the RDT&E and Production phases. It is based on the aircraft role (ROLE), the complexity of its intake system (SHOCK), the use of advanced materials in the airframe (UMATAD), and the use of low-observable features (FLO). The calculation makes use of the FORTRAN command AINT, which ignores the decimal values of the number, cutting it down to an integer value:

$$FATF = 1 - \frac{AINT(ROLE/2.1)}{10} + \frac{AINT(SHOCK/1.1)}{10} + (UMATAD \times 0.2) + 0.25 \times (FLO^2 - 1) \quad . \tag{7-3}$$

7.2.2 Number of Partners

The number of partners in collaborative aircraft programmes is calculated by dividing the number of production aircraft by the largest single buy. This means that for programmes such as Eurofighter, the effective number of partners is approximately 2.67 (UK buy is assumed to be 232 out of 620), even though there are four real partners:

$$NPART = \frac{NPROD}{LARGE} \quad . \tag{7-4}$$

7.2.3 AMPR Mass

One of the most important parameters in the calculation of the aircraft acquisition cost is the AMPR mass. AMPR mass is equivalent to the airframe unit mass, in pounds, and is defined by MIL-STD-1374 as follows:

Empty mass minus: wheels, tires, tubes & brakes; engines; rubber or nylon fuel cells; starters and APU; instruments; batteries & electrical power supply and conversion; avionics equipment (not including wiring and installation); air conditioning, anti-icing & pressurisation units; cameras and other surveillance equipment; trapped fuel and oil (if included in empty mass). Thus, the equation for the calculation of airframe unit mass is given as:

$$WAMPR = \begin{pmatrix} MTE - MPG - 0.25 \times MU - 0.75 \times MSF \\ -MSE - 0.5 \times MSC - 0.75 \times MCI - MAX \\ -MSA - MGCA - MPU \end{pmatrix} \times 2.2046 \quad (lb.), \tag{7-5}$$

where the masses are defined in section 4.3.

7.2.4 Consumer Price Index

The initialisation of variables continues with the calculation of the Consumer Price Index, CPI. This value allows for the inflation in the cost of finished goods, and is based on data published monthly by the United States Bureau of Labor Statistics. All equations that contain the cost escalation factor (CPI) have been normalised to 1983 dollars. The CPI is thus the ratio of cost for the accounting year (YRACC) and 1983.

$$CPI = \frac{4.7183 \times YRACC - 294.12}{100} \quad . \tag{7-6}$$

Figure 7-1 shows a plot of the consumer price index from 1970 to 1994 for the average of all American cities, normalised for 1983 = 100.0, and the best-fit line from which the above equation was taken. Although the equation gives only a simple linear model, this will suffice, because changes in CPI should not significantly affect the choice of competing aircraft designs as all designs would use the same economic models.

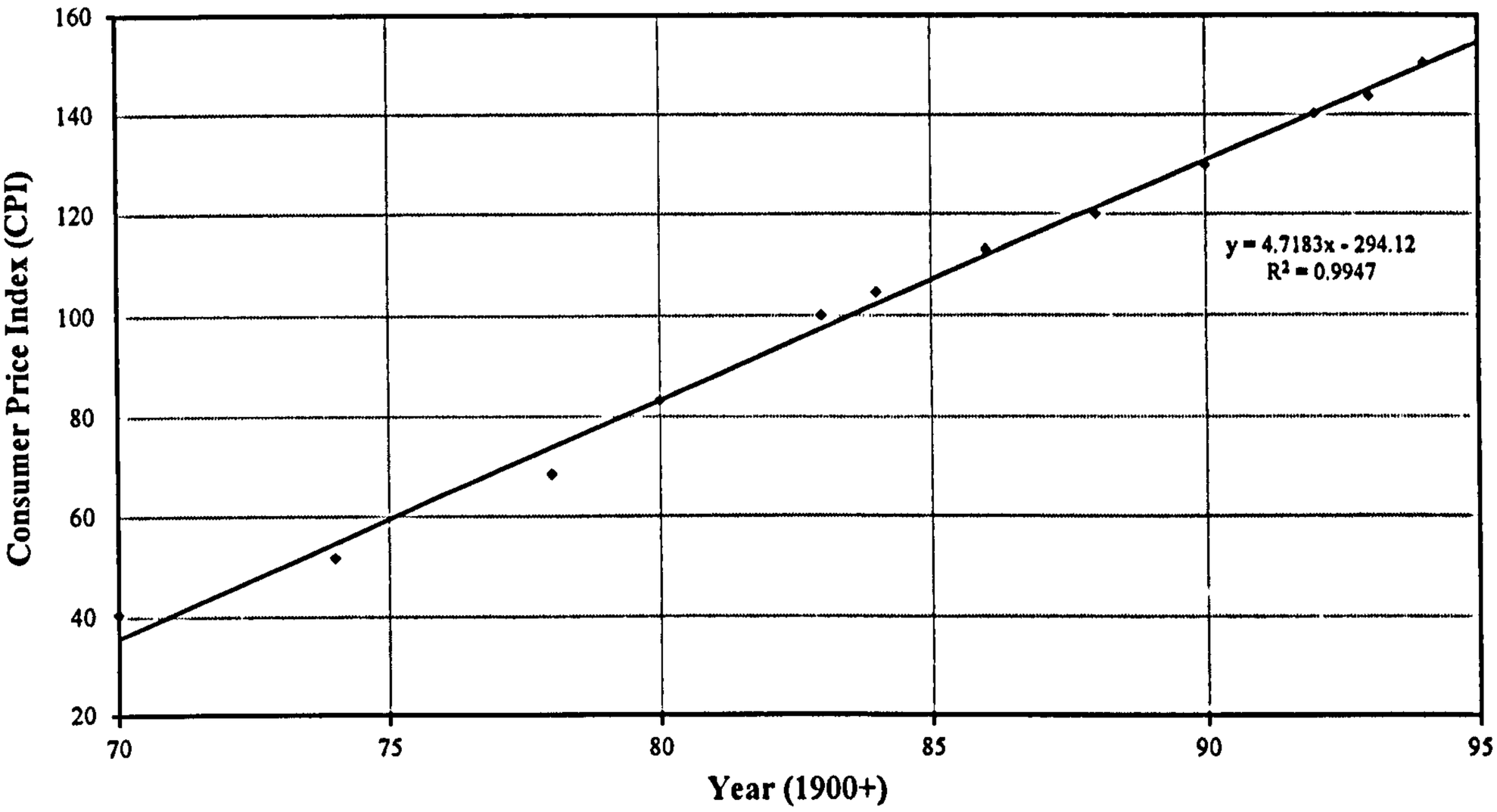


Figure 7-1. Consumer Price Index, 1983 = 100.0.

7.2.5 Labour Rates

The hourly labour rate varies with airframe manufacturer because of different rates, factors, and overhead. With the current business environment and projected reductions in military procurement, increased competition, and the desire to reduce overhead and other business operating expenses, some rates may actually level off or drop in the near future. However, it is assumed that past trends in labour rates will continue.

Figure 7-2 gives an approximation of estimated industry averages (including overhead), based on "then year" dollars, of engineering (CLE), tooling & quality control (CLT), and manufacturing labour rates (CLM). The equations below have an extra factor built in, that for security (FS), which comes from the LCC input file. The labour rates are taken from Burns[21], and are based on data from the US Bureau of Labor Statistics.

The equations for the calculation of labour rates are based only on the discipline to which they apply and the accounting year, and are expressed in US$ per hour. The engineering (CLE), quality/tooling (CLT), and manufacturing labour rates (CLM) are calculated as follows:

$$CLE = (0.0191 \times YRACC^2 - 0.5432 \times YRACC - 37.964) \times FS \quad (\$/hour), \qquad (7\text{-}7)$$

$$CLT = (0.0054 \times YRACC^2 + 1.0755 \times YRACC - 89.215) \times FS \quad (\$/hour), \qquad (7\text{-}8)$$

$$CLM = (0.0132 \times YRACC^2 \times -0.5627 \times YRACC - 13.679) \times FS \quad (\$/hour). \qquad (7\text{-}9)$$

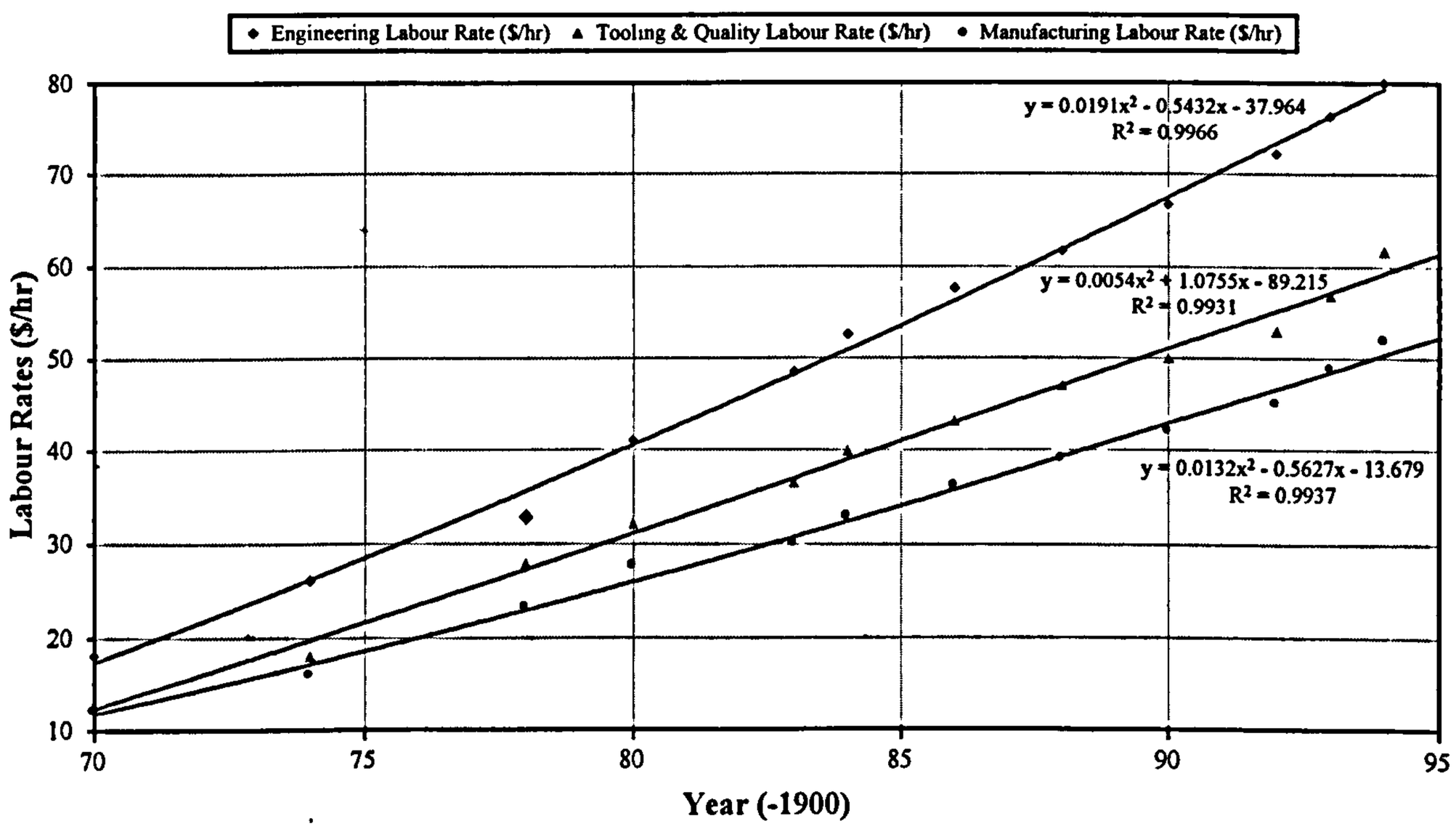


Figure 7-2. RDT&E and Production Labour Rates, US$/hour.

7.2.6 Advanced Materials Factor

The use of advanced materials can reduce the mass of the airframe and result in significant down-sizing of the aircraft. Advanced materials generally increase engineering effort, tooling, material cost, and fabrication cost, per unit mass. The substitution of aluminium with carbon/epoxy would increase initial recurring cost for a fixed mass, but the down-sizing process produces an aircraft of reduced initial cost. To account for the use of advanced materials, a simple cost factor is calculated, based on work done by Burns[21], and shown in Figure 7-3.

The cost factor for advanced materials (FAMA) is a function of the percentage, by mass, of advanced materials used for structural applications. Fibreglass laminate & fibreglass honeycomb structures are not considered advanced materials, and the mass fraction of advanced material is the mass of composite and advanced titanium alloy structures divided by the aircraft empty mass (UMATAD).

$$FAMA = 4.4028 \times UMATAD^4 - 5.7025 \times UMATAD^3 + 2.8658 \times UMATAD^2 + 0.3168 \times UMATAD + 1 \qquad (7\text{-}10)$$

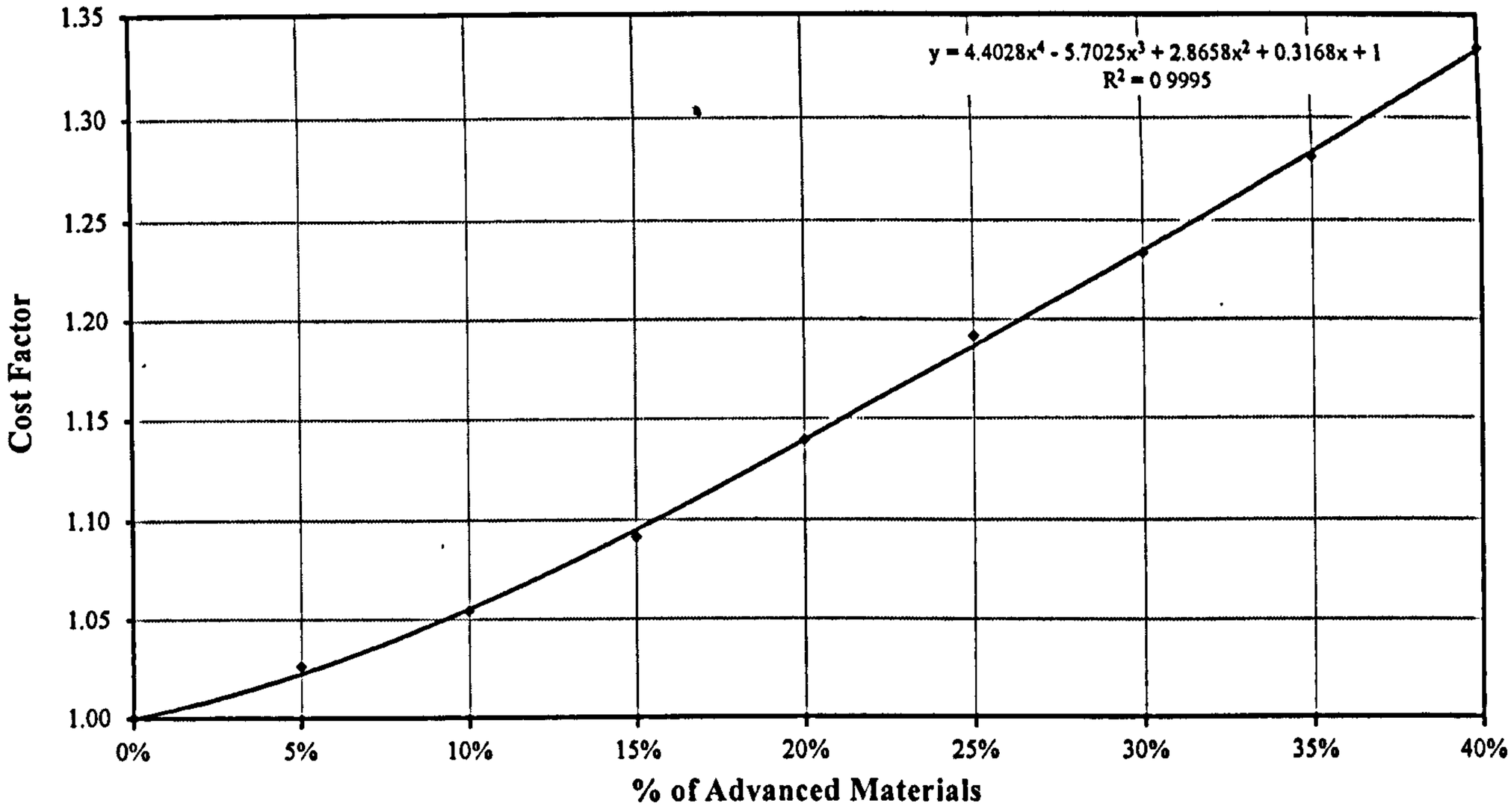


Figure 7-3. Materials Cost Factor vs. % by Mass of Advanced Materials.

7.2.7 Collaborative Factor

A separate factor to allow for the collaborative nature of modern aircraft programmes has been developed from a simple algorithm suggested by Mr P. Pugh, formerly of the Directorate of Project, Time, and Cost Analysis (DPTCAn), a division of the Ministry of Defence. It makes use of the number of partners previously calculated. The value of FPART should range from 1 for single-country, single-manufacturer programmes, and increase with the number of partners and countries involved. Current aircraft will therefore have FPART values of 1 for Gripen, approximately 1.5 for F-22 (one main design company and two other partners), and 2.67 for Eurofighter (calculated with Equation 7-2), with its multi-national work share of two major and two minor partners. The factor of 0.33 was determined by inspection for the few multi-national programmes for which recent data exists, and assuming that the current 'prime contractor' arrangements being used in the US are equivalent to a multi-partner development with slightly more than one partner, depending on the number of major sub-contractors.

$$FPART = NPART^{0.33} \quad . \tag{7-11}$$

7.3 Research, Development, Test, & Evaluation

As previously stated, this phase takes account of all costs incurred up until the construction of the first production aircraft. Several of the factors of the previous section are utilised in the calculation of RDT&E costs. Unless stated otherwise, the method for the calculation of airframe development costs should be assumed to be taken directly from the method developed by Burns[21], which is similar to many other airframe development cost estimating models, such as those by Hess[43] and Resetar[79].

Current-generation aircraft contain greater amounts of advanced composites and new technology than older generation aircraft and require more hours of design and analysis time than conventional sheet metal construction. The increased use of computers for design and analysis has improved the engineer's efficiency, which has helped offset some of the increased labour hours. This has been accounted for with the advanced design tool factor, FADT, added to the method by the author, which is supplied by the user in the LCC input file.

The methodology breaks the development procedure in to many different activities, and can thus be thought of as an 'Activity-Based Costing' procedure. Whatever the description chosen for the naming of the methodology, it is based on parametric estimating techniques for a wide range of commercial and military aircraft, with several of the previous and following factors to allow for the differences in the designs. Because of the activity-based approach, many of the RDT&E actions are calculated as an effort quantity (hours), and multiplied by the labour rates previously calculated.

7.3.1 Airframe Development

The total development airframe engineering hours (EDAE) can be estimated as follows:

$$EDAE = 0.066 \times WAMPR^{0.796} \times VMAX^{1.538} \times NRDTE^{0.183} \times FATF \times FAMA / FADT \times FPART \quad \text{(hours)}. \qquad (7\text{-}12)$$

It should be noted that WAMPR is in pounds, as given in Equation 7-5, and VMAX is in knots true airspeed, calculated from the maximum Mach number at altitude.

Engineering cost for the development phase can now be estimated:

$$CDAE = EDAE \times CLE \quad (\$). \qquad (7\text{-}13)$$

Development support is the non-recurring manufacturing effort undertaken to support engineering during the RDT&E phase of an aircraft program. It includes the cost of manufacturing labour and materials required to support mock-up requirements, test components, and other hardware needed for airframe design and development work. The development support cost (CDDS) can be estimated using:

$$CDDS = 0.0356 \times WAMPR^{0.903} \times VMAX^{1.93} \times NRDTE^{0.346} \times CPI \times FATF \times FS / FADT \times FPART \quad (\$). \qquad (7\text{-}14)$$

Flight test cost (CDFT) includes all costs incurred by the aircraft manufacturer to complete flight test, except the cost of the production of the flight test aircraft. It includes flight test engineering planning, data reduction, manufacturing support, flight test instrumentation, spares, fuel and oil, pilot's salary, facility rental, and insurance. A separate factor is included for advanced technology testing (FATT), from the LCC input file, and is a judgement factor to allow for low-observable, STOL, VTOL, and variable geometry features.

$$CDFT = 0.00558 \times WAMPR^{1.19} \times VMAX^{1.401} \times NRDTE^{1.281} \times CPI \times FATT \times FS \quad (\$). \qquad (7\text{-}15)$$

Tooling effort (EDTP) includes hours required for tool design, N/C programming, tool planning, tool fabrication, production test equipment, maintenance of tooling, normal changes and production planning. Tooling hours are dependent on production rate and construction materials used for aircraft components. For example, carbon/epoxy wing skins are likely have a higher tooling cost than aluminium ones, because aluminium skins may be machined from plate with minimal tooling and the carbon/epoxy skins require more complex tooling.

The aircraft production rate is a variable that affects tooling cost. Higher production rates result in a higher tooling cost and will cause the unit flyaway cost to be slightly higher for a given production quantity. The fluctuation of production rates, away from the maximum, during a production run cannot be simulated, because the reduction of production rate during manufacture will lead to higher overhead costs per aircraft (unused tools, hangar space and equipment), which may lead to a higher unit cost, without a reduction of any prior non-recurring investment.

$$EDTP = 5.083 \times WAMPR^{0.768} \times VMAX^{0.899} \times NRDTE^{0.18} \times RATE^{0.066} \times FAMA / FADT^{0.5} \times FPART \quad (\text{hours}). \qquad (7\text{-}16)$$

The tooling cost is then the product of the tooling effort and the tooling labour rate:

$$CDTP = EDTP \times CLT \quad (\$). \qquad (7\text{-}17)$$

Development manufacturing labour hours (EDML) include those hours necessary for machining, fabrication, and assembly of the major structure (AMPR weight), installation of purchased parts, government furnished equipment, and subcontractor assemblies and components.

$$EDML = 43.61 \times WAMPR^{0.76} \times VMAX^{0.549} \times NRDTE^{0.554} \times FAMA / FADT^{0.5} \times FPART \quad (\text{hours}). \qquad (7\text{-}18)$$

Manufacturing labour cost for development is the product of the manufacturing effort and the manufacturing labour rate:

$$CDML = EDML \times CLM \quad (\$). \qquad (7\text{-}19)$$

Quality control includes the task of inspecting fabricated and purchased parts, sub-assemblies, and assembled components for standards specified by drawings, and specifications. The quality control department is also responsible for weighing of detail parts, subassemblies, and even completed aircraft at some manufacturing facilities. The quality control effort requirement (EDQL) is scaled from the manufacturing effort; the cost of that labour (CDQL) is the product of the effort and the labour rate.

$$EDQL = 0.13 \times EDML \quad \text{(hours)}, \tag{7-20}$$

$$CDQL = EDQL \times CLT \quad (\$). \tag{7-21}$$

Manufacturing material and equipment includes the raw material, hardware and purchased parts required for the fabrication and assembly of the airframe, excluding engines and avionics. The cost of manufacturing materials for the development program (CDMME) is estimated with the following equation:

$$CDMME = 96.677 \times WAMPR^{0.692} \times VMAX^{0.639} \times NRDTE^{0.803} \times CPI \times FAMA \times FLO \quad (\$). \tag{7-22}$$

7.3.2 Engine Development

Actual engine cost data is extremely difficult to obtain, and the method used is taken directly from a model developed by Birkler[16] for the estimation of development and production costs for aircraft turbine engines. The engine cost methodology presented here correlates data from 16 turbojet and turbofan engines to produce a single equation for the engine development costs before entry into service:

$$CDP = \left(\frac{-845.804 + 0.0005 \times Thrust + 249.838 \times M + 0.313 \times TT4}{0.6 \times 0.833 \times FADT} \right) \times CPI \times FPART \times 1.0E6 \quad (\$), \tag{7-23}$$

where Thrust is the engine sea-level static gross thrust (lbf.), M is the Mach number for supersonic designs, or 1 otherwise, and TT4 is the maximum turbine inlet temperature (°R). The Birkler report gives the values in millions of 1980 dollars, explaining the constants 0.833 and 1.0E6. The only comparison with real data that has been possible is for the EJ200 engine for Eurofighter: a report by the National Audit Office[28] quotes the UK contribution to engine and accessory development costs at 94M£901. In order to correlate the model above with the quoted value, the factor of 0.6 was added. This is probably pessimistic for other engine development programmes, as the development of EJ200 has dragged on for nearly 14 years, compared with the very short development schedule for the Rolls Royce Trent, at just under 3 years[38].

Although the cost of the engine development process has been accounted for, the cost of the engines for the development aircraft (CDPF) is yet to be calculated. The model for engine production costs is taken from a USAF Flight Dynamics Laboratory report[96], and is presented with the calculations for engine production costs, in section 7.4.2. The number of engines purchased for the RDT&E phase depends on the number of test aircraft (NRDTE), excluding stationary test aircraft (NSTAT), and the number of engines, plus one spare, per aircraft:

$$N = (NRDTE - NSTAT) \times (NENG + 1) \ . \tag{7-24}$$

7.3.3 Avionics Development

Reference material on the subject of avionics development[56] suggests that there are no clear trends in the relation of avionics development costs to conceptual aircraft design variables. Development and production costs of individual avionics systems are more dependent on the performance of that particular system, which is not measured in aircraft-design terms. For instance, the cost of the radar for a combat aircraft is more likely to be measured in terms of radar functionality and technical descriptors (power, frequency range, fidelity, steering mechanism, etc.) than in terms of performance of the aircraft in which it is fitted.

In the absence of any better models, avionics cost is based on the mass of avionics (either a user input or calculated by the aircraft synthesis section of the program). The avionics cost per pound varies with the type of equipment, such as radar, core processors, racks, antennas, etc., but is averaged for the purposes of the model. Avionics installation, including wiring and supports, are included in the AMPR mass. Avionics uninstalled equipment mass, or 90% of the total avionics mass, is used for the development cost estimate, which is based on published cost data for Eurofighter 2000[28]:

$$CDV = 853820 \times WV \times FPART \times CPI \quad (\$), \tag{7-25}$$

The cost of the avionics fit for the development aircraft is based on the uninstalled avionics mass and the number of flying development aircraft:

$$CDVF = 3950 \times WV \times (NRDTE - NSTAT) \times CPI \quad (\$), \tag{7-26}$$

where the uninstalled avionics mass, WV, is in pounds.

7.3.4 Total Development

The total development cost for military aircraft programmes is simply the sum of the separate RDT&E activities:

$$\begin{aligned} CRDTE = {} & CDAE + CDDS + CDFT + CDTP + CDML + CDQL \\ & + CDMME + CDP + CDPF + CDV + CDVF \end{aligned} \quad (\$). \tag{7-27}$$

7.4 Production

As previously stated, this section includes; production engineering design, production investment (manufacturing facilities, tooling, jigs and fixtures), manufacturing labour, quality control, material and equipment, profit, overheads, administration, and purchasing of engines and avionics systems. The airframe production cost models are taken from the activity-based cost model derived by Burns[21], whilst the engine cost model is again taken from the AFFDL report[96]. Due to the lack of suitable models or data, avionics cost models are based on uninstalled avionics mass.

Because the methods, and many of the equations, are similar to those of the previous section, the cost estimating relationships are given without explanation, except where not defined previously.

7.4.1 Airframe Production

Production engineering effort and cost are estimated as follows:

$$EPAE = 0.066 \times WAMPR^{0.796} \times VMAX^{1.538} \times (NRDTE + NPROD)^{0.183} \times FATF \times FAMA / FADT^{0.5} \times FPART - EDAE \quad \text{(hours)}, \tag{7-28}$$

$$CPAE = EPAE \times CLE \quad (\$), \tag{7-29}$$

where EDAE is the engineering development effort calculated in the previous section.

Production tooling effort and cost are calculated thus:

$$EPTP = 5.083 \times WAMPR^{0.768} \times VMAX^{0.899} \times (NRDTE + NPROD)^{0.18} \times RATE^{0.066} \times FAMA / FADT^{0.5} \times FPART - EDTP \quad \text{(hours)}, \tag{7-30}$$

$$CPTP = EPTP \times CLT \quad (\$). \tag{7-31}$$

The manufacturing labour costs are calculated similarly, and take account of the development labour required:

$$EPML = 43.61 \times WAMPR^{0.76} \times VMAX^{0.549} \times (NRDTE + NPROD)^{0.554} \times FAMA / FADT^{0.5} \times FPART - EDML \quad \text{(hours)}, \tag{7-32}$$

$$CPML = EPML \times CLM \quad (\$). \tag{7-33}$$

It should be noted that the factor for advanced design tools has appeared in the last three equations for the estimation of manufacturing effort. This reflects advances that have occurred in the development of automated systems to speed the manufacturing process. These include computer-controlled machines, CAD/CAM interfaces, automated ordering and stock control systems, flexible tooling, and just-in-time (JIT) processes.

Quality control effort is again based on manufacturing labour effort:

$$EPQL = 0.13 \times EPML \quad \text{(hours)}, \tag{7-34}$$

$$CPQL = EPQL \times CLT \quad (\$). \tag{7-35}$$

The cost of the material and equipment for the RDT&E phase is subtracted from the cost of the material and equipment for the production phase:

$$\text{CPMME} = 96.677 \times \text{WAMPR}^{0.692} \times \text{VMAX}^{0.639} \times (\text{NRDTE} + \text{NPROD})^{0.803} \times \text{CPI} \times \text{FAMA} \times \text{FLO} - \text{CDMME} \quad (\$). \tag{7-36}$$

7.4.2 Engine Production

The estimation of aircraft engine costs is very difficult, as discussed earlier. The model used for estimating engine production costs is taken from a USAF Flight Dynamics Directorate report[96], and is derived from a correlation of costs for 13 turbojet and turbofan engines. This method was chosen over the Birkler RAND model[16], because it contains cost drivers that are more intuitively appealing than those in the RAND model. However, the RAND model was developed later, and contains data for more engines, so there may be a case for substitution of the current model with the RAND model.

Before the estimation of engine production costs can take place, several factors must be calculated. The first, DTT, is a measure of engine technology relative to the state-of-the-art, and is a function of engine thrust/weight, turbine inlet temperature (TT4), and the year of entry-into-service (YRIOC). As the in-service date is delayed behind the estimated technological limit, a reduction in production cost is observed.

$$\text{DTT} = 5.92781 \times \text{LOG}\left(\frac{\text{TPGH}}{\text{NENG} \times \text{MPG} \times \text{G}}\right) + 38.0163 \times \text{LOG}(\text{TT4}) - (\text{YRIOC} - 2.0) - 234.577 \quad . \tag{7-37}$$

The second factor, PRESS, defines the engine internal pressure, which is as measure of the complexity affecting the number of stages and number of parts:

$$\text{PRESS} = 2116 \times (\text{RPCH} + 0.7 \times \text{RPCH} \times \text{MLOW}^2 - 1) \quad , \tag{7-38}$$

where RPCH is the total compressor pressure ratio, and MLOW is the maximum flight Mach number at sea level, from the aircraft input file.

The following equation gives the engine production cost for the first unit:

$$\text{CPP} = \frac{5.7363\text{E}-6 \times 1.0595^{\text{DTT}} \times \text{Thrust}^{0.47333} \times \text{PRESS}^{0.64738} \times \text{CPI}}{0.542 \times 0.845 \times 1000^{\text{S}}} \times 1.0\text{E}6 \, . \tag{7-39}$$

where Thrust is the engine sea-level static thrust in lbf., and S is the progress or learning curve slope factor, assumed to be calculated from a 94.5% learning curve, so that:

$$\text{S} = \frac{\text{LOG}(0.945)}{\text{LOG}(2)} \quad .$$

The true average cumulative engine cost (CPP), in 'then-year' dollars, is now calculated from the above cost value:

$$\text{CPP} = \left[\frac{\text{N}^{(\text{S}+1)} - 1}{\text{S}+1} + \frac{\text{N}^{\text{S}} + 1}{2} + \frac{\text{S} \times \left(\text{N}^{(\text{S}-1)} - 1\right)}{12} - \frac{\text{S} \times (\text{S}-1) \times (\text{S}-2) \times \text{N}^{(\text{S}-3)} - 1}{720}\right] \times \text{CPP} - \text{CDPF} \quad . \tag{7-40}$$

7.4.3 Avionics Production

The avionics cost is again based on the uninstalled avionics mass:

$$CPV = 3950 \times WV \times CPI \times NPROD \quad (\$) \tag{7-41}$$

7.4.4 Total Aircraft Production

The total production cost of military aircraft programs is the sum of the relevant parts:

$$CPROD = CPAE + CPTP + CPML + CPQL + CPMME + CPP + CPV \quad (\$). \tag{7-42}$$

7.4.5 Acquisition Unit Cost and Recurring Flyaway Cost

The acquisition unit and recurring flyaway cost can be estimated with the following equations, which were defined in section 7.1.

$$CRF = \frac{CPROD}{NPROD} \quad (\$). \tag{7-1}$$

$$CACQ = \frac{CRDTE + CPROD}{NRDTE + NPROD} \quad (\$). \tag{7-2}$$

Both of these values can be specified as the objective function within COMBAT, so that the effects of different cost functions on the design of the aircraft can be investigated.

To demonstrate the accuracy of the results available from the production cost model, the graphic below is presented. Figures for the RDT&E phase have also been calculated, but published RDT&E data is difficult to obtain, and the figures quoted are often clearly not for RDT&E alone. Therefore, a comparison of quoted and calculated RDT&E costs would not necessarily produce a meaningful result.

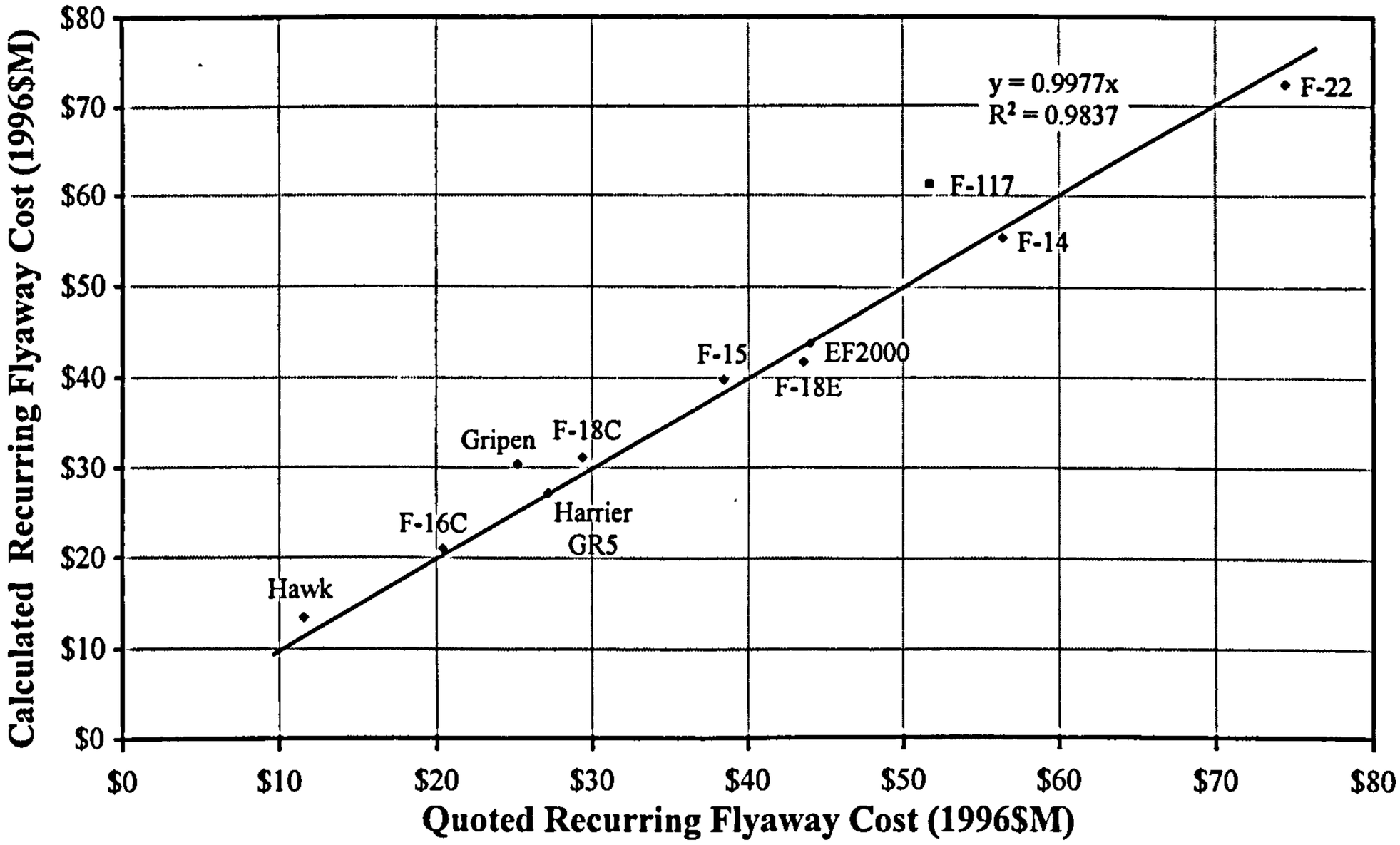


Figure 7-4. Recurring Flyaway Cost Comparison - 1996M$.

Figure 7-4 shows a comparison of quoted and calculated recurring flyaway costs. The results are seen to be excellent, with a best-fit line gradient of 0.9977, and a coefficient of determination of 0.9837. It should be noted that the calculations shown do not use the engine cost calculations shown above, but rather make use of the simpler equations given for engine costs in the paper by Burns[21].

Two aircraft give reduced accuracy, namely the JAS39 Gripen and the Lockheed F-117 Nighthawk. The F-117 is a peculiar aircraft, and there are a number of reasons that may contribute to its relatively low recurring flyaway cost:

- Although the aircraft is technologically advanced in terms of its radar cross section and concept, it is made entirely of flat surfaces, and even the wings are not cambered. Airframe manufacture will thus be relatively simple, and therefore cheap, with the absence of any complex double-curvature surfaces.
- It was designed to use off-the-shelf components, the undercarriage, wheels and brakes coming from previous USAF/USN aircraft. The propulsion system comprises two General Electric F404-GE-F1D2 engines, a derivative of the engine designed to power the F-18, with only the afterburner removed for use on the F-117.

The cost of the Gripen cannot be explained in such easy terms, and it would appear that it has been understated for one, or more, of the following reasons:

- The quote should read (for 300 aircraft) FY1994£20-25 million, not FY1994$20-25 million. The calculations from the method illustrated above suggest a recurring flyaway cost of FY94$31.3 million, which is approximately FY94£20.8 million. The main argument for this being likely is that over 3780 F-16s have been built, and its selling price is still FY94$20 million, compared to the 300 Gripen aircraft proposed at the moment.
- The number of aircraft being used for the cost estimate is inaccurate. However, this is unlikely, as the number of aircraft required to reduce the unit recurring flyaway cost to FY94$25 million is approximately 800.
- The cost quoted is the cost to the aircraft manufacturer, not its selling price to the Swedish Government, i.e. the cost of the aircraft less profit, although this would only increase the cost by approximately 8 - 10%.
- The figures released are deliberately low to mislead the public, and perhaps the rest of the aerospace industry.

7.5 Ground Support Equipment and Initial Spares

This area of LCC is very difficult to estimate because of the equipment requirements for a peculiar weapon system. In keeping with suggestions made by several members of the costing community, GSE&IS cost is simply factored from the recurring flyaway cost.

$$CGSE = 0.1 \times CRF \quad (\$). \tag{7-43}$$

This is by no means an ideal solution, but a lack of data for this life cycle phase prevents any further analysis or modelling capability.

7.6 Reliability and Maintainability

The prediction of Reliability and Maintainability (R&M) plays a vital role in the cost analysis of any programme, as it will, to a large extent, determine the bulk of the cost of the operation and support requirement of the system. R&M is particularly difficult to predict at the conceptual design stage, because much of the R&M for a given component is determined by the detailed design of that component, which is obviously not available.

The most common way of circumventing this problem is to break the aircraft down into large systems, and predict the defect rate and maintenance requirement for each system separately. This methodology was first published by Harmon *et al.*[40], and the method was up-dated and changed slightly by Serghides[86], when it was used to predict the defect rate and defect man-hour rates for RAF aircraft. A newer model was required to ensure correct prediction for modern aircraft, and this development was undertaken by Whittle[104].

It was the intention to make use of this model for the current LCC project, but late in the project timescale, differences in the requirements and capabilities of the Whittle model were discovered. These differences have prevented the use of this up-dated model, and the older Serghides model is used. However, the Serghides model would be replaced if further time was available, as it is not without faults, some of which are outlined below:

- The data from which the model was derived is for the last generation of combat aircraft, and does not contain any factors the use of advanced materials or other modern design practices.
- The data is taken from reports produced by the Maintenance Analysis and Computing Establishment (MACE), RAF Swanton Morley. The collecting procedures used at MACE were poor, and a general consensus of opinion, even from within the RAF, has emerged that the MACE data should be treated with extreme caution.
- Some of the systems used by the model are not significant factors. For example, the oxygen system is treated as a separate system but, for Tornado and more modern aircraft, makes up less than 0.2% of the total maintenance effort. The breakdown of the areas to be modelled should be better planned, with groupings such as 'Crew Systems', to enable the study of uninhabited vehicles in future studies.
- The powerplant can be a major driver of maintenance requirements, yet the reliability of the engine is modelled as a function of thrust and the number of engines. Thus, the effect of significant changes to the engine, such as de-rating, are not taken into account.
- Many of the estimating equations in the Serghides methodology are based primarily on mass. This will result in an LCC model that will converge on a result that is very similar to a minimum mass optimisation study. Whilst not incorrect, it is felt that a more representative set of R&M drivers would result in aircraft more likely to be optimised for LCC.

These reservations aside, the R&M models have enabled the study to prove that the concept of LCC optimisation is feasible, and to provide a suitable building block for future development of the LCC models.

7.6.1 Arising Maintenance

Arising, or unscheduled, maintenance results from the failure of a component or system on the aircraft. The prediction of total defect rates and maintenance effort requirement is based on a systematic approach, as discussed previously, and is intended to represent the total maintenance effort required to repair any defects that arise during peacetime operations. The equations for the prediction of Defect Rates (failures per thousand flying hours), and Defect man-hour rates (hours of effort required per thousand flying hours) are given below.

The variable YR52 is the time difference (in years) between the year of initial operational capability (YRIOC) and 1952, and is used in the calculation of the system technology improvement factors.

For the Environmental Control System, defect rate (DSA) and maintenance effort (ESA) are given as:

$$DSA = (17.13 + 7.907 \times NENG + 5.927E-5 \times MTE) \times (1.134 - 0.01563 \times YR52) \quad , \tag{7-44}$$

$$ESA = (53.07 + 3.956E-3 \times MTE + 24.76 \times HFM) \times (0.9588 \times 0.975^{YR52}) \tag{7-45}$$

where NENG is the number of engines, MTE is the aircraft empty mass, and HFM is the height of the fuselage centreline from the ground. This is a measure of the ease of access to distributed systems, such as the ECS.

Flying Controls:

$$DSC = (36.85 + 16.7 \times NCP - 0.07298 \times SW) \times (1.081 - 9.604E-3 \times YR52), \tag{7-46}$$

$$ESC = \left(429.7 + 1.222 \times \frac{TPGH}{1000} + 71.44 \times NCP\right) \times (0.9694 \times 0.995^{YR52}) \quad . \tag{7-47}$$

where NCP is the number of crew, SW is the gross wing area, and TPGH is the total sea-level static thrust available.

Fuel System:

$$DSF = EXP\left(2.85 + 0.7012 \times NENG - 1.683E-3 \times \frac{TPGH}{1000}\right) \times (1.567 - 0.04897 \times YR52) \quad , \tag{7-48}$$

$$ESF = \left(-12.57 + 115.8 \times HFM + 0.8559 \times \frac{TPGH}{1000}\right) \times (1.269 \times 0.972^{YR52}) \quad . \tag{7-49}$$

Hydraulics and Pneumatics:

$$DSH = EXP\left(3.438 + 0.2742 \times NENG + 8.082E-4 \times \frac{TPGH}{1000}\right) \times (1.303 - 0.02316 \times YR52) \quad , \tag{7-50}$$

$$ESH = \left(-367.8 + 179.8 \times HFM - 0.5396 \times \frac{TPGH}{1000}\right) \times (1.093 \times 0.994^{YR52}) \quad . \tag{7-51}$$

Undercarriage System:

$$DU = (252.7 - 0.0182 \times MB) \times (1.518 - 5.648E-3 \times YR52) \quad , \tag{7-52}$$

$$EU = EXP(6.22 + 2.445E-5 \times MTE) \times (1.149 \times 0.999^{YR52}) \quad , \tag{7-53}$$

where MB is the maximum payload to be carried by the aircraft.

Crew Oxygen:

$$DSO = EXP(1.623 + 4.553E-5 \times HMAX + 0.08466 \times NCP) \times (1.128 - 0.01064 \times YR52) \quad , \tag{7-54}$$

$$ESO = EXP(2.228 + 1.159E-4 \times HMAX) \times (1.346 \times 0.9898^{YR52}) \quad , \tag{7-55}$$

where HMAX is the maximum operational altitude of the aircraft, and is calculated by the aircraft synthesis routines.

Miscellaneous Utilities:

$$DSM = (-13.5 + 18.47 \times MMAX) \times (1.397 - 0.04526 \times YR52) \quad , \tag{7-56}$$

$$ESM = (-92.77 + 123.0 \times MMAX) \times (1.069 \times 0.92^{YR52}) \quad , \tag{7-57}$$

where MMAX is the maximum level flight Mach number, at altitude.

Aircraft Structure:

$$DS = (3.49 + 117.6 \times NENG + 2.742E-5 \times MTE) \times (1.182 - 0.02487 \times YR52) \times (1.0 - UMATAD^2) \quad , \tag{7-58}$$

$$ES = (523.2 + 0.127 \times MTE + 1.924 \times HFM) \times (1.062 \times 0.9835^{YR52}) \times (1 - UMATAD) \quad . \tag{7-59}$$

where UMATAD is the fraction of the aircraft empty mass made up of advanced materials. This factor was added during the development of COMBAT to take account of the effect of advanced materials on the airframe defect and repair rates.

Powerplant:

$$DP = EXP\left(3.919 + 2.996E-3 \times \frac{TPGH}{1000} + 0.01776 \times NENG\right) \times (1.295 - 0.02531 \times YR52) \quad , \tag{7-60}$$

$$EP = EXP\left(5.821 + 0.1833 \times HFM + 2.576E-3 \times \frac{TPGH}{1000}\right) \times (1.034 \times 0.988^{YR52}) \quad . \tag{7-61}$$

Armament System:

$$DSB = (-206.9 + 27.43 \times NHARD + 123.2 \times MMAX) \times (0.6555 + 1.92E-3 \times YR52), \quad (7\text{-}62)$$

$$ESB = (-1708.0 + 1173.0 \times MMAX + 158.7 \times NHARD) \times (0.8649 \times 0.9803^{YR52}), \quad (7\text{-}63)$$

where NHARD is the number of stores carriage points, from the aircraft input file.

Avionics Systems:

$$DAX = (97.6 + 2.39E-3 \times MTT - 1.781E-3 \times RCOMB) \times (0.9529 + 1.867E-3 \times YR52), \quad (7\text{-}64)$$

$$EAX = \left(-2670.0 + 963.0 \times HFM - 4.919 \times \frac{TPGH}{1000}\right) \times (0.2827 \times 0.999^{YR52}), \quad (7\text{-}65)$$

where RCOMB is the combat radius, and is calculated in the aircraft synthesis routines.

Navigation and Communication:

$$DSN = \left(53.33 + 0.3136 \times \frac{TPGH}{1000} + 0.01326 \times RCOMB\right) \times (1.05 - 7.358E-3 \times YR52), \quad (7\text{-}66)$$

$$ESN = \left(159.1 + 4.047 \times \frac{TPGH}{1000} + 28.35 \times HFM\right) \times (0.9091 \times 0.9699^{YR52}). \quad (7\text{-}67)$$

Electrical and Instruments:

$$DSE = (-234.5 + 0.02227 \times HMAX + 39.14 \times MMAX) \times (1.156 - 0.01716 \times YR52), \quad (7\text{-}68)$$

$$ESE = \left(-63.71 + 138.9 \times HFM + 1.601 \times \frac{TPGH}{1000}\right) \times (1.112 \times 0.9858^{YR52}). \quad (7\text{-}69)$$

The total defect rate (DT) and the total arising maintenance effort (ET) are now calculated by summing the values for the individual systems per flight hour (FH):

$$DT = \frac{\begin{pmatrix} DSA + DSC + DSF + DSH + DU + DSO + DSM \\ +DS + DP + DSB + DAX + DSN + DSE \end{pmatrix}}{1000} \text{ (defects/FH)}, \quad (7\text{-}70)$$

$$ET = \frac{\begin{pmatrix} ESA + ESC + ESF + ESH + EU + ESO + ESM \\ +ES + EP + ESB + EAX + ESN + ESE \end{pmatrix}}{1000} \text{ (man-hours/FH)}. \quad (7\text{-}71)$$

7.6.2 Scheduled Maintenance

The scheduled, or preventative, maintenance required by the aircraft is modelled using a method taken from a paper by Harmon *et al*[40]. The method predicts the scheduled inspection and maintenance effort requirements using a similar method for the arising maintenance prediction. The effort is split into five discrete sections, those being; pre-flight, post-flight, periodic (avionics), special (propulsion), and periodic (shop - propulsion). These last three names represent the equivalent RAF maintenance areas of 2nd line avionics, and 1st and 2nd line propulsion, respectively.

The models are very simple, and are again based mainly on mass and thrust (which is proportional to mass, assuming a constant thrust/weight ratio). This is undesirable, but as there is less data available for scheduled maintenance than there is for arising maintenance, it is extremely difficult to check the accuracy of the models, let alone start to generate new ones. Factors for the reduction of scheduled maintenance time and improved design processes have been built in using a time based 'Improvement Factor'. These result in a small reduction in scheduled maintenance effort per year of delay of introduction after 1952 (the year of the first aircraft in the arising maintenance model). The values were guessed at intuitively, although some guidance was taken from Whittle's Technology Improvement Factors for his arising maintenance model.

The pre-flight inspection and maintenance effort is based on the aircraft mass (MTT), the percentage of composite materials in the airframe (UMATAD), and the aircraft IOC year relative to 1952 (YR52). The composite materials factor is used in the denominator to take account of the larger aircraft size for a given mass:

$$\mathrm{EMSPF} = \frac{\left(0.3763 + \frac{0.0179 \times \mathrm{MTT} \times 2.2046}{1000}\right)}{\sqrt{(1 - \mathrm{UMATAD})}} \times 0.99^{\mathrm{YR52}} \quad . \tag{7-72}$$

Post-Flight:

$$\mathrm{EMSPO} = \frac{\left(1.2556 + \frac{8.2\mathrm{E}{-3} \times \mathrm{MTT} \times 2.2046}{1000}\right)}{\sqrt{(1 - \mathrm{UMATAD})}} \times 0.99^{\mathrm{YR52}} \quad . \tag{7-73}$$

Periodic (avionics) - 2nd Line Avionics maintenance - is based on the mass of the avionics suite, and a factor from the Modular Life Cycle Cost Model report[96], that uses the packing density of the fuselage (ROAC) to account for avionics accessibility :

$$\mathrm{EMSAX} = \left(0.3266 + \frac{0.842 \times \mathrm{MAX} \times 2.2046}{1000} + \frac{0.0387 \times \mathrm{ROAC}}{16.019}\right) \times 0.98^{\mathrm{YR52}} . \tag{7-74}$$

Special (propulsion) is equivalent to 1st line propulsion inspection and maintenance, and is based on the maximum engine thrust (thousands of pounds force):

$$\mathrm{EMSP} = \left(0.3444 + \frac{0.0467 \times \mathrm{TPGH}}{4448}\right) \times 0.985^{\mathrm{YR52}} \quad . \tag{7-75}$$

Periodic (shop - propulsion) is equivalent to 2nd line scheduled engine inspection and maintenance:

$$\text{EMSPS} = \left(0.1603 + \frac{0.0221 \times \text{TPGH}}{4448}\right) \times 0.985^{\text{YR52}}. \quad (7\text{-}76)$$

Total Scheduled Maintenance is simply the sum of the above effort requirements:

$$\text{EMST} = \text{EMSPF} + \text{EMSPO} + \text{EMSAX} + \text{EMSP} + \text{EMSPS}. \quad (7\text{-}77)$$

7.6.3 Total Maintenance Effort

The total maintenance effort is the sum of the total arising maintenance effort (ET), and the total scheduled inspection and maintenance effort (EMST):

$$\text{EMT} = \text{ET} + \text{EMST}. \quad (7\text{-}78)$$

7.7 Operation and Support (O&S)

The operation and support (O&S) costs for modern combat aircraft can be split into several major parts, all of which will be contributors to the cost of using combat aircraft in a peacetime operating regime. In wartime, the cost of operation and support becomes much less important, with all resources made available to win the particular conflict.

O&S costs are composed of: operation (mission) personnel; support personnel; service allowances, personnel support, and training; unit level consumption; contract costs for airframe, avionics and propulsion; sustaining support funds; and basing overheads and upkeep. The breakdown of Operation and Support costs follows the methods and structure suggested by the US Office of the Secretary of Defense in their 'Operating and Support Cost Estimating Guide'[9].

The Operation and Support cost is calculated for one Main Operating Base (MOB), as this will have a significant effect on the staff requirements for the particular aircraft, especially with regard to the number of senior Officers. This gives the added bonus that the new model allows different basing levels to be investigated, whereas previous cost models have been for the total number of aircraft procured. The cost for the individual aircraft contains the total deflated O&S cost divided by the number of aircraft on the base.

7.7.1 Operation Personnel

Aircrew costs are calculated from the number of active flight-crew members needed to operate the aircraft, given by the number of aircraft (NAC), the number of crew per aircraft (NCP), and the 'crew ratio' (FMOF). For combat aircraft (fighters and attack aircraft), the crew ratio is normally 1.25, and one aircraft is usually kept spare:

$$\text{NOMOF} = \text{FMOF} \times \text{NCP} \times \left(\text{NAC} - \frac{\text{NAC}}{13} \right) . \tag{7-79}$$

The number of First Line Operation Personnel is calculated from the number of aircraft (NAC), the annual flying time per aircraft (FHY), and the scheduled maintenance effort per flying hour (EMST). All personnel numbers are estimated from the values quoted as typical by Chaffey[22], but a US Navy study report[5] was also used as the basis of the personnel estimating relationship algorithms. The numbers of Officers and Enlisted personnel for First Line Operations and Administration are calculated as follows:

$$\text{NOMOM1} = \frac{\text{NAC}}{13} \times 2.0 \quad \text{for 1st Line operations officers, and} \tag{7-80}$$

$$\text{NOMEM1} = \left(1.67 + 0.0398 \times \frac{\text{FHY}}{12} \times \text{EMST} \right) \times \text{NAC} \quad \text{for enlisted.} \tag{7-81}$$

$$\text{NOMOA1} = 1.0 + \frac{\text{NAC}}{13} \quad \text{for 1st Line administrative officers, and} \tag{7-82}$$

$$\text{NOMEA1} = \frac{\text{NAC}}{13} \times 5.0 \quad \text{for enlisted administrators.} \tag{7-83}$$

Second Line Operation and Administration personnel are calculated similarly, but the numbers of officers and civilians are scaled from the enlisted personnel requirement, which is itself calculated from the total arising maintenance effort:

$$\text{NOMEM2} = \left(1.67 + \frac{0.0575 \times \text{FHY} \times \text{ET}}{12}\right) \times \text{NAC} \quad \text{for 2nd Line enlisted,} \tag{7-84}$$

$$\text{NOMOM2} = 2.0 + \frac{\text{NAC}}{13} \times 2.0 + 0.014 \times \text{NOMEM2} \quad \text{for officers, and} \tag{7-85}$$

$$\text{NOMCM2} = 0.0577 \times \text{NOMEM2} \quad \text{for 2nd Line civilians.} \tag{7-86}$$

The numbers of Officers, Enlisted, and Civilian personnel are now calculated by summing the previously calculated values:

$$\begin{aligned} \text{NOMO} &= \text{NOMOF} + \text{NOMOM1} + \text{NOMOA1} + \text{NOMOM2}, \\ \text{NOME} &= \text{NOMEM1} + \text{NOMEA1} + \text{NOMEM2}, \\ \text{NOMC} &= \text{NOMCM2}. \end{aligned} \tag{7-87}$$

The cost for each of the groups is found from the numbers calculated above, and the relevant pay rate, for officers, enlisted, and civilians, (OPR, EPR and CPR) respectively.

$$\begin{aligned} \text{COMO} &= \text{NOMO} \times \text{OPR} \ (\$), \\ \text{COME} &= \text{NOME} \times \text{EPR} \ (\$), \\ \text{COMC} &= \text{NOMC} \times \text{CPR} \ (\$). \end{aligned} \tag{7-88}$$

The total Operation Personnel requirement and cost are then calculated:

$$\begin{aligned} \text{NOM} &= \text{NOMO} + \text{NOME} + \text{NOMC}, \\ \text{COM} &= \text{COMO} + \text{COME} + \text{COMC} \ (\$). \end{aligned} \tag{7-89}$$

7.7.2 Support Personnel

The Support Personnel requirement is determined by the numbers of aircraft and operation personnel on the base. The numbers of Officer, Enlisted, and Civilian support personnel required for the Operations Wing are given by the following equations:

$$\text{NSMOO} = 4.0 + 0.6155 \times \text{NAC} + 0.0555 \times \text{NOMOF} \ . \tag{7-90}$$

$$\text{NSMEO} = 8.0 + 5.8974 \times \text{NAC} + 0.0265 \times \text{NOM} \ . \tag{7-91}$$

$$\text{NSMCO} = \frac{\text{NOMOF}}{45} \ . \tag{7-92}$$

The numbers of Officer, Enlisted, and Civilian support personnel required for the Administration Wing are given by the following equations:

$$\text{NSMOA} = 3.0 + 0.00837 \times \text{NOM} \ , \tag{7-93}$$

$$\text{NSMEA} = 8.0 + 0.0656 \times \text{NOM} \ , \tag{7-94}$$

$$\text{NSMCA} = 0.0705 \times \text{NOM} \ . \tag{7-95}$$

7.7.3 Total Personnel Numbers and Cost

The total number of Operation and Support Personnel by rank is calculated by summing the previous values, and the costs are found from the totals of the Officer, Enlisted, and Civilian numbers, multiplied by the relevant pay rate:

$$\begin{aligned} NOSMO &= NOMO + NSMOO + NSMOA \ , \\ NOSME &= NOME + NSMEO + NSMEA \ , \\ NOSMC &= NOMC + NSMCO + NSMCA \ . \end{aligned} \tag{7-96}$$

$$\begin{aligned} COSMO &= NOSMO \times OPR \ (\$), \\ COSME &= NOSME \times EPR \ (\$), \\ COSMC &= NOSMC \times CPR \ (\$). \end{aligned} \tag{7-97}$$

The total number of Operation and Support Personnel is then the sum of all of the personnel on the Main Operating Base, as calculated in the previous sections:

$$NOSM = NOSMO + NOSME + NOSMC \ . \tag{7-98}$$

The total O&S personnel cost is the sum of the previous cost values:

$$COSM = COSMO + COSME + COSMC \ (\$). \tag{7-99}$$

7.7.4 Service Allowances, Personnel Support and Training

This section of the O&S model attempts to capture the costs of Officer and Enlisted personnel training costs, training funds, and permanent change of station allowances. The models were adapted from the US Navy report[5], and the values were calibrated against available RAF Cost Of Support Spreadsheet[11] (COSS) data. The structure follows that suggested by the US Office of the Secretary of Defense, Cost Analysis and Improvement Group in their 'Operating and Support Cost Estimating Guide'[9].

Officer training costs are factored from the number of Officer, Enlisted and total Operation personnel numbers:

$$CSOT = (0.0643 \times NOMO + 0.0028 \times NOM + 0.0001 \times NOME) \times OPR \ (\$). \tag{7-100}$$

Enlisted training costs are calculated in a similar manner, but make use of the enlisted pay rate, EPR:

$$CSET = (0.1294 \times NOME + 0.0232 \times NOM + 0.0077 \times NOMO) \times EPR \ (\$). \tag{7-101}$$

Training funds are estimated from the numbers of ranked operation personnel and the total value, and are factored to bring them in line with inflation. The final constant is included to account for the original USN report giving the cost in thousands of FY78$US.

$$CSTF = (0.0042 \times NOME + 0.052 \times NOMO + 0.1128 \times NOM) \times \frac{CPI}{0.739 \times 10^{-4}} \ (\$). \tag{7-102}$$

The Officer and Enlisted permanent change of station (PCS) values take account of the cost of the continual rotation of personnel within the Royal Air Force. The values are therefore based on a given value per change of station, and the numbers of Officers and Enlisted personnel on the Main Operating Base. Officer PCS is given by:

$$\text{CSOPCS} = 1795 \times \frac{\text{CPI}}{1.5884} \times \text{NOSMO} \quad (\$), \tag{7-103}$$

and Enlisted personnel PCS is given by:

$$\text{CSEPCS} = 940 \times \frac{\text{CPI}}{1.5884} \times \text{NOSME} \quad (\$). \tag{7-104}$$

The total amount for service allowances, and training is the sum of the previous values:

$$\text{CSASTT} = \text{CSOT} + \text{CSET} + \text{CSTF} + \text{CSOPCS} + \text{CSEPCS} \quad (\$). \tag{7-105}$$

This concludes the personnel cost calculations for the Main Operating Base. The cost per aircraft is calculated by dividing the total personnel cost by the number of aircraft on the base.

7.7.5 Unit Level Consumption

This section of the O&S model attempts to capture the costs for all consumables used in maintaining the aircraft, including: petroleum, oil, and lubricants; maintenance materials; miscellaneous support supply; depot level reparables; and temporary additional duty. The algorithms are again taken from the US Navy O&S cost methodology[5], and have been calibrated with RAF COSS data, where possible.

Fuel cost is calculated from the aircraft fuel load (MTGF), the total number of flight hours per year (FHY), the design sortie time (TIME), and the cost of fuel (per litre), which comes from the LCC input file.

$$\text{COF} = \frac{\text{MTGF}}{0.788} \times \frac{\text{FHY}}{\text{TIME}} \times \text{CFUEL} \times \left(\text{NAC} - \frac{\text{NAC}}{13} \right) \quad (\$). \tag{7-106}$$

The cost of other Petroleum, Oil, and Lubricants required by the aircraft is assumed to be 3.6% of the annual fuel cost:

$$\text{COPOL} = 0.036 \times \text{COF} \quad (\$). \tag{7-107}$$

Maintenance material cost is the cost of materials consumed in the operation, maintenance, and support of the aircraft system and associated support equipment at the base level. It is determined by the total monthly flying rate, the total maintenance burden of the aircraft (EMT), the consumer price index, and its maximum speed (VMAX). This last variable acts as a technology driver, and other factors to allow for composite materials would be added if data were available to allow effective calibration.

$$\text{COMMAT} = 0.94 \times \left(6.9313 + 0.1028 \times \frac{\text{FHY}}{12} \times \text{EMT} + 0.145 \times \text{VMAX} \right) \times \text{FHY} \times \frac{\text{CPI}}{0.739} \times \text{NAC} \quad (\$). \tag{7-108}$$

Miscellaneous support supply costs are calculated in the same manner as above, and take account of supplies and equipment expended in support of mission personnel:

$$\text{COMSS} = 0.5 \times \left(6.9313 + 0.1028 \times \frac{\text{FHY}}{12} \times \text{EMT} + 0.145 \times \text{VMAX}\right) \times \text{FHY} \times \frac{\text{CPI}}{0.739} \times \text{NAC} \quad (\$). \tag{7-109}$$

Depot Level Reparable (also called exchangeables) cost is the value associated with the replacement of the initial spares stock. DLRs may include repairable individual parts, assemblies, or sub-assemblies that are required on a recurring basis for the repair of major end items of equipment. Their cost is calculated from the total maintenance burden for the aircraft, its maximum flight speed, the total number of flight hours per year, and the consumer price index:

$$\text{CODLR} = 4.25 \times \text{EMT}^{1.2234} \times \text{VMAX}^{0.2486} \times \text{FHY} \times \frac{\text{CPI}}{0.739} \quad (\$). \tag{7-110}$$

Temporary additional duty cost is calculated as a cost per aircraft, multiplied by a corrected consumer price index factor. It takes account of the cost of personnel travel for training, administrative or other purposes such as crew deployments, rotations, etc.

$$\text{COTAD} = \text{NAC} \times \frac{11000}{1.5884} \times \text{CPI} \quad (\$). \tag{7-111}$$

Transport costs are the costs involved in moving primary mission and support supplies and equipment, repair parts, secondary items, POL, and ammunition to and from the MOB. They are factored from the costs incurred for the other unit level consumption activities, and by the contract costs, calculated below:

$$\text{COUTRN} = 0.0379 \times \begin{pmatrix} \text{COF} + \text{COPOL} + \text{COMMAT} + \text{COMSS} \\ + \text{CODLR} + \text{COAC} + \text{COPC} \end{pmatrix} \quad (\$). \tag{7-112}$$

There is quite a large cost that cannot be accounted for directly, and is added under 'Other' costs. It accounts for the cost of special support equipment, communications, vehicles, computers, administrative support, and service contracts for custodial services. It is calculated from several of the other costs:

$$\text{COUOTH} = 0.25 \times \begin{pmatrix} \text{COF} + \text{COPOL} + \text{COMMAT} + \text{COMSS} \\ + \text{CODLR} + \text{COTAD} + \text{COUTRN} \end{pmatrix} \quad (\$). \tag{7-113}$$

Total Unit Level Consumption is then given by the sum of its constituent parts:

$$\text{COULCT} = \text{COF} + \text{COPOL} + \text{COMMAT} + \text{COMSS} + \text{CODLR} + \text{COTAD} + \text{COUTRN} + \text{COUOTH} \quad (\$). \tag{7-114}$$

7.7.6 Contracts

The contract costs for the aircraft comprise what was historically thought of as Third and Fourth Line maintenance. With the re-structuring of the RAF, it was advised by senior Officers that it may be more applicable to treat these costs as annual contract values, rather than using the work-breakdown approach like the one for First and Second Line maintenance. The costs can be split among the three main aircraft systems - airframe, propulsion, and avionics- and supply.

The Mechanical contract cost calculation is based on the total maintenance burden of the aircraft (EMT), the all-up-mass of the aircraft (MTT), the number of aircraft (NAC), the average annual flight time (FHY), and the consumer price index (CPI). These drivers all seem reasonable, but it would be of benefit to include the effects of the amount of composite materials, and other new technologies. A lack of consistent data has prevented these extra factors from being added.

$$\mathrm{COAC} = \left[\begin{array}{l}\left(-9.7977 + 5.2085\times\mathrm{EMT} + 1.1902\times\mathrm{MTT}\times\dfrac{2.2046}{1000}\right)\times 1000 \\ \times\dfrac{\mathrm{FHY}}{160}\times\mathrm{NAC} + (32.20965 + 7.688\times\mathrm{EMT})\times\mathrm{FHY}\times\mathrm{NAC}\end{array}\right]\times\frac{\mathrm{CPI}}{0.739} \tag{7-115}$$

Engine contract cost calculations are more complex, and require a greater number of factors, many of which have been determined from comparison with historical data.

The unit cost for an engine overhaul (ERO) and for an engine repair (ERM) are given using the following equations:

$$\mathrm{ERO} = 5.574 + 4.527\times\frac{\mathrm{TPGH}}{4448} + 70.71\times\mathrm{FD} \quad (\$), \tag{7-116}$$

$$\mathrm{ERM} = 8.9434 + 1.235\times\frac{\mathrm{TPGH}}{4448} + 11.321\times\mathrm{FD} \quad (\$). \tag{7-117}$$

Where:

TPGH is engine sea-level static thrust (N), and FD is a dummy variable to allow for the increased cost of maintaining turbofan engines. If the bypass ratio > 0, FD = 1, else = 0.

$$\mathrm{COPC} = 1.9^{(\mathrm{NENG-1})}\times\frac{(\mathrm{FPROR}\times\mathrm{ERO}) + \mathrm{ERM}}{(1+\mathrm{FPROR})\times\mathrm{DAR}}\times 1000\times\frac{\mathrm{CPI}}{0.739}\times\mathrm{NAC}\times\mathrm{FHY} \quad (\$). \tag{7-118}$$

Where:

FPROR = the overhaul/repair ratio, i.e. the number of engines overhauled divided by the number of engines repaired. This factor is set in the LCC input file, and should be left at 0.4 unless a significant change in engine technology reduces this ratio.

DAR = the depot arrival rate, in operating hours, i.e. the total number of operating hours accumulated by the engines divided by the number of engines requiring depot (3rd Line) repair. This has been set to 250 hours, but could

be extended by de-rating of the engines. It has not been possible to make accurate estimates of the amount of extension that could be achieved by de-rating due to a lack of data.

NENG = the number of engines installed per aircraft.

Avionics contract costs have proven extremely difficult to estimate, as avionics systems have undergone major technological development in all recent aircraft programmes. This has prevented a consistent and adequate set of data from being collected, although the lack of provision of a model from Special Project Services (Cost Forecasting) at Abbey Wood in Bristol (formerly Directorate of Project Time and Cost Analysis - DPTCAn), as was promised, has also proven to be a serious setback. This has resulted in avionics O&S contract costs being based simply on uninstalled avionics mass, the number of aircraft flying hours, and consumer price index. This is clearly unacceptable, but was unavoidable due to circumstances outside the control of the author.

$$COVC = 0.5 \times NAC \times WV \times CPI \times FHY \quad (\$). \tag{7-119}$$

The lack of an avionics contract cost model may not be a major cause of error in the optimisation process, as the cost of the avionics maintenance will be driven primarily by the performance of the avionics suite, and not the aircraft. Thus, the actual LCC figures will be affected, but the optimisation may not, depending on whether the LCC study depends on the trade between costs at different stages in the Life Cycle.

The Supply contract cost captures the cost of shipping airframe, engine and avionics components from the base to the contractor, and also contains some of the costs for the supply of unit level consumption materials:

$$COSC = 0.037 \times \begin{pmatrix} COF + COPOL + COMMAT + COMSS \\ + CODLR + COAC + COPC \end{pmatrix} \quad (\$). \tag{7-120}$$

The total contract cost is then the sum of the airframe, engine, avionics, and supply components:

$$COCT = COAC + COPC + COVC + COSC \quad (\$). \tag{7-121}$$

7.7.7 Sustaining Support

Replacement support equipment cost is the cost incurred to replace equipment that is needed to operate or support an aircraft, aircraft sub-system, training system, or other support equipment. As the cost of the initial ground support equipment was based only on the production cost of the aircraft, so the replacement support equipment is similar:

$$CSER = 0.0025 \times CRF \times NAC \quad (\$). \tag{7-122}$$

The cost of Modification kit procurement includes the cost of procuring and installing modification kits and modification kit initial spares. This does not include the cost of modifications to improve the operational capability not called for in the original design or performance specification.

$$CSMK = 0.003 \times CRF \times NAC \quad (\$). \tag{7-123}$$

Sustaining Engineering Support includes all labour, material and overhead costs incurred in providing continued systems engineering and program management oversight for the aircraft system. It also includes determination of system integrity, maintenance of operational reliability, and conformance with the required specifications and standards.

$$\begin{aligned} CSSE = {} & 0.2396 \times CODLR + 0.098 \times (COAC + COPC) \\ & + 0.123 \times (COF + COPOL + COMMAT + COMSS) \quad (\$). \\ & - COSC - COUTRN \end{aligned} \tag{7-124}$$

Total sustaining support is then the sum of the costs for replacement support equipment, modification kit procurement, and sustaining engineering support:

$$CSST = CSER + CSMK + CSSE \quad (\$). \tag{7-125}$$

7.7.8 Installation Support Funds

In the US OSD-CAIG report[9], installation support funds form part of a larger heading 'Indirect Support', along with personnel support. As the personnel costs were all accounted for in earlier sections, the method presented here was felt to be more valid, although the outcome of the total O&S cost is not affected. Installation support costs are incurred by the personnel pay and allowances, material, and utilities needed for the maintenance of the base:

$$CSISF = 3565 \times \frac{CPI}{1.5884} \times NOSM \quad (\$). \tag{7-126}$$

7.7.9 Total Operation and Support Cost

The total O&S cost is simply the sum of all of the previously calculated values. For simplicity, only the values of the individual main components are added:

$$COST = COSM + CSASTT + COULCT + COCT + CSST + CSISF \quad (\$). \tag{7-127}$$

The value obtained from the above calculation is the total O&S cost for one year in 'accounting-year' dollars. As the aircraft will be operated for a life of approximately 30 years, the 'discounted' value must be added for each year to reach the true total. Alternatively, using the methods from the Government 'Green Book'[7], the through-life O&S cost can be calculated directly from the first annual cost:

$$COSTD = COST \times \frac{\left(1 - \frac{1}{(1+RED)^{TLIFE}}\right)}{RED} \quad (\$). \tag{7-128}$$

Where RED is the economic discount rate, from the LCC input file. The majority of Government projects are deflated at 6% per annum, that being the rate at which money is said to accumulate if invested, ignoring inflation. Thus, the cost for a life of thirty years at a deflated rate of 6% p.a. is found to be only 13.7648 times the first annual cost, as opposed to the 30 times that would be applied if discounting were ignored.

7.8 Disposal

As already discussed, the cost of aircraft disposal depends very much on what decision is taken regarding the disposal procedure. Some combat aircraft are sold at the end of a designated time period, even though they may have substantial airframe flying lives remaining. However, after much discussion with senior RAF Officers and costing professionals, it was decided that the disposal method to be used for this program was the break-up and disposal of the aircraft.

The chosen method of calculating disposal cost is to add the estimated cost of labour to dismantle the aircraft, the cost of disposing of the non reusable items, and subtract the value of the reusable items. The resale of systems, such as the engines and avionics, is thought to be unlikely, as technology in these areas changes so quickly that, for the moment, the value of these items has been neglected.

The disposal labour effort has been estimated as 5% of the production labour of the aircraft. Although the disposal labour rate will have increased significantly in then-year currency, this is ignored, as it is essentially the same as inflation, which is removed by presenting figures in constant economic terms.

$$CJM = 0.05 \times EPML \times CLJ \quad (\$). \tag{7-129}$$

The cost of disposal of composite materials is approximately FY83$26/tonne of material and FY83$85/skip ($26.8m^3$); the resale value of aluminium and other structural alloys has been assumed to be FY83$5/lb. The materials disposal costs become:

$$CJAMAT = (NPROD + NRDTE) \times UMATAD \times \frac{WAMPR}{2205} \times \left(26 + \frac{85}{2 \times 0.85 \times 26.8}\right) \times CPI \quad (\$), \text{ and} \tag{7-130}$$

$$CJA = -5 \times (NPROD + NRDTE) \times (1. - UMATAD) \times WAMPR \times CPI \quad (\$). \tag{7-131}$$

Where:

UMATAD is the fraction of the aircraft empty mass made up of advanced materials, WAMPR is the airframe mass, as defined in section 7.2.3, and NRDTE and NPROD are the numbers of development and production aircraft, respectively.

The total disposal cost (in future-year) dollars is the sum of the separate components:

$$CJT = CJM + CJAMAT + CJA \quad (\$). \tag{7-132}$$

The deflated disposal cost can be calculated using a separate calculation from the 'Green Book'[7] for the cost of £1 in TLIFE years time:

$$CJTD = \frac{CJT}{(1 + RED)^{TLIFE}} \tag{7-133}$$

Where RED is again the economic discount rate, as used in Equation 7-128.

7.9 Life Cycle Cost

The total Life Cycle Cost per aircraft is simply the sum of the different cost phases already calculated, apportioned to different numbers of aircraft, depending on the Life Cycle phase:

$$CLCT = \frac{(CRDTE + CPROD)}{(NRDTE + NPROD)} + \frac{CGT}{NPROD} + \frac{COSTD}{NAC} + \frac{CJTD}{(NRDTE + NPROD)} (\$). \quad (7\text{-}134)$$

Although it has not been possible to compare the results of an LCC calculation with a real aircraft, the various components of the LCC have been checked against available data. The results are of the right order of magnitude, but a definite error band cannot be given because of the lack of models in certain areas (Avionics O&S), and because of the huge numbers of possible variations in the input data. However, the shortcomings outlined in the body text should not prevent the LCC model from providing useful trade studies, and, with further development, from becoming an excellent conceptual aircraft design tool.

7.10 Summary

Chapter 7 - Life Cycle Cost Model - has:

- Presented a costing structure for the calculation of combat aircraft life cycle costs, with O&S costs mostly conforming with the US Office of the Secretary of Defense, Cost Analysis and Improvement Group, in their 'Operating and Support Cost Estimating Guide'[9]. The structure is complete, even though individual models within the structure could be improved.

- Given detailed cost estimating relationships (CER's) for each of the main life cycle phases, together with examples of accuracy, where consistent, non-sensitive data has been available.

- Provided a methodology for the estimation of the maintenance effort required to sustain an aircraft in an available state for combat. Shortcomings of the model have been highlighted, and suggestions for improvements have been given.

- Briefly explained the concept of deflating costs during the life of a project, and provided formulae for the rapid estimation of recurring costs.

8 Generation of Results

The aim of this chapter is to demonstrate the capabilities of the software tool described previously. The choice of role, capability and mission specification will have a significant effect on the size, shape, and cost of a proposed aircraft system; the numbers procured and the method of operation will also influence life cycle costs.

The first sections of the chapter present a demonstration of optimiser convergence and validate the accuracy of the aircraft sizing and cost methodologies against real data for the F-16C and Eurofighter 2000 aircraft. The remainder of the chapter is structured to mimic the selection process that would take place in the early design stages of an air combat aircraft. By using the COMBAT design and optimisation synthesis, merits of alternative configurations can be compared in a consistent and thorough manner, with visibility of configuration, performance, and cost data. This process also serves to highlight some of the factors, not included in this research, that need to be considered before the configuration choice is finalised.

A study on the effect of composite materials usage on selected candidate designs is presented, and the effect of changing the objective function from gross mass to LCC is investigated. The final study is an investigation into the effects of More Electric Aircraft technologies on the candidate designs. The results from all of the above studies are presented, and discussed where appropriate.

8.1 Introduction

Historically, aircraft configuration, planform shape, crew numbers, and other basic design options were set in the requirements phase, or early in the design process, often without detailed studies of the economic consequences of those choices. The preceding chapters have detailed the aircraft synthesis model and its operation with the LCC module and optimisation tool RQPMIN. The software tool comprising these three modules is capable of performing the studies identified, and it is the aim of this chapter to demonstrate that capability.

Before any studies are performed, optimiser convergence is presented and discussed, and the COMBAT aircraft synthesis is validated using model configurations for the F-16C and Eurofighter 2000 aircraft. Following the validation process, several conceptual configurations are generated using COMBAT, with the aim of showing how the process of selecting a concept may be improved with the new tool. Various studies of configuration options and new technologies are also presented.

In order to prevent the investigations from becoming too large and perhaps detracting from emphasis on model capability, only a single mission has been investigated. The mission chosen is a Combat Air Patrol (CAP) and air combat mission, as described in section 8.2.1. Point and mission performance, engine, procurement, and economic data are all detailed in the following sections.

8.1.1 Optimiser Convergence

Before any results of the optimisation procedure are presented, the convergence of COMBAT needs to be established, to ensure that the design variables are following a logical sequence. This confirms that the optimisation function is operating correctly, increasing confidence in the validity of the results presented in later sections.

Figure 8-1 shows the variation in the input variables with increasing numbers of *function evaluations*, i.e. calls by RQPMIN to the aircraft synthesis and cost estimating routines. It can be seen from the plot that the starting configuration variables were started at some distance from the final values (relative value = 1.0), and that a solution close to a feasible design was reached after about 150 iterations. However, this design probably did not meet all of the constraints, and would not be acceptable because the *radius of perifeasibility*, discussed in section 3.3.5, would still have been too large at that point. The optimiser can be seen to continue to investigate the gradients at the current trial point, but make sudden jumps to different points in the design space on successive Newton Iterations. In this manner, RQPMIN is ensuring that no other minima are within the radius of perifeasibility.

This process continues until, at about 850 function evaluations, the value of the objective function at the best trail point cannot be bettered using *forward differences*. At this point, RQPMIN switches to *central differences* to calculate the function gradients, which is a more accurate method, but requires twice as many function calls as forward differences. These processes are described in greater detail in section 3.3.9, and in the 'RQPMIN User Guide'[92].

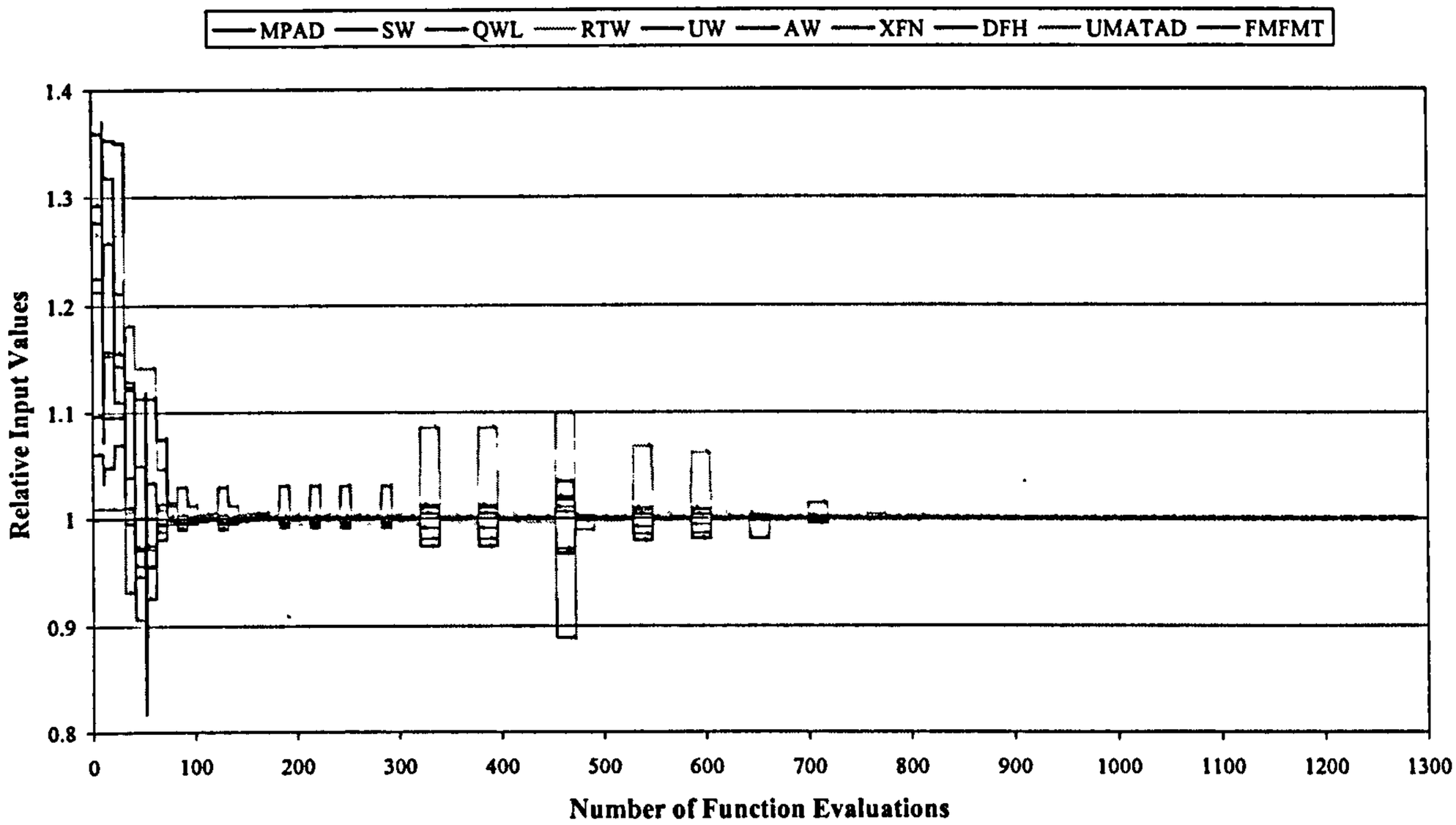


Figure 8-1. Example RQPMIN/COMBAT Convergence Plot.

Table 8-1 shows example convergence start and finish points for the LCC-optimised twin engine canard aircraft described in later sections, with Run 1 corresponding to the results shown in Figure 8-1. The optimisation was started from three different points; one much larger than the final solution, one roughly the same size as the final solution, and one smaller than the final solution. The first two optimisation runs converged to the same point, whilst the third converged, but was forced into a local minimum. It is the author's experience that optimisation runs started from configurations that are smaller than the final design are less likely to converge to what appears to be a global optimum.

	Run 1		Run 2		Run 3	
Variable	**Start**	**Finish**	**Start**	**Finish**	**Start**	**Finish**
Air Mass Flow (kg/s)	100.000	74.264	75.000	74.264	60.000	69.889
Wing Area (m^2)	61.000	47.841	50.000	47.841	40.000	47.014
Leading-edge Sweep (°)	60.000	49.616	53.000	49.616	30.000	48.981
Wing Thickness/Chord	0.0645	0.0511	0.0500	0.0511	0.0350	0.0392
Wing Aspect Ratio	3.250	2.529	2.205	2.529	2.000	2.321
Fuselage Length (m)	16.500	15.491	16.000	15.491	14.000	16.977
Fuselage Diameter (m)	2.200	1.974	2.000	1.974	1.850	1.915
Adv. Materials Fraction	34.00	33.71	34.5	33.71	25.00	34.68
Total Fuel Fraction	0.255	0.212	0.220	0.212	0.200	0.206
Life Cycle Cost		**131.789**		**131.789**		**133.073**

Table 8-1. Example Optimisation Start and Finish Values.

Optimised solutions are more likely to converge to the same point if started from a point that is close to, or larger than, the final configuration. Starting the optimisation from a point that is far from the final solution has the disadvantage that the number of iterations will probably be higher than for a run started closer to the final point. The problem of local minima is well documented for many optimisation techniques, and is to be expected for complex objective functions, such as the LCC algorithms described in Chapter 7. The only way to minimise the chances of converging to a local minimum is to start the optimisation run from a number of points, which should preferably be larger than the final configuration is expected to be. This is not always possible, especially at the start of an optimisation study when the final likely aircraft size may not be known. Initial runs will return a result that should be used to re-start the study from an over-sized aircraft concept.

The procedures to produce optimised solution were followed for all of the results presented in following sections. However, even with the precautions outlined above, some of the runs would not converge to the same point. In these few cases, the solution with the lowest value of the objective function was used. This highlights another problem with optimisation, which is the question of whether the global minimum has been found. Without exhaustive searching of the objective function from many different start points, it is very difficult to be certain of finding the global minimum. However, for the purposes of demonstrating the tool, and the level of detail of the inputs, it is felt that the optimiser is generally converging to the global minimum, or points close enough to it for the results to be meaningful.

8.1.2 Combat Validation

In order to prove the accuracy of the methodology of aircraft design and cost estimation, two real aircraft were modelled; the Lockheed F-16C, with the Pratt and Whitney F100-PW-220 engine, and Eurofighter 2000, with the Eurojet 200 engine. All of the data was taken from Jane's[39,98], with additional engine design parameters from Table 5-3. The aircraft were generated using the COMBAT aircraft design synthesis and optimisation model, although the optimisation process was not allowed to alter the design variables. In this way, the complete aircraft could be modelled and the empty mass calculated by internal iterations to reach the final value. The remaining calculated values are then compared to demonstrate the accuracy of the model.

Certain input values are not published for aircraft, because they are surrogate variables generated for the purposes of this methodology. Specifically, these are the fuselage effective diameter, and the fuel fraction. Fuselage effective diameter is the equivalent diameter of the fuselage such that the fuselage dimensions can be controlled by only one input, whilst still allowing design flexibility (see section 4.2.2). The fuel fraction is simply the maximum internal fuel mass divided by the gross mass.

The relevant data are shown in Table 8-2. Figures in bold are those input to the models to ensure that the overall configuration shapes are the same as the real aircraft. All other values are calculated from those parameters, and from the aircraft, engine, and cost data files, which use the data presented in Appendix A and Appendix B. Output data from COMBAT for the two example configurations is included at Appendix C.1.

Aircraft	F-16C			Eurofighter 2000		
	Quoted	Calc.	%	Quoted	Calc.	%
Wing Area (m^2)	**27.87**	**27.87**		**50.00**	**50.00**	
Leading Edge Sweep (°)	**40.00**	**40.00**		**53.00**	**53.00**	
Wing Aspect Ratio	**3.00**	**3.00**		**2.205**	**2.205**	
Thickness/Chord Ratio	**0.04**	**0.04**		**0.05**	**0.05**	
Wing Taper Ratio	**0.235**	**0.235**		**0.15**	**0.15**	
Fuselage Length (m)	**15.03**	**15.03**		**15.08**	**15.08**	
Effective Diameter (m)		**1.90**			**2.08**	
Tail Area (m^2)	5.92	5.93	0.17%	2.40	2.37	-1.25%
Fin Area (m^2)	4.00	4.56	14.00%	4.78	4.867	1.82%
Engine Mass Flow (kg/s)	**103.4**	**103.4**		**73.82**	**73.82**	
Engine Max. Thrust (kN)	105.7	105.5	-0.19%	88.6	89.9	1.47%
Engine Mil. Thrust (kN)	63.9	66.5	4.07%	58.4	60.1	2.91%
Engine Max. Diameter (m)	1.181	1.09	-7.71%	0.851	0.937	10.11%
Engine Length (m)	4.855	4.691	-3.38%	4.026	3.910	-2.88%
Engine Mass (kg)	1444	1477	2.29%	1012	1026	1.38%
Internal Fuel Mass (kg)	3162	3178	0.51%	4700	4703	0.06%
Aircraft Empty Mass (kg)	8663	8645	-0.21%	9999	10073	0.74%
Advanced Materials (%)		**1.72**			**33.52**	
Fuel Fraction (%)		**17.7**			**22.0**	
Excess Fuel Volume (m^3)		-0.000	0.00%		0.008	0.13%
Total Development Cost ($)				15542	14979	-3.62%
Recurring Flyaway Cost ($)	22.85M	20.81M	-8.93%	54.25M	51.65M	-4.79%

Table 8-2. Quoted and Calculated Aircraft Parameters.

The accuracy of the results is very good, with only the F-16 fin size and the engine diameters giving significant errors for the aircraft parameters. The close correlation between the aircraft empty mass, fuel mass, and excess fuel volume are significant, as this demonstrates that the aircraft are of the correct size for internal fuel storage, once the volumes of all of the internal systems have been accounted for. This shows the high levels of accuracy of all of the aircraft sizing modules, which will produce aircraft of the correct overall size. Lack of data for the aircraft design missions prevents direct comparison of those results.

Cost data are seen to have slightly higher errors for the production phases, although this could be partly due to the time differences between the quoted and calculated values - quoted in FY92$M, but calculated and compared in FY00$M. The development cost prediction for Eurofighter (compared with the 1997 Major Projects Report[29]) is seen to be excellent, varying by only 3.62%. Although the numbers involved are large, changes and restructuring to the programme have caused cost rises of more than this value.

Overall, the results are impressive, and allow high confidence in the utility of the models as the individual studies are now attempted.

8.2 Parameter Definition

8.2.1 Point and Mission Performance Parameters

To produce realistic results, a plausible set of point and mission performance parameters was required, such that a range of aircraft options could be designed to a common specification. The performance requirements are not meant to represent the specifications for any particular aircraft, either existing or under development, and were determined by the author to be representative of an air combat mission.

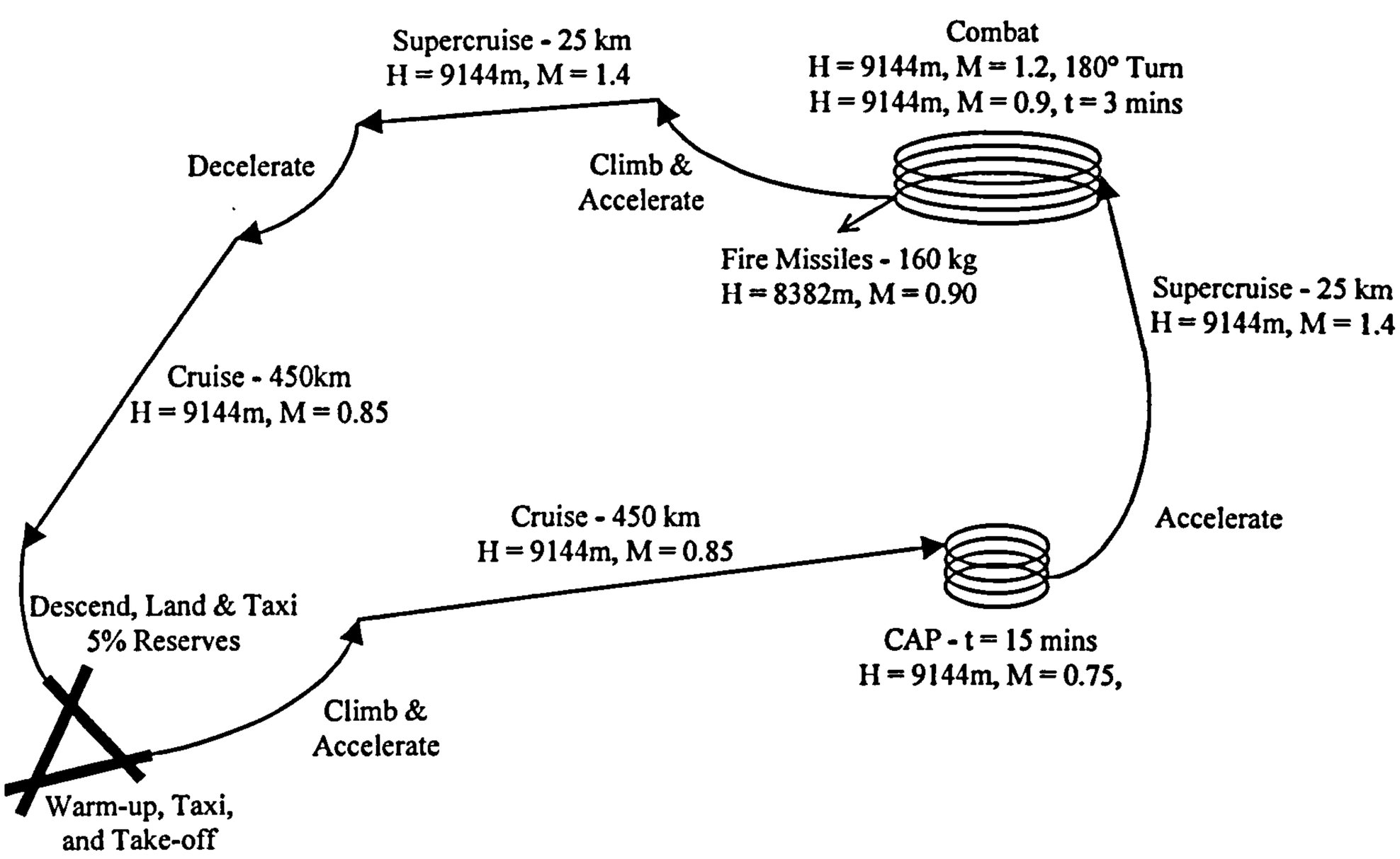


Figure 8-2. Aircraft Mission Performance Requirements.

The aircraft is required to perform the following 18-phase mission (although up to 30 segments could be specified), with a weapons load of two short-range and two medium-range air-to-air missiles, and without external fuel tanks:

- Start the engine(s) and run for one minute at maximum dry (military) power
- Take off from sea level (to 15m altitude) using reheat
- Climb and accelerate to 9144m (30000ft) and Mach 0.85 without reheat
- Cruise 450km at 9144m and M=0.85 to the combat air patrol area
- Decelerate to M=0.75 and9144m, and remain on CAP for 15 minutes

Following the detection of a new threat, the aircraft is required to visually confirm the identity of the threat, and engage if necessary:

- Accelerate at maximum wet power to M=1.4 and 9144m
- Supercruise 25km at M=1.4 and 9144m to identify the threat

The threat is found to be 'unfriendly', and the aircraft must engage in air combat:

- 180° sustained turn at 9144m and M=1.2 – full reheat
- Air combat at M=0.9 for 3 minutes – full reheat.
- Fire two short-range air-to-air missiles (mass 160kg) at M=0.9 and 8382m (27500)ft

Having defeated the enemy, the aircraft must retreat quickly, and return to its base:

- Climb/accelerate at maximum wet power to 9144m and M=1.4
- Supercruise 25km at M=1.4 and 9144m to evade any other local threats
- Decelerate to M=0.85 and 9144m
- Cruise 450km at 9144m and M=0.85
- Descend and land

After taxiing and ground movements equivalent to 1 minute at maximum dry power, the mission is completed with 5% reserve fuel remaining.

The point performance constraints for the aircraft are as follows; weapons load refers to the mission weapons load (normally 475kg, or 600kg for internal carriage):

- Take-off and Landing at sea level with 95% fuel and 100% weapons load with ground-roll distances of 250m and 500m respectively.
- Attained turn load factor of 7.0g at 12750m (41830ft) and M=1.4, with 50% fuel and 100% weapons load.
- Attained turn load factor of 7.5g at 10805m (35450ft) and M=0.9, with 50% fuel and 100% weapons load.
- Sustained turn load factor with reheat of 5.0g at 6050m (19850ft) and M=0.8, with 50% fuel and 100% weapons load.
- Sustained turn load factor with reheat of 3.0g at 12825m (42075ft) and M=1.6, with 50% fuel and 100% weapons load.
- Maximum level flight speed of M=2.0 at 11500m (37730ft), with 50% fuel and 100% weapons load.
- Maximum level flight speed of M=1.2 at sea level, with 80% fuel and 100% weapons load.
- Climb/accelerate from M=0.25 at sea level to M=1.5 at 9144m (30000ft) in 1.5 minutes, with initial fuel and weapons loads of 50% and 100% respectively.

Maximum limit design load factor was set at 9g, and ultimate load factor was calculated according to the configuration (see section 4.3.1). A mass of 100kg per crew member is used for all configurations, but cockpit volume varies as described in section 4.3.4.

8.2.2 Engine Parameters

The engine input variables and thermodynamic cycle were based loosely on data, published and assumed, for the EJ200 engine used in the Eurofighter 2000 multi-role combat aircraft, and have been kept fixed for these studies.

The engine is a twin-spool mixed exhaust turbofan engine with reheat, having a two-shock inlet and a convergent-divergent (con-di) nozzle system. Bypass ratio is 0.41, maximum compressor pressure ratio is 26, and fan pressure ratio is set to 4.45 at the design point (sea-level static conditions). Maximum turbine inlet temperature (T_{T4}) is 3120K, whilst the maximum temperature in the flame reheat duct is 3800K.

Bleed and power off-take values have been set to representative levels; bleed air is 0.85% of the core flow, high-pressure (HP) turbine cooling flow is set to 5% of core flow, and the low-pressure (LP) turbine is uncooled. Power off-take from the LP shaft has been assumed to be zero, whilst the HP shaft provides approximately 75kW of power for aircraft and engine systems, scaled with engine mass flow rate.

Component pressure ratios contribute to the efficiency of the engine, and were set to 0.97 for the diffuser, 0.943 for the burner, 0.943 for the afterburner, 0.97 (maximum) for the mixer, and 0.957 for the nozzle. These figures were interpolated from the relevant cells of Table 5-3, section 5.2, assuming that the IOC date places the engine one-third of the way between Technology Levels 3 and 4. Polytropic efficiencies for the turbomachinery components are set at 0.85 for the LP compressor (fan), 0.883 for the HP compressor, 0.875 for the HP turbine, and 0.893 for the LP turbine.

The final set of component design data determines mechanical and thermodynamic efficiency. Thermodynamic efficiencies of the burner and afterburner are set at 0.983 and 0.95 respectively, whilst the HP and LP spools have mechanical efficiencies if 0.98 and 0.99 respectively. Power take-off mechanisms have efficiencies of 0.98 for both LP and HP spools, although the LP off-take is not used in this design. The ratio of theoretical flow exit pressure to flow entry pressure (P0/P9) is set to 1.0, and the nominal Mach number at station 5 (core stream mixer entry) is 0.4.

All of the figures above refer to the aircraft at the design condition; away from the design point, the engine parameters will vary by a wide margin. In order to prevent engine performance from becoming unrealistic, constraints are applied to some of the parameters above. T_{T4max} is set at 3200K, T_{T7max} to 3800K, T_{T3} to 1630K, and maximum compressor pressure ratio to 26.0.

The above inputs result in an engine with a maximum specific thrust of 1.218kN/kg/s with reheat, and 0.815kN/kg/s at maximum dry throttle, which is typical for a modern high-performance military aircraft engine. Figure 8-3 shows Maximum and Military specific thrust against year of initial operational capability (IOC), and Figure 8-4 shows Military and Maximum TSFC against IOC Year. SLS engine thrust specific fuel consumption (TSFC) data is calculated as 47.8mg/Ns at maximum (wet) thrust, and 22.7mg/Ns at maximum military (dry) thrust, compared with quoted figures[39] for EJ200 of 47.7mg/Ns and 22.6mg/Ns for maximum and military thrust, respectively.

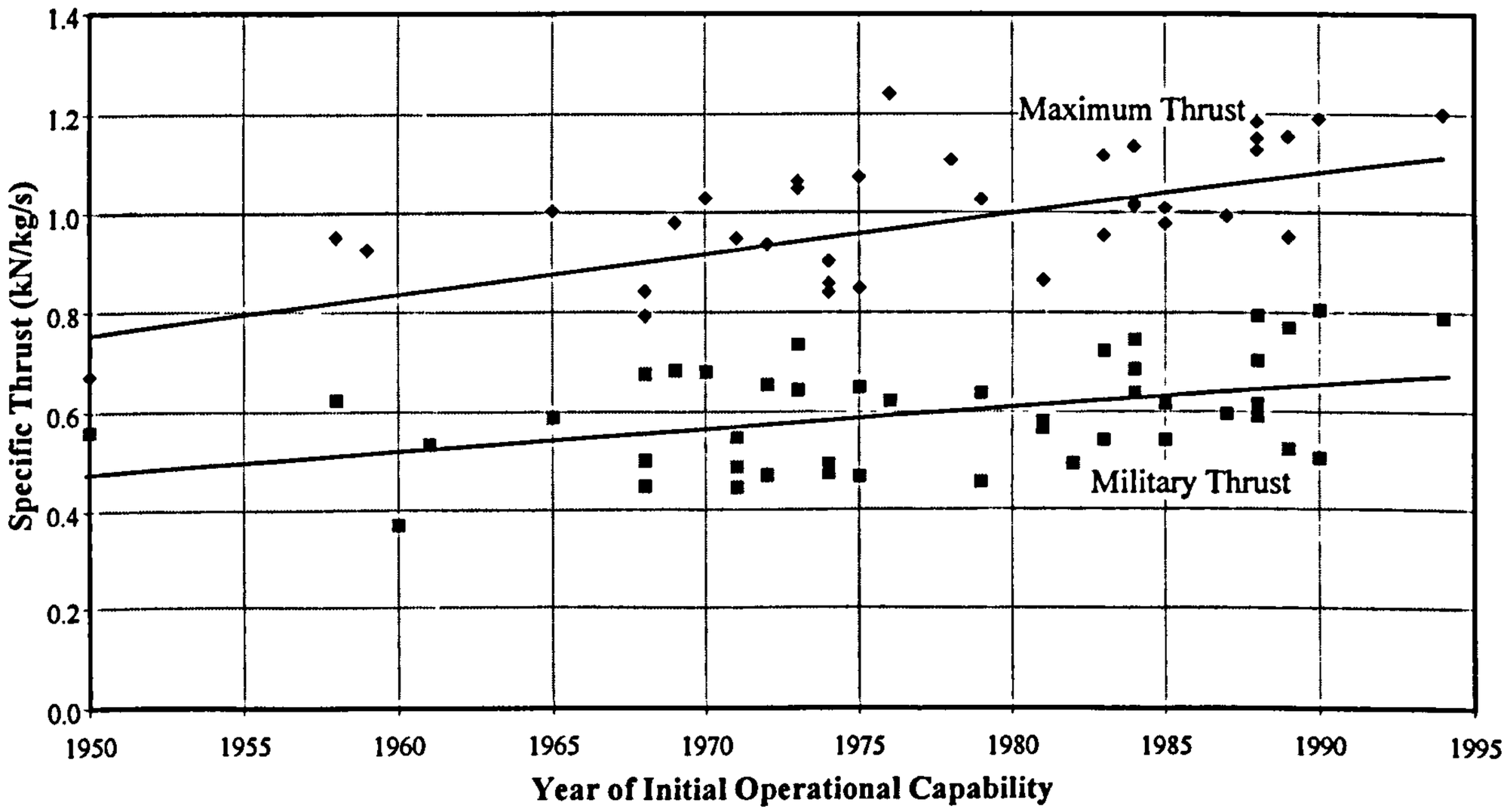


Figure 8-3. Maximum and Military Specific Thrust vs. IOC Year.

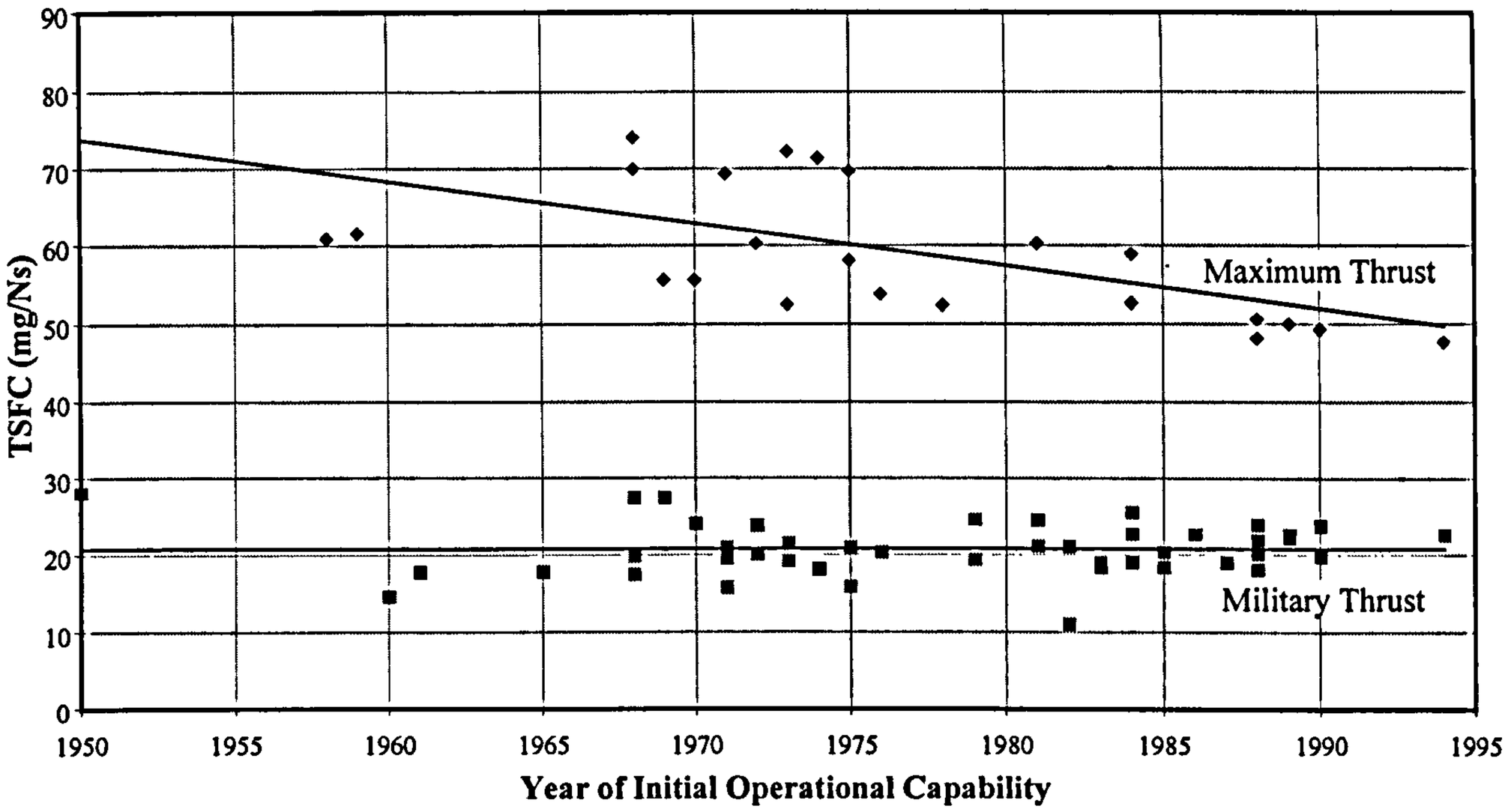


Figure 8-4. Maximum & Military TSFC vs. IOC Year.

The calculated uninstalled performance figures for the example engine are seen to be entirely plausible, and form a good basis for the remainder of the study to take place.

8.2.3 Procurement and Economic Parameters

The final aircraft definition input file required by the synthesis model is the LCC data file. For the purposes of these studies, it has been assumed that nine development airframes will be built, two of which will be static test airframes. Production will be by an International consortium, and the largest procurement will be 232 aircraft out of a total of 620, giving 2.672 effective partners. The production rate for the aircraft will be 10 per month; i.e. production is completed in 62 months.

Factors for the design and production phases are; security (1.2), design tool factor (1.3), and the advanced technology flight test factor (1.2). These are the figures used to generate the data for Eurofighter 2000 (now Typhoon), presented in Figure 7-4. The aircraft will not include very low-observable technology, as this could have a significant impact on the design and costs of the resulting configuration. Instead, some low-observable technology will be included, resulting in a low-observable factor of 1.2.

The year of Initial Operational Capability is set to 1996, although the fiscal year is 2000. The Consumer Price Index (CPI), i.e. the ratio of cost of finished goods in 2000 to 1983 is found to be 1.777, and the economic discounting rate is set at 6%. Fuel cost is set to $0.50 per litre (including delivery and storage charges).

The aircraft will operate for a period of 25 years, flying an average of 300 hours per year. They will be operated from Main Operating Bases, holding 3 squadrons of 13 aircraft each, and each aircraft will have 1.25 aircrew per seat. The factor for the rate of engine overhaul/repair ratio, as used in the calculation of engine contract costs (Equation 7-118) is set at 0.4. This has been found to be the right order of magnitude from US VAMOSC data.

8.3 Configuration Options

One of the most obvious choices in the design of any aircraft is the overall configuration of the airframe. The aircraft synthesis methodology described in earlier chapters is capable of modelling three main types of aircraft configuration; aft-tail, delta, and canard-delta. In order to select the aircraft having minimum life cycle costs for the specified parameters, several optimised designs must be generated before manual comparison to determine the most appropriate configuration.

LCC-optimised configurations were generated for single and twin-engine versions of aft-tail, delta, and canard aircraft, the latter having options of 1 or 2 crew, and 1 or 2 fins. The reason for the secondary choices was to investigate the effects of crew numbers as, until recently, RAF policy was normally to have two-man crews. This study will determine the economic consequences of that policy, so that cost/benefit analyses could be performed, although this would rely heavily on 1-vs.-2 crew mission effectiveness studies, and are not part of this research. Two fins may become necessary for either stealth or high AoA reasons; the inclusion of twin-fin options gives an indication of the implications of these choices, allowing improved decision-making.

8.3.1 Single Engine Configurations

Figure 8-5 shows the LCC-optimised single engine configurations generated using the COMBAT output data presented in Appendix C.1. Only single fin, single seat options are presented. The aft-tail aircraft is not a fully converged solution, and the shape is not an elegant design; the reasons for this are discussed in later paragraphs.

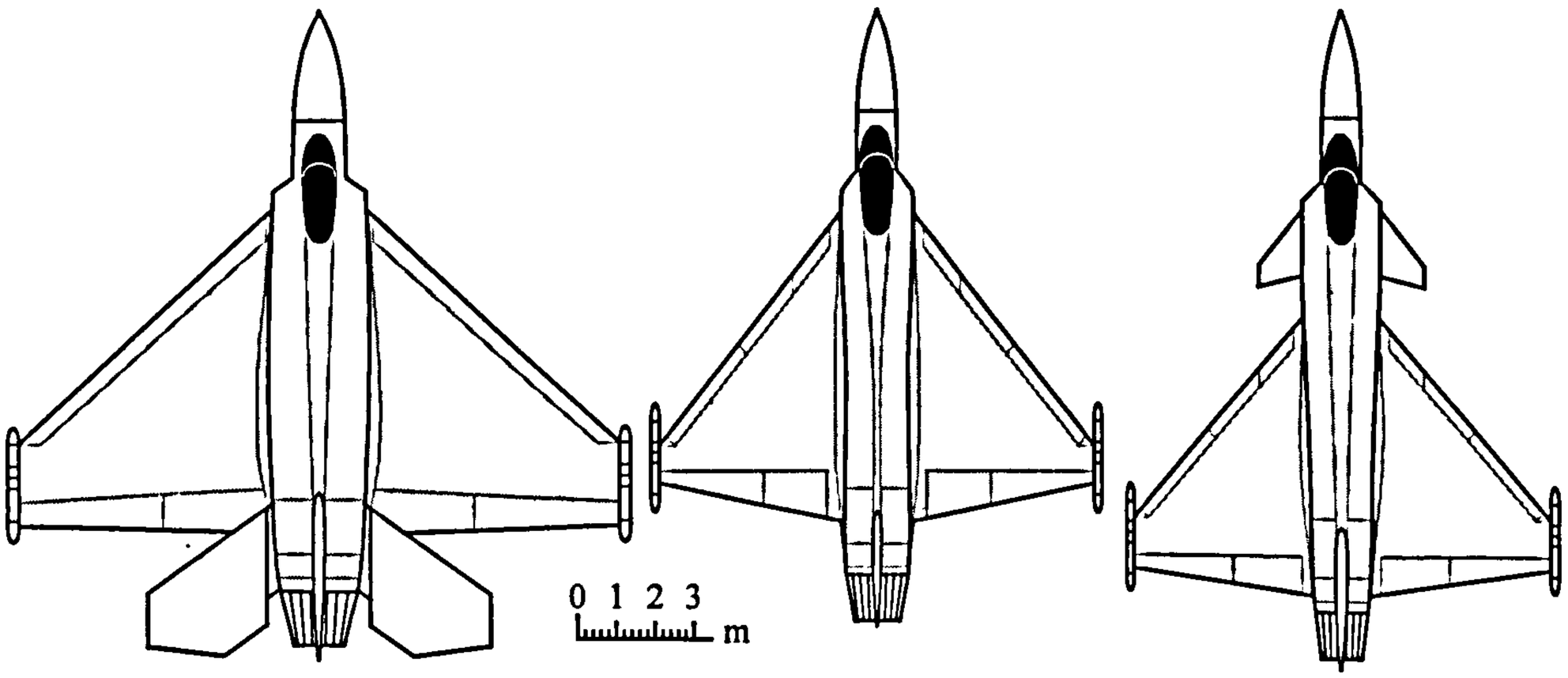
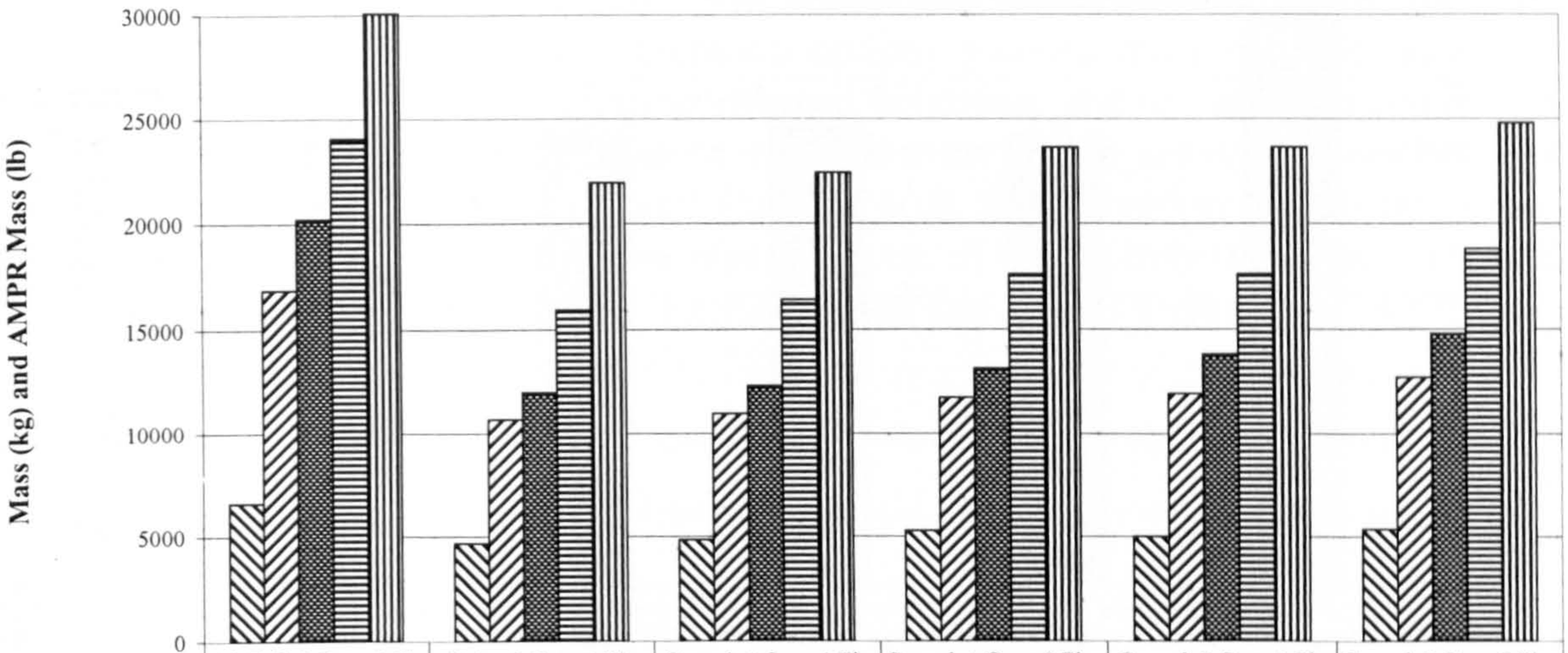


Figure 8-5. Single Engine Configurations.

Figure 8-6 shows aircraft fuel, empty, AMPR, mission, and gross mass values for the above aircraft configuration solutions. As can be seen, the delta has the lowest mass, due to the lack of mass and drag from the horizontal tail surface. The canard configurations follow, increasing in order of gross mass, and the equivalent canard aircraft (one crew, one fin) is found to have empty mass and gross mass increases of only 2.76% and 2.17%, respectively, over the delta aircraft.



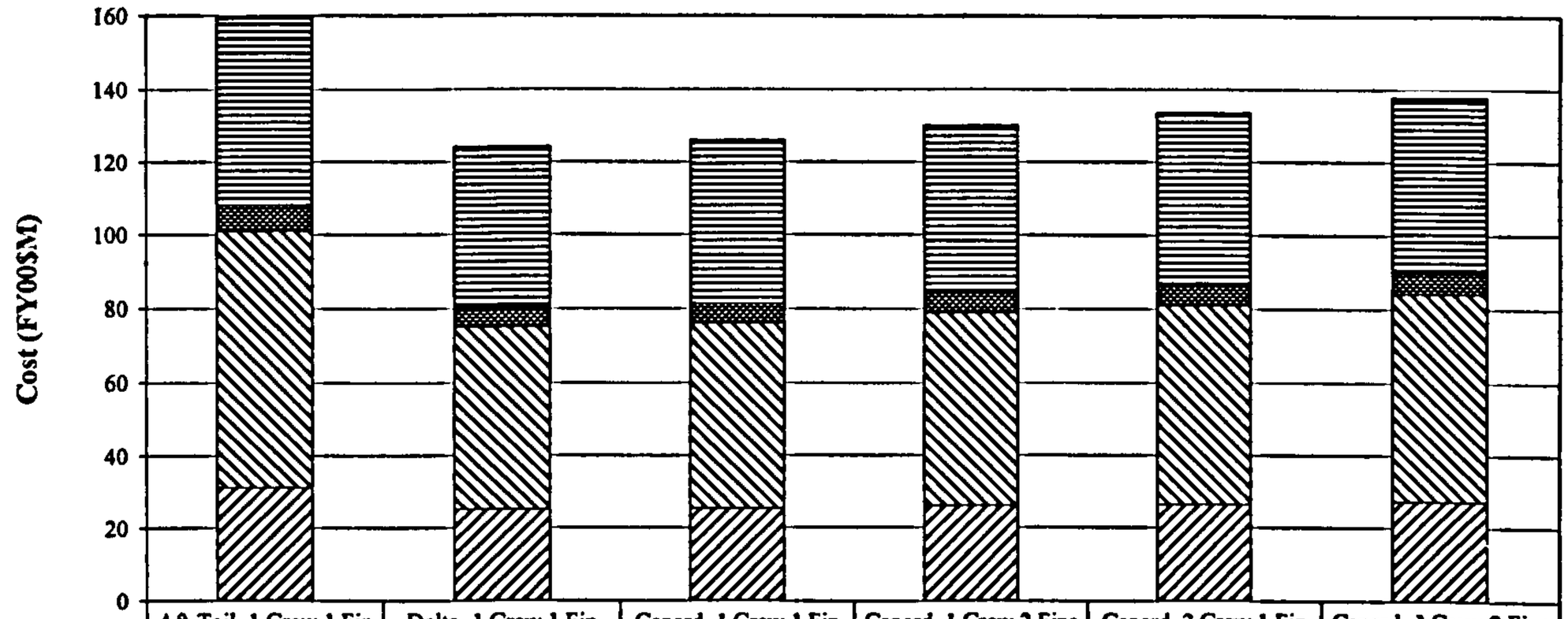
	Aft-Tail 1 Crew 1 Fin	Delta 1 Crew 1 Fin	Canard 1 Crew 1 Fin	Canard 1 Crew 2 Fins	Canard 2 Crew 1 Fin	Canard 2 Crew 2 Fins
Fuel Mass	6610	4674	4857	5271	4996	5367
Empty Mass	16877	10679	10974	11754	11931	12742
AMPR Mass (lb)	20201	11984	12311	13159	13812	14807
Mission Mass	24062	15928	16406	17600	17602	18784
Gross Mass	30087	21953	22431	23625	23627	24809

Figure 8-6. Single Engine Configuration Masses.

Increasing the number of crew and/or fins for the canard configuration produces predictable results, with the two-crew, twin-fin aircraft suffering mass increases of 19.3% and 13.0% for empty and gross mass, respectively. Although not intended to fulfil the same aim, it is interesting to note that addition of a fin or extra crew member resulted in almost identical aircraft mission mass. The difference between the fuel and empty masses is significant; fuel mass for the twin-fin aircraft increased by a larger amount, due to its higher drag coefficient, although this is offset by lower empty mass.

The aft-tail aircraft fares much worse with the demanding mission and point performance requirement described earlier, with empty mass and gross mass increases of 58.0% and 37.1%, respectively. Although the results are presented with the other configuration options, this configuration failed to meet some of the point performance requirements, and is thus not a converged solution. The aircraft shown fails to meet both attained turn requirements, the climb/acceleration constraint, and had a very small deficiency in its mission fuel volume. Although these constraints have been violated, the solution was close enough to the specification to prevent its complete exclusion. As seen in Figure 8-5, it looks the least plausible of all of the configurations.

Figure 8-7 shows the above configurations in terms of their discounted LCC breakdown. The delta aircraft is seen to be the cheapest option, as well as having the lowest mass, and this will be helped by the delta's reduced support costs due to its lower maintenance requirement, caused by having fewer control surfaces. As expected, the significant increase in size of the aft-tail aircraft has resulted in much higher costs, although the cost increases (28.5% of LCC) are lower than the proportional mass increases (37.1% gross mass). The costs of the canard aircraft have also increased, but, again, the cost increases are lower than the equivalent mass increases. The single-crew, single-fin canard aircraft is only 1.4% more expensive, in LCC terms, than the delta (gross mass increase 2.17%).



	Aft-Tail 1 Crew 1 Fin	Delta 1 Crew 1 Fin	Canard 1 Crew 1 Fin	Canard 1 Crew 2 Fins	Canard 2 Crew 1 Fin	Canard 2 Crew 2 Fins
Total	159.493	124.094	125.833	129.855	133.377	137.843
Disposal	0.286	0.180	0.184	0.194	0.205	0.218
O&S	51.115	43.702	44.280	45.344	46.620	47.650
GSE&IS	7.086	5.081	5.166	5.380	5.537	5.784
Production	69.844	50.087	50.918	53.030	54.573	57.011
RDT&E	31.162	25.043	25.285	25.906	26.443	27.181

Figure 8-7. Single Engine Configuration Cost Breakdown.

The cost of the canard 'enhancement' aircraft increases with the gross mass, although the cost increase is less than the mass increase in every case. Whilst not related, it is interesting to note that the addition of the extra crew member causes a larger increase in cost than the addition of the extra fin, as a result of the larger empty and AMPR masses.

8.3.2 Twin Engine Configurations

The same set of configuration options was run as for the single engine aircraft; i.e. single crew, single fin aircraft for aft-tail, delta, and canard configurations, with the latter having a combination of two crew and twin fin options. Figure 8-8 shows the planforms resulting from the COMBAT output data in Appendix C.3 for the single seat, single fin, twin engine configuration options.

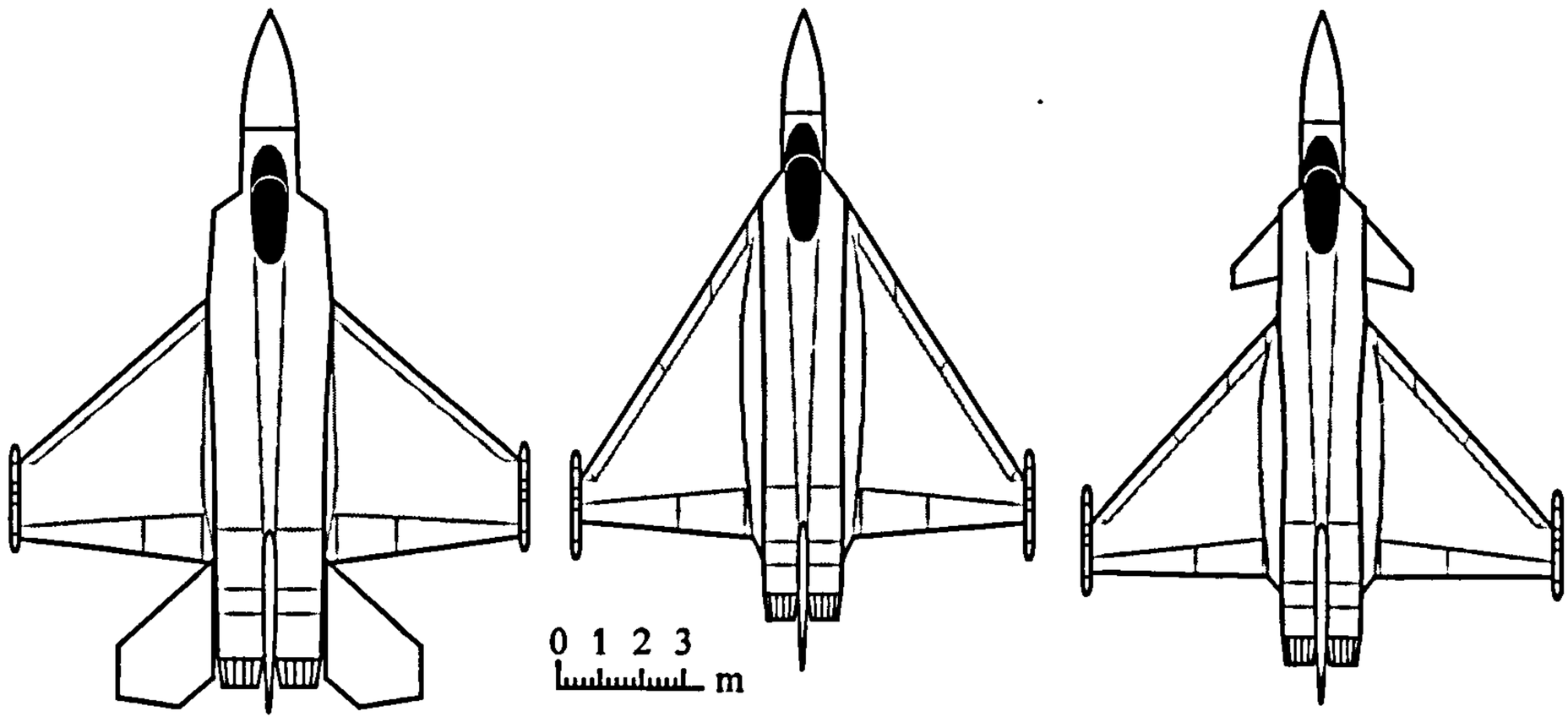
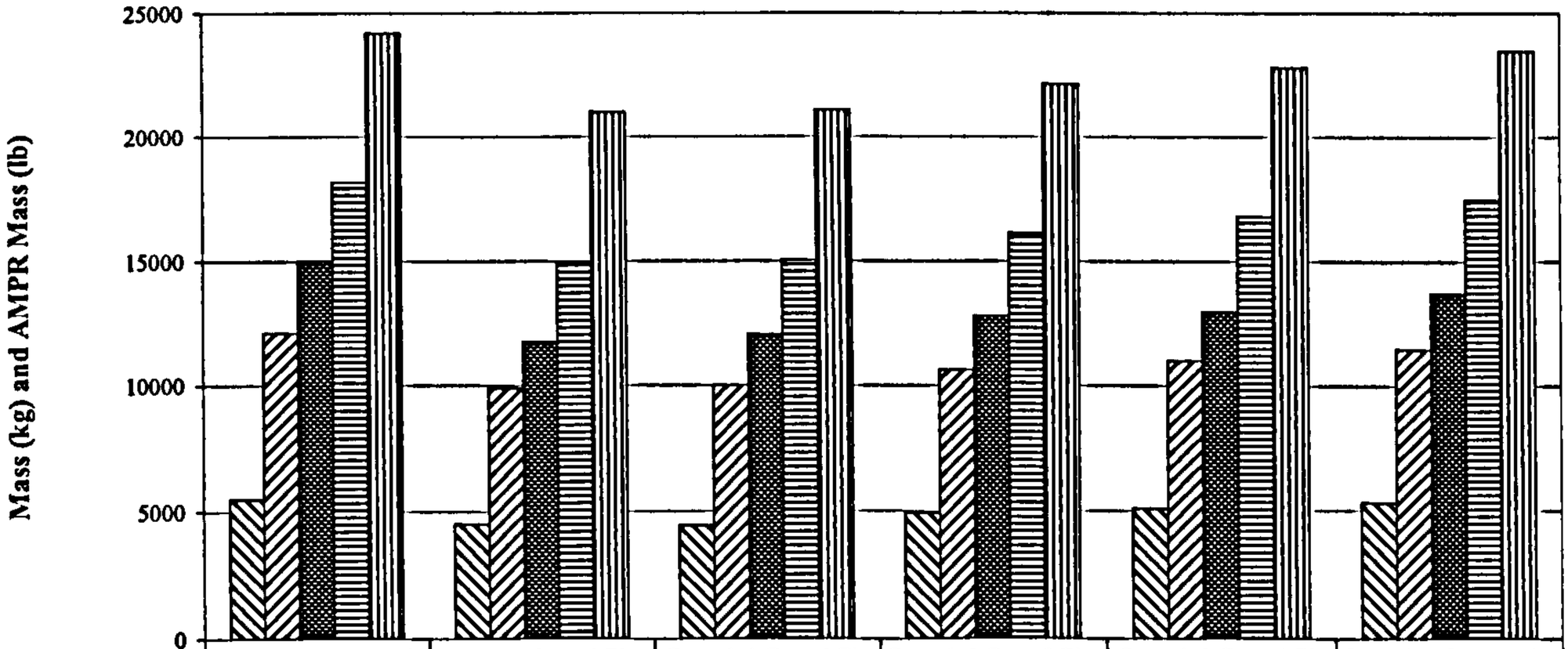


Figure 8-8. Twin Engine Configurations.

The results of the mass totals for the twin engine configurations are shown in Figure 8-9. The twin engine aircraft configuration options produce more realistic results, in that all of the results consist of fully converged solutions, and the aft-tail aircraft now meets all of the performance and mission requirements. The mass differences between the competing designs have narrowed slightly, with the exception of the two crew, single fin canard option. It has a gross mass increase of 8.62% over the delta, compared with the corresponding single-engine aircraft, which had a mass increase of 7.62%.

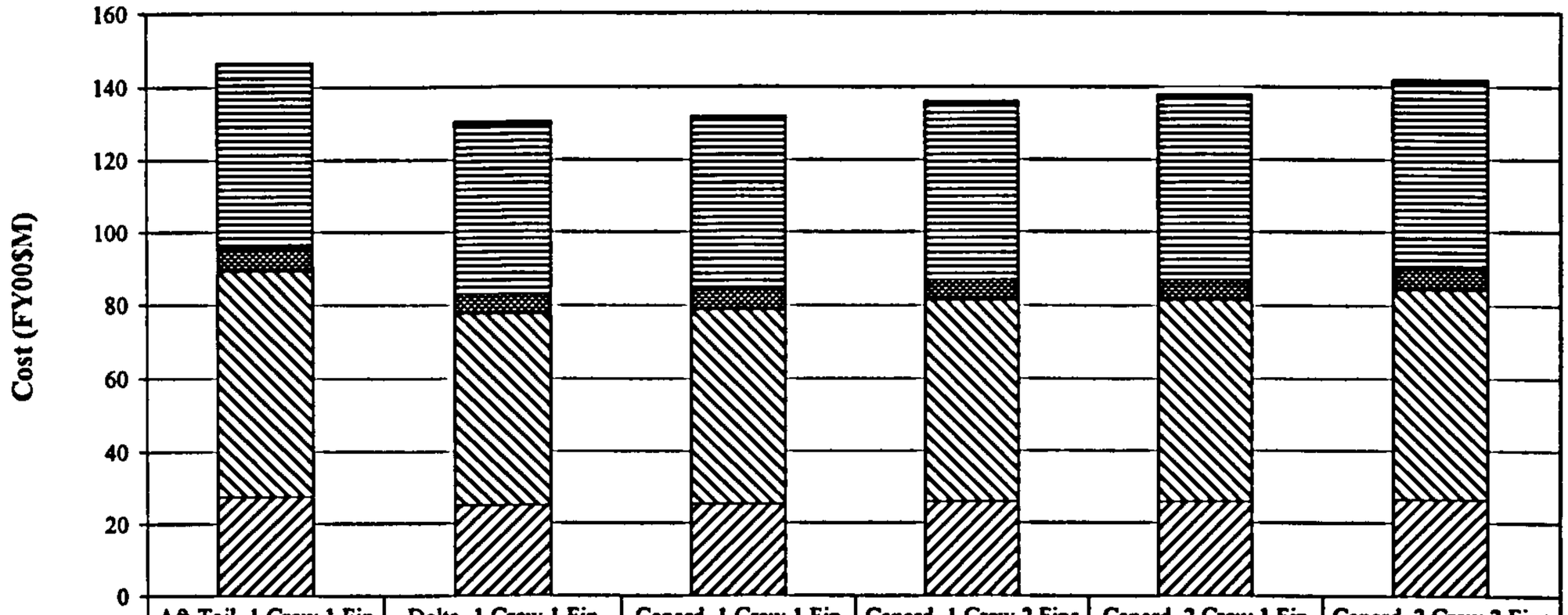


	Aft-Tail 1 Crew 1 Fin	Delta 1 Crew 1 Fin	Canard 1 Crew 1 Fin	Canard 1 Crew 2 Fins	Canard 2 Crew 1 Fin	Canard 2 Crew 2 Fins
Fuel Mass	5475	4509	4457	4921	5095	5304
Empty Mass	12113	9880	10001	10632	11003	11470
AMPR Mass (lb)	15027	11760	12068	12802	12977	13718
Mission Mass	18162	14964	15033	16127	16773	17449
Gross Mass	24187	20989	21058	22152	22798	23474

Figure 8-9. Twin Engine Configuration Masses.

Of particular interest is the gross mass difference between the delta and the equivalent canard aircraft, which has narrowed to just 0.33%, although the empty mass difference is higher, at 1.22%. The major difference between these two aircraft is that the canard has slightly smaller engines (by 0.63%), which was not the case for the single engine configuration comparison, and leads to a reduction in internal fuel mass of 52kg (1.15%). Whilst not significant in itself, further investigation revealed that the reason for the differences between the two aircraft is that the performance constraints driving the designs are different. The delta is driven mainly by the second attained turn, first sustained turn requirement, and the time to climb constraint, whilst the canard design is driven mainly by the landing distance, and the time to climb.

The influence of these constraints results in aircraft that make use of different configuration attributes to reach the same overall performance. Under the confines of this study, the perceived detrimental effect of adding foreplanes is much less severe than may have otherwise been thought. It should be noted that relaxation of the second attained turn constraint (7.5g at M=1.4 and 12750m) would result in a smaller delta aircraft, but that the canard configuration option would remain unchanged. This highlights the need for careful choice of point performance specifications during the requirements definition phases.



	Aft-Tail 1 Crew 1 Fin	Delta 1 Crew 1 Fin	Canard 1 Crew 1 Fin	Canard 1 Crew 2 Fins	Canard 2 Crew 1 Fin	Canard 2 Crew 2 Fins
Total	146.599	130.390	131.789	135.849	137.884	141.888
Disposal	0.227	0.181	0.186	0.195	0.194	0.206
O&S	51.107	47.169	47.381	48.639	50.651	51.549
GSE&IS	6.243	5.352	5.436	5.643	5.646	5.872
Production	61.542	52.756	53.587	55.627	55.657	57.876
RDT&E	27.481	24.932	25.199	25.744	25.736	26.385

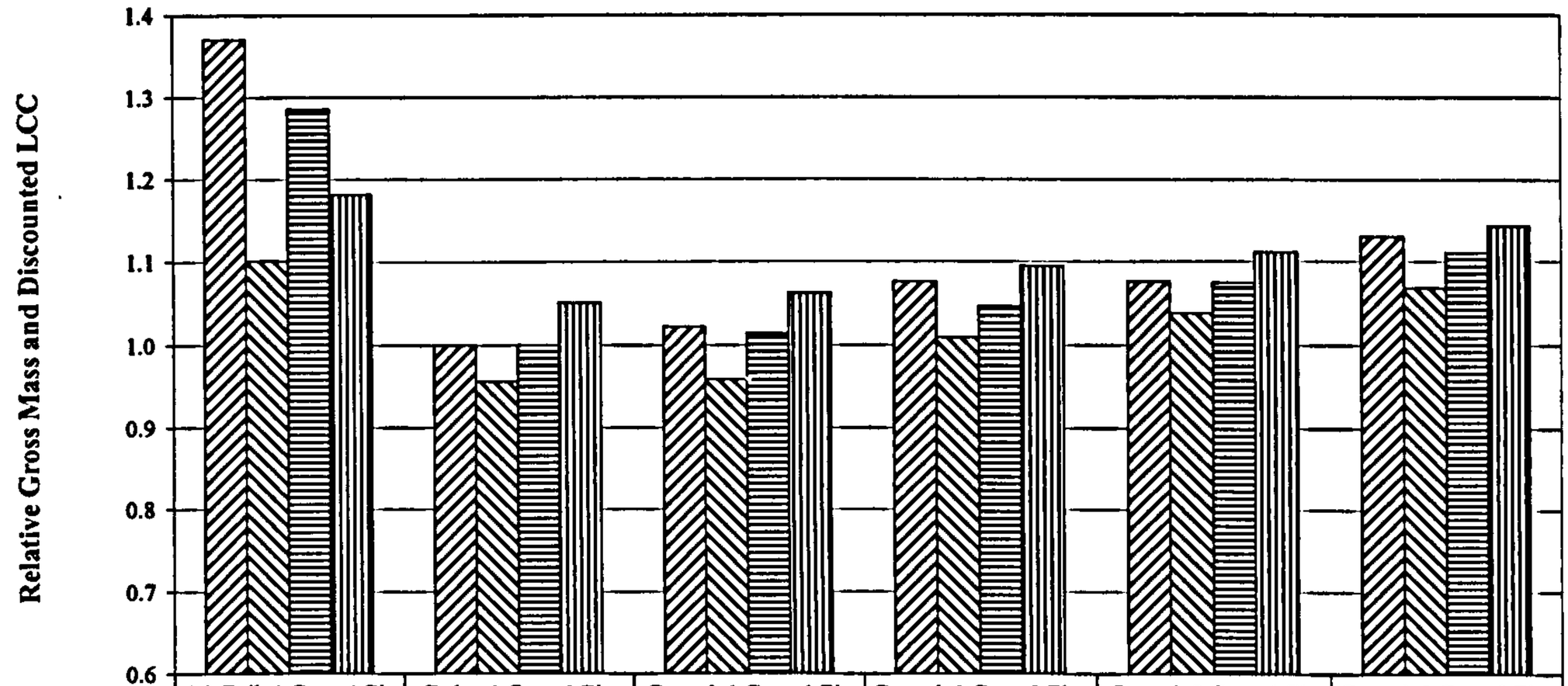
Figure 8-10. Twin Engine Configuration Cost Breakdown.

Figure 8-10 shows the cost breakdown for the twin-engine configuration options. The trend across the options is seen to be very similar to that for the single engine configurations and, with the exception of the first canard option, the cost increases are again proportionally lower than the equivalent aircraft mass increments. The delta configuration is seen to be the cheapest option for both single and twin engine configurations, although the cost of the equivalent canard aircraft is now much more comparable, having risen only 1.07%. The single-crew, single-fin canard aircraft breaks the cost/mass increment trend primarily because of its very low gross mass increase, as described above.

8.3.3 Single Engine vs. Twin Engine Comparison

In order to directly compare the effects of changing the number of engines, the results of the two studies have been collated. They are presented in Figure 8-11 as relative mass and cost values, compared with the single engine delta configuration. It can be seen that for every equivalent aircraft configuration (except the single engine aft-tail aircraft, which is not a converged solution), the mass of the twin engine aircraft is lower, but the discounted LCC has increased. This is an interesting result for two reasons: for a given set of requirements, it is generally perceived that a single engine aircraft will be smaller and lighter than a twin engine configuration, and; it shows that for significant configuration changes cost is not always proportional to mass.

The difference between the masses of single and twin engine configuration options is due to two effects highlighted by the engine sizing methodology, detailed in section 5.2.4. It would appear that for a given thrust requirement, two smaller engines result in a lower total propulsion system mass and a lower total volume requirement than a larger single engine. This appears counter-intuitive, as the two-engine solution appears to have doubling of many components – gearboxes, housings, fuel pumps, etc.



	Aft-Tail 1 Crew 1 Fin	Delta 1 Crew 1 Fin	Canard 1 Crew 1 Fin	Canard 1 Crew 2 Fins	Canard 2 Crew 1 Fin	Canard 2 Crew 2 Fins
Mass - Single	1.371	1.000	1.022	1.076	1.076	1.130
Mass - Twin	1.102	0.956	0.959	1.009	1.038	1.069
LCC - Single	1.285	1.000	1.014	1.046	1.075	1.111
LCC - Twin	1.181	1.051	1.062	1.095	1.111	1.143

Figure 8-11. Relative Mass and LCC – Single vs. Twin Engine.

Using the data for the single-crew, single fin canard aircraft, the results shown in Table 8-3 are found using the methods presented at section 5.2.4. The 'Single Engine' results refer to the data for the single-engine aircraft, whilst 'Twin Engine 1' shows the effect of changing from one large engine of 160.8 kg/s air mass flow rate, to two smaller engines, each of 80.4 kg/s. 'Twin Engine 2' shows the data for the real twin engine canard aircraft, which has been re-sized to take advantage of the reduced mass and volume of the two-engine solution. Below the calculated results, data are presented for two Russian engines, the Aviadvigatel D30-F6 from the MiG 31, and the Klimov RD33 from the MiG-29, with the percentage changes being presented relative to a single D30F6. In all cases, results for twin engine solutions are presented as the total for the two engines, except for maximum length and diameter values.

The Aviadvigatel and Klimov engines have been chosen because they have similar overall design parameters, were both introduced within one year of each other, and have an air mass flow rate ratio of very close to 2:1. Data for the two engines can be found in Appendix B.

Engine	Air (kg/s)	T_{MAX} (kN)	T_{MIL} (kN)	D_{MAX} (m)	L_{MAX} (m)	Mass (kg)	Vol. (m^3)	% Mass	% Vol.
Single Engine	160.8	195.9	131.0	1.321	6.328	2807	8.678	0.0	0.0
Twin Engine 1	160.8	195.9	131.0	0.974	4.124	2281	6.147	-18.7	-29.2
Twin Engine 2	148.6	180.9	121.0	0.940	3.924	2067	5.452	-26.4	-37.2
Aviadvigatel D30F6	150.0	186.1	93.2	1.282	7.040	2416	9.085	0.0	0.0
Klimov RD33	152.0	162.8	98.8	1.000	4.229	2110	6.643	-12.7	-26.9

Table 8-3. Single vs. Twin Engine Mass and Volume Comparison.

These data would seem to support the theory that twin engine configurations should be smaller and lighter than an aircraft powered by a larger single engine of equivalent thrust. However, the difference in the thrust/weight ratio for the two engine options - 7.16 for the large engine, as opposed to 8.92 for the smaller engine - is a cause for concern, given that the engines are identical in all respects other than the mass flow rate. Despite a thorough review of the engine mass prediction model, and the engine data presented in Appendix B, no reason can be found for the apparent discrepancy. That said, even if the mass prediction algorithms were erroneous, the lower volume of the twin engine design would still offer a benefit, as over 2.5m^3 of volume (before scaling) is made available, in this example, by changing from single to twin engines.

Despite a reduction in aircraft mass, the LCC for the equivalent twin engine aircraft is higher. Development costs are reduced between 0.3% and 2.9% in changing from single to twin engine designs, mainly as a result of the lower airframe mass, although this is offset slightly by higher advanced materials fractions and the related increase in the advanced technology cost factor. Production and GSE&IS costs increase between 1.5% and 5.3% for the different configurations, mainly as a result of the increased cost of engine production. As with the development phase, changes in the fraction of advanced materials and AMPR mass values between the designs also contribute, leading to some twin engine designs having costs closer to the equivalent single engine options, although engine costs and total production costs are higher in every case.

The largest cost component increase is in Operation and Support (O&S) costs, which rises between 7.0% and 8.6% for the various configuration options. This is primarily caused by the very large increase in engine contract costs caused by having two engines instead of one. Although the cost per engine is less for the smaller engines, this is overwhelmed by doubling the number of engines. This may be expected, as the labour cost will probably double for engine overhaul, and spares cost, whilst lower, will certainly not halve in cost. Many of the other O&S costs are driven by the aircraft maintenance requirement, for which mass is a significant driver, and subsequent scaling of personnel numbers, which is strongly influenced by the number of aircraft on the operating base.

Disposal costs make up only a very small fraction of the total LCC, once the discounting process has been applied. For this reason, disposal cost will not have a significant impact on discounted LCC, unless a completely different method of disposal is adopted, e.g. re-sale instead of aircraft break-up and component disposal. One of the major drivers of disposal cost, apart from aircraft mass, is the advanced material fraction, which affects the airframe disposal cost. However, the disposal cost difference between aircraft that make use of advanced materials and those that do not is only in the order of 11%, or approximately 0.01% of discounted LCC.

8.3.4 Initial Configuration Selection

The previous sections have shown that for a given size, aft-tail aircraft configurations do not have sufficient manoeuvre performance to compete, in cost terms, with the delta and canard-delta designs. This is particularly true for the single engine aft-tail aircraft, which was not capable of meeting the performance requirements, and consequently would not produce a converged result. The difficulties the aft-tail options have in meeting the mission requirements are due to the reduction in total lift of the aircraft for high-g instantaneous manoeuvres, as described in section 6.1.3. These deficiencies, coupled with the very demanding time-to-climb requirement, cause the size of this aircraft configuration to spiral upward, increasing mass and LCC.

Although the twin engine aft-tail aircraft is probably just as capable as the canard or delta configurations, and will probably have a configuration advantage in terms of RCS, the cost increases are felt to be disproportionately large. For the total procurement of 620 aircraft, the total difference in life cycle cost between the aft-tail aircraft and the delta configuration, which is the cheapest option, would be FY00$10 billion. Lacking a very strong case for the selection of this aircraft over the other options, it is not felt to be a cost-effective option. Therefore, the aft-tail configuration has been rejected as an option for further studies.

If the primary role for the aircraft is air combat, as specified in the mission profile, another factor that becomes increasingly important is aircraft agility, which is discussed comprehensively by Kalviste[53] and Skow[90]. Agility does not appear to have a single, clear definition, but can be thought of as 'the rate of change of manoeuvre', or 'the ability of an aircraft to shift from one manoeuvre to another in minimum time', or 'the ability to rapidly change both the magnitude and direction of the velocity vector'. The turn performance calculations in COMBAT are based only on steady-state manoeuvres, using a combination of planform type, wing area, and aircraft maximum lift coefficient (for attained turn rates), or aerodynamic efficiency and engine thrust (for sustained turn rates). Neither of these is a measure of the proficiency of an aircraft at transient manoeuvring between steady-state conditions, i.e. its agility.

The agility of an aircraft will be determined primarily by the level of control authority available, which in turn is determined by the number of control surfaces, and the effectiveness of each of those surfaces in changing the aircraft flight vector. Canard and aft-tail aircraft have a double advantage over deltas in that they have extra control effectors (horizontal control surfaces), and larger moment arms between those effectors and the aircraft c.g. The larger pitching moments that the horizontal control surfaces can generate will enable greater rotational acceleration about the lateral axis, reducing the time between steady-state flight phases. The extra control authority of aft-tail and canard aircraft has another advantage; it allows the control of aircraft having neutral or negative static stability. This can result in higher pitch accelerations, further increasing agility. The presence of the horizontal control surfaces, depending on their application and operation, can have two other important implications for agility. If the horizontal tail surfaces can be operated differentially, extra roll authority can be imparted to the aircraft. Finally, all-moving tails and canards should retain greater effectiveness at supersonic speeds than the elevons used on delta aircraft, which are situated at the rear of the wing and may be operating in regions of shock-separated flow.

For all of the above reasons, aft-tail and canard aircraft are likely to be more agile than delta configurations. Having already rejected aft-tail aircraft because of their excessive growth encountered for these requirements, the competition is now limited to delta and canard-delta configurations. For the total procurement of 620 aircraft, the discounted LCC differences between the equivalent canard and delta aircraft are FY00M$1078 for the single engine option, or FY00M$867 for the twin engine design. Given these relatively small increases in total cost, and the large combat effectiveness benefits of having the canard, only canard configurations will be carried forward for further studies.

The number of engines has a much larger effect on the cost and size of competing designs, but is one of the most difficult decisions to make, for a number of reasons. One of the main advantages from having two engines is the increased probability that an aircraft can return to base following an engine failure, although engine failures are twice as likely to occur for a twin engine aircraft. Twin engine aircraft should be more survivable during wartime scenarios than equivalent single engine designs, as they are smaller, and will have reduced visual, radar, and acoustic signatures. Assuming that engine detectability is determined mostly by thermodynamic cycle, twin engine aircraft will have lower signatures, as they will require less thrust for a given flight phase. The fact that the engines are separated slightly may also help reduce IR signature, as there will now be two smaller sources, rather than one larger footprint. A twin jet efflux should also cool more quickly, due to the larger external 'surface area' of the jets.

In terms of the probability of the engine(s) being damaged in war, the twin engine option should have an advantage, if the probability of a hit is proportional to the target area presented. From frontal and rear aspects, the probability of a hit on the two smaller engines is 35% less than the single engine option ($0.86m^2$ and $1.37m^2$, respectively, from Table 8-3). The area ratio from the plan/lower view has reduced by slightly less, at 29% ($8.36m^2$ and $5.89m^2$), but the target area from the side is much lower, with a reduction of almost 65% ($8.35m^2$ and $2.95m^2$). Although the target area is lower, vulnerability for twin engine aircraft may not be lower, as damage to one engine may result in both engines being lost, due to their proximity. This method may be a gross approximation, but it does illustrate another benefit of twin over single engine options.

The argument for twin engine over single engine configurations is strong, as it results in smaller, lighter, and more survivable aircraft. However, the discounted LCC increases 4.52% in changing from a single to a twin engine canard option, equivalent to a total for the 620 aircraft of FY00$M3693. As this is a considerable sum, further analysis would be required, and the final decision may be purely political. For that reason, both single and twin engine designs will continue to be considered.

8.3.5 Internal Weapons Carriage

With the increased use of stealth technologies, and emphasis on reducing radar cross-section (RCS), a feature that is becoming increasingly common is the internal carriage of weapons. This has two beneficial effects for the aircraft: it lowers the RCS, which can reduce detectability and improve aircraft survivability, and; it reduces friction drag by reducing the surface area exposed to the external airflow. The detrimental effects of carrying weapons internally are that the airframe will expand to accommodate the extra volume of the weapons, increasing airframe mass, wetted area, and drag. The presence of another internal bay and a more complicated missile launch system will further increase aircraft empty mass.

In order to keep the results as consistent as possible, the internal weapons to be carried for the following example are the same weapons as were carried in previous examples, with mass and volume penalties as described above. The dimensions of the weapons are shown in Figure 8-12, and the bays required to hold them were assumed to be 2.9m × 0.4m × 0.4m for each of the short range missiles, and 3.8m × 0.5m × 0.5m for each of the medium range missiles, giving a total volume of 2.83m^3. Total volume was increased to 2.9m^3 and the total weapons mass was increased from 475kg to 600kg, to allow for the internal bay structure and launch installation/mechanism.

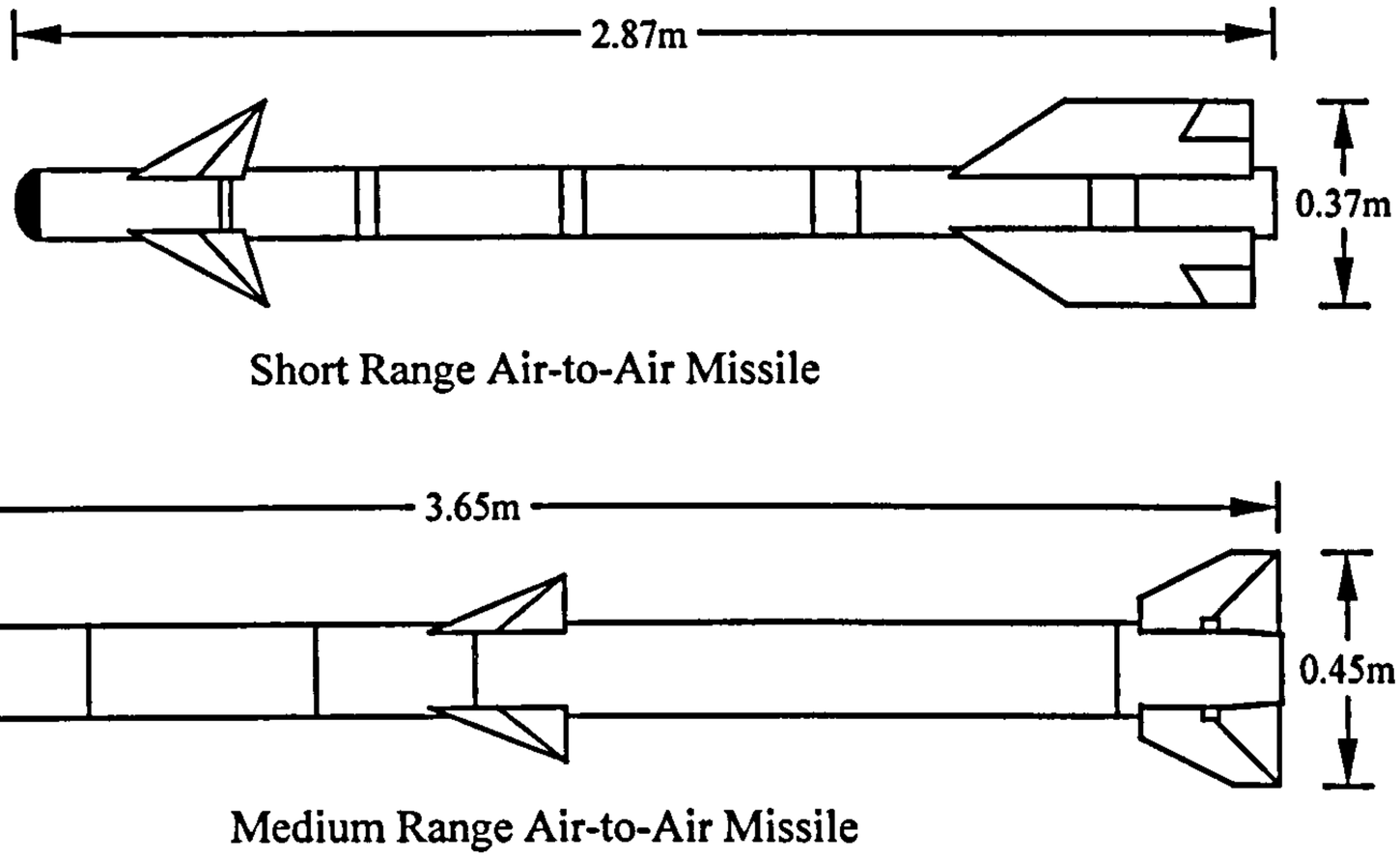
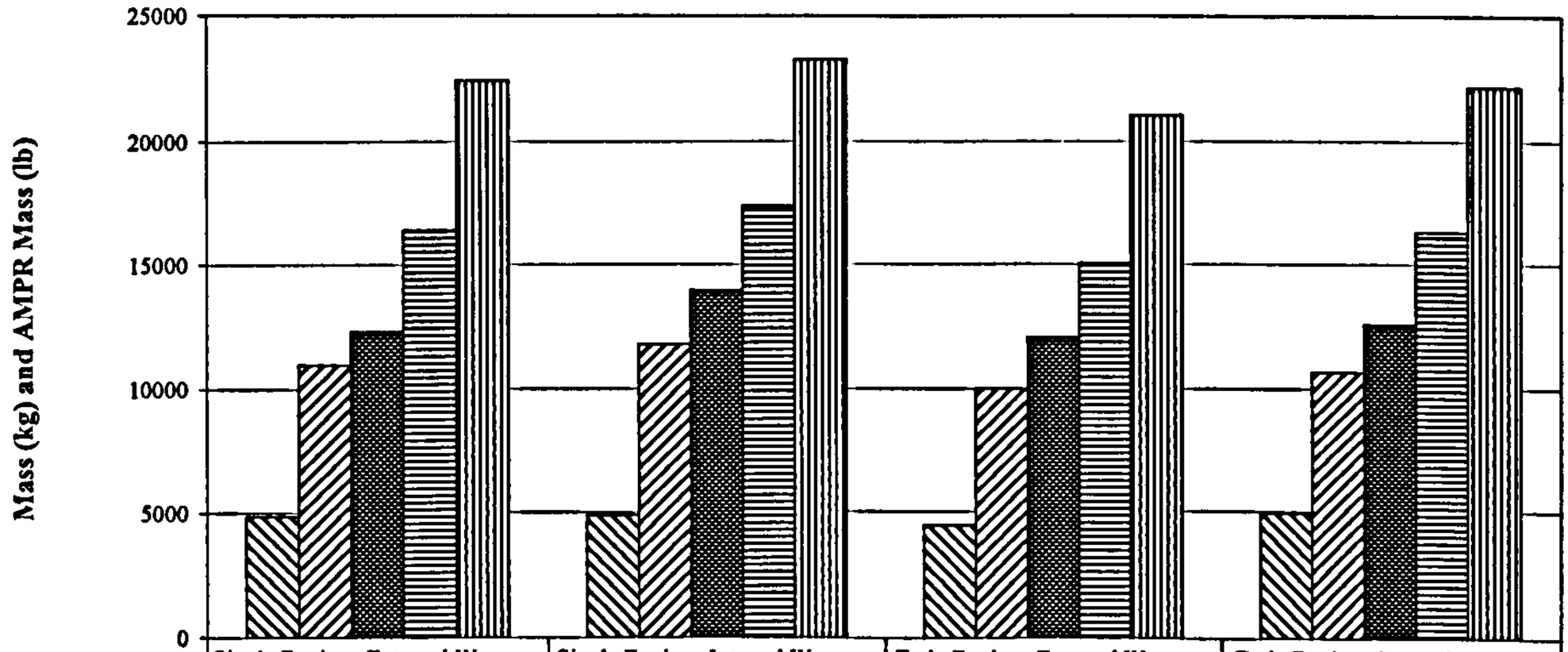


Figure 8-12. Weapon Dimensions.

The results shown in Figure 8-13 summarise the mass effects of internal carriage of the weapons shown above on the single-crew, single-fin canard configuration. The gross masses of the single and twin engine aircraft options have increased by 3.76% and 5.42%, whilst empty masses are found to have risen by 7.48% and 6.55%, respectively. The difference in the change in AMPR mass between the engine options is found to be much more significant, with the single engine AMPR mass increasing by 13.47%, as opposed to the twin engine option, which has an AMPR mass increase of 4.45%.

The AMPR mass difference is due to the change in the point performance constraints that are actively driving the design. For the single engine external carriage option, the driving constraints are landing distance, first attained turn (7g at M=1.4 and 12750m),

and time-to-climb. Once internal carriage is specified, wing area increases significantly, probably as a result of the extra volume required, causing landing constraint and time-to climb to become the only driving constraints. With the effective reduction in wing loading (10.0%), the thrust/weight ratio drops by 3.0%, as only a very small increase in engine mass flow rate occurs to meet the point performance requirements.



	Single Engine External Weapons	Single Engine Internal Weapons	Twin Engine External Weapons	Twin Engine Internal Weapons
Fuel Mass	4857	4880	4457	4943
Empty Mass	10974	11795	10001	10656
AMPR Mass (lb)	12311	13969	12068	12605
Mission Mass	16406	17375	15033	16299
Gross Mass	22431	23275	21058	22199

Figure 8-13. Mass Totals – Internal vs. External Weapons Carriage.

The twin engine designs have the same point performance drivers for both external and internal carriage, resulting in more of a simple re-sizing than the single engine design process. As with the single engine solution, wing loading reduces with the introduction of internal carriage, but by a smaller amount (4.5%), although the thrust/weight ratio is now seen to increase by 6.7%. This demonstrates a fundamental difference in the response of the alternative engine options in meeting the new design requirements. The single engine design uses a large increase in wing area to achieve the point performance constraints. The twin engine design uses a larger engine mass flow rate (thrust) to achieve the same goal, but this does not increase the AMPR mass to the same extent as an increase in wing area.

The cost effects of internal weapons carriage are shown in Figure 8-14. They follow logical trends identified in previous sections; for a given overall configuration type, cost is generally a function of mass. There is no reason why this should not be the case, as the aircraft is being kept at a very similar technological level, and scaled in size. If the costs were not to follow the size (mass) trend, this would give cause for concern. Discounted life cycle costs have increased by 5.49% and 2.91% for the single and twin engine options, which is the opposite to the trend in gross mass increase (3.76% and 5.42% respectively). This serves to highlight the complex interaction between LCC and mass for competing aircraft designs, and the difficulty in specifying point performance constraints to represent a balanced performance requirement.



	Single Engine External Weapons	Single Engine Internal Weapons	Twin Engine External Weapons	Twin Engine Internal Weapons
Total	125.833	132.744	131.789	135.625
Disposal	0.184	0.210	0.186	0.191
O&S	44.280	45.074	47.381	49.107
GSE&IS	5.166	5.602	5.436	5.597
Production	50.918	55.218	53.587	55.172
RDT&E	25.285	26.640	25.199	25.557

Figure 8-14. Cost Breakdown – Internal vs. External Weapons Carriage.

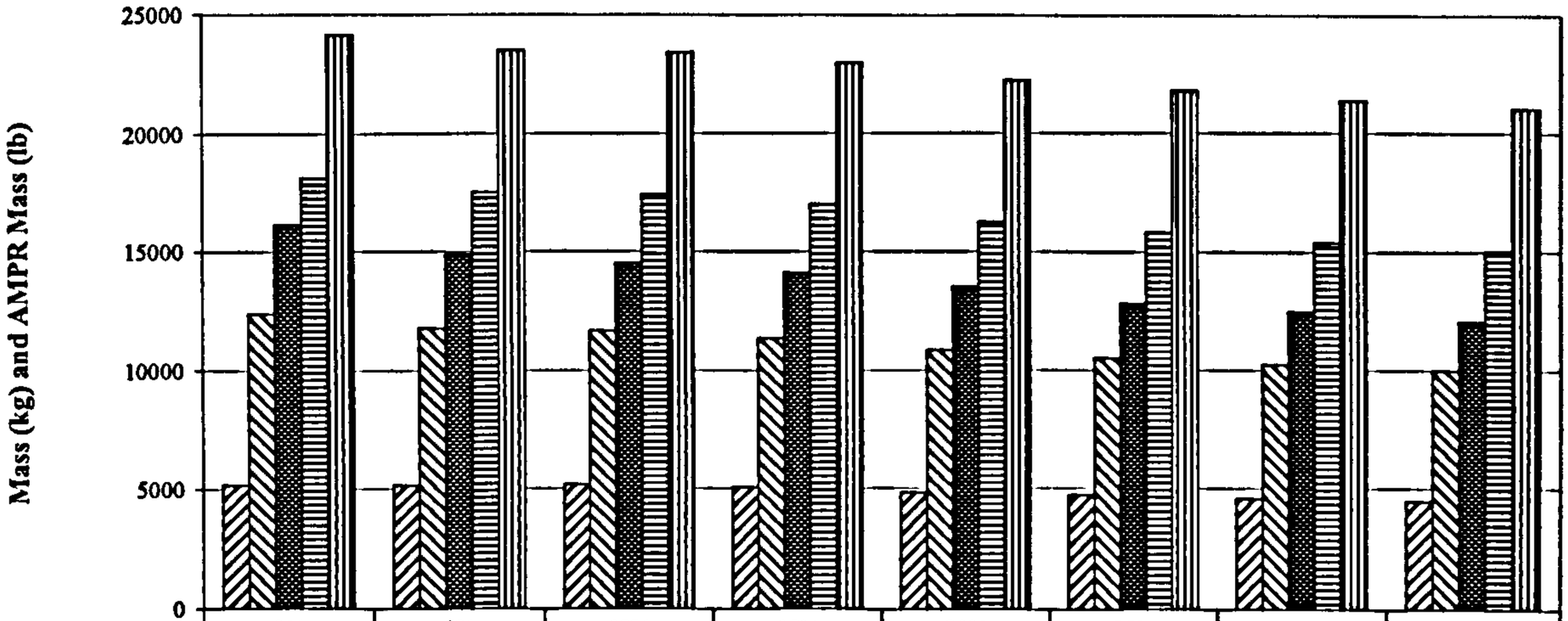
The aircraft requirements refer to a low-observable (LO) design, rather than a very-low observable (VLO) design, in order to prevent very large potential cost increases, and reduce programme risk. Internal weapons carriage will reduce RCS, a major contributor to observability, although infra-red, visual, acoustic, exhaust particulate emissions, and other factors also contribute to the overall aircraft signature. However, the RCS benefits of internal weapons carriage for aircraft not making use of VLO technologies is disputed, and raises questions over the use of this option to meet the current requirements. The discounted programme LCC increase for the total of 620 internal-carriage aircraft would be FY00M$4285 for the single engine option, and FY00$2378 for the twin engine configuration. Without detailed aircraft descriptions, RCS simulations, and their impact on military effectiveness, a definitive decision is not realistic. However, given the basic terms of the aircraft description and the cost increases incurred by internal weapons carriage, a strong case could be made for not choosing internal carriage.

8.3.6 Effect of Advanced Materials

Much debate has taken place in recent years over the financial implications of including advanced materials in combat aircraft designs. The use of advanced composite materials will reduce the mass of the aircraft, but will increase the cost per unit mass for the acquisition phase, and may not result in cost savings in O&S or life cycle terms. Figure 8-15 shows the mass results generated for the single-crew, single-fin canard aircraft by fixing the empty mass fraction of advanced materials at discrete levels from 0% through to the maximum permissible value for this design of 33.71%.

The mass totals decrease with increasing use of advanced materials, and follow a downward trend in all cases except the fuel mass for the 10% materials fraction aircraft. This is probably a result of the optimisation process being based on LCC for these

examples, but will also be influenced by the mass reduction method described in section 4.3.1. The initial mass saving is assumed to reduce the mass of the empennage linearly by up to 30%, but once the empennage is entirely made up of composite materials, the mass saving is next applied to the wing, which has a maximum mass reduction factor of 16%. The overall structural mass saving becomes the average of the two mass reduction factors, reducing the effect of adding advanced material after the initial rapid drop.



	0%	5%	10%	15%	20%	25%	30%	33.71%
Fuel Mass	5177	5144	5179	5042	4810	4722	4573	4457
Empty Mass	12380	11775	11654	11358	10856	10509	10239	10001
AMPR Mass (lb)	16133	14869	14538	14130	13548	12817	12492	12068
Mission Mass	18132	17494	17408	16975	16241	15806	15387	15033
Gross Mass	24157	23519	23433	23000	22266	21831	21412	21058

Advanced Materials Fraction by Empty Mass

Figure 8-15. Mass Totals vs. Advanced Materials Fraction.

The above effect probably plays a large part in the advanced material LCC results, shown in Figure 8-16. As can be seen, the LCC values do not change by more than 2.1% from the minimum value achieved, and do not follow a clear trend for any of the cost components. For this study, acquisition costs are most strongly influenced by the aircraft AMPR mass and the advanced materials factor, which increases with the advanced materials fraction, as shown in Figure 7-3. The down-sizing effect on the aircraft slightly exceeds the increase in airframe costs for the acquisition phase, and this difference reaches a maximum of 1.79% for the 5% advanced materials option.

In general, O&S costs decrease with increasing use of advanced materials, although the relatively large scaling effect on the 5% materials design causes the 10% advanced materials option breaks this trend. The overall trend is primarily due to the down-sizing of the aircraft, with decreases in all of the O&S cost components. The largest savings are realised for the engine contract costs, mechanical contract costs, and personnel costs. Personnel costs are mainly driven by the required maintenance effort, which is split into scheduled and unscheduled (arising) activities. Scheduled maintenance effort effects are small, varying by a maximum of 0.82%. This is because the effort required to perform routine maintenance on advanced materials is assumed to be larger than for conventional metallic components, due to the difficulty in detecting and repairing beyond-visible impact damage and other composite-specific problems. This has the effect of cancelling out the aircraft down-sizing benefits. Unscheduled maintenance

effort decreases by 14.9% with increasing materials usage, because of the down-sizing process and the reduction in structural arisings. Unscheduled structural defects should show an overall reduction in effort with increasing advanced materials fraction, as fatigue, corrosion, and other shortcomings of metallic components are removed.

Disposal costs show the largest percentage change of all of the major cost components, but because of the relatively small contribution of this life cycle phase, LCC effects are minimal, as discussed at the end of section 8.3.3.



	0%	5%	10%	15%	20%	25%	30%	33.71%
Total LCC	134.551	132.329	132.914	132.960	132.080	131.796	131.955	131.789
Disposal	0.165	0.162	0.167	0.172	0.175	0.178	0.184	0.186
O&S	49.958	49.254	49.449	49.026	48.393	48.087	47.582	47.381
GSE&IS	5.412	5.293	5.347	5.384	5.370	5.379	5.429	5.436
Production	53.346	52.174	52.703	53.073	52.930	53.017	53.517	53.587
RDT&E	25.670	25.176	25.248	25.304	25.211	25.135	25.243	25.199

Figure 8-16. LCC Breakdown vs. Advanced Materials Fraction.

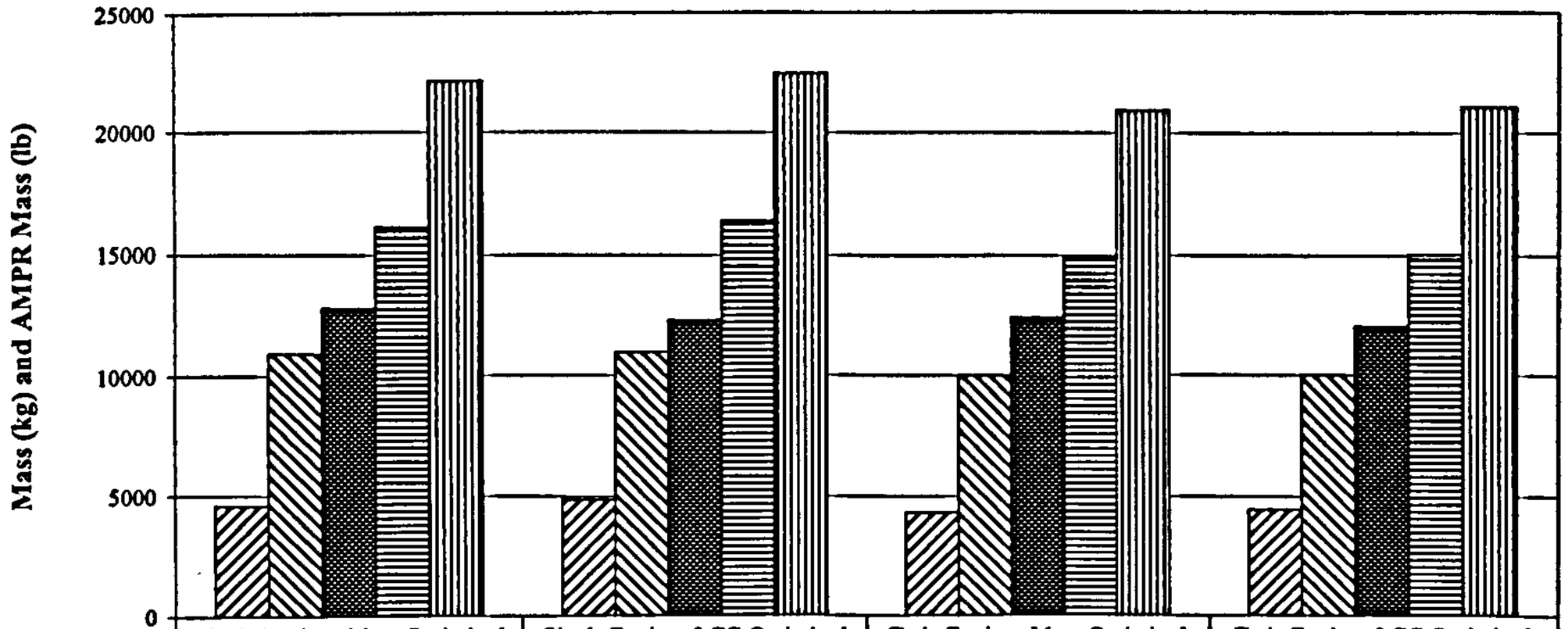
The trend in cost with increasing advanced materials fraction is a complex function of aircraft size, mass, and performance, with local minima, for the above example, at 5%, 25%, and 33.7%, the maximum allowable advanced materials usage fraction. However, in every case where the optimisation starting point for the level of advanced materials was over 15%, the minimum cost solution has made use of the maximum permissible amount of advanced materials. Below 15%, RQPMIN has occasionally returned results consistent with a local minimum solution. Given the low gradients and the 'flatness' of the objective function, demonstrated in Figure 8-16, this is not surprising.

With the increasing use of advanced materials for military and civil engineering programmes, it is likely that in-service costs of military equipment will reduce. This should prompt modifications to the LCC model, to reflect advances in composite material maintenance techniques, and result in 'stronger' cost vs. materials usage trends.

8.3.7 Influence of the Objective Function

A fundamental assumption made throughout this piece of work has been that minimum LCC would be returned by an aircraft configuration generated using the LCC objective function. In order to demonstrate that this is true, results were generated to show the effect of changing from a conventional mass-optimised approach to the current LCC-optimised technique. Single and twin engine versions of the single-crew, single-fin canard aircraft were optimised for minimum gross mass, and the mass results for the different objective functions are shown in Figure 8-17.

As can be seen, the empty mass changes are very small, increasing by only 0.41% and 0.04% for the single and twin engine options, whilst gross mass increases are only slightly larger, being 1.33% and 0.65%, respectively. The differences in fuel mass (increases of 5.40% and 3.08%) and AMPR mass (reductions of 3.97% and 2.76%) give an indication of the design changes taking place that are not reflected in the mass totals.



	Single Engine Mass-Optimised	Single Engine LCC-Optimised	Twin Engine Mass-Optimised	Twin Engine LCC-Optimised
Fuel Mass	4608	4857	4324	4457
Empty Mass	10929	10974	9997	10001
AMPR Mass	12820	12311	12411	12068
Mission Mass	16112	16406	14896	15033
Gross Mass	22137	22431	20921	21058

Figure 8-17. Mass Totals – Mass and LCC-Optimised Solutions.

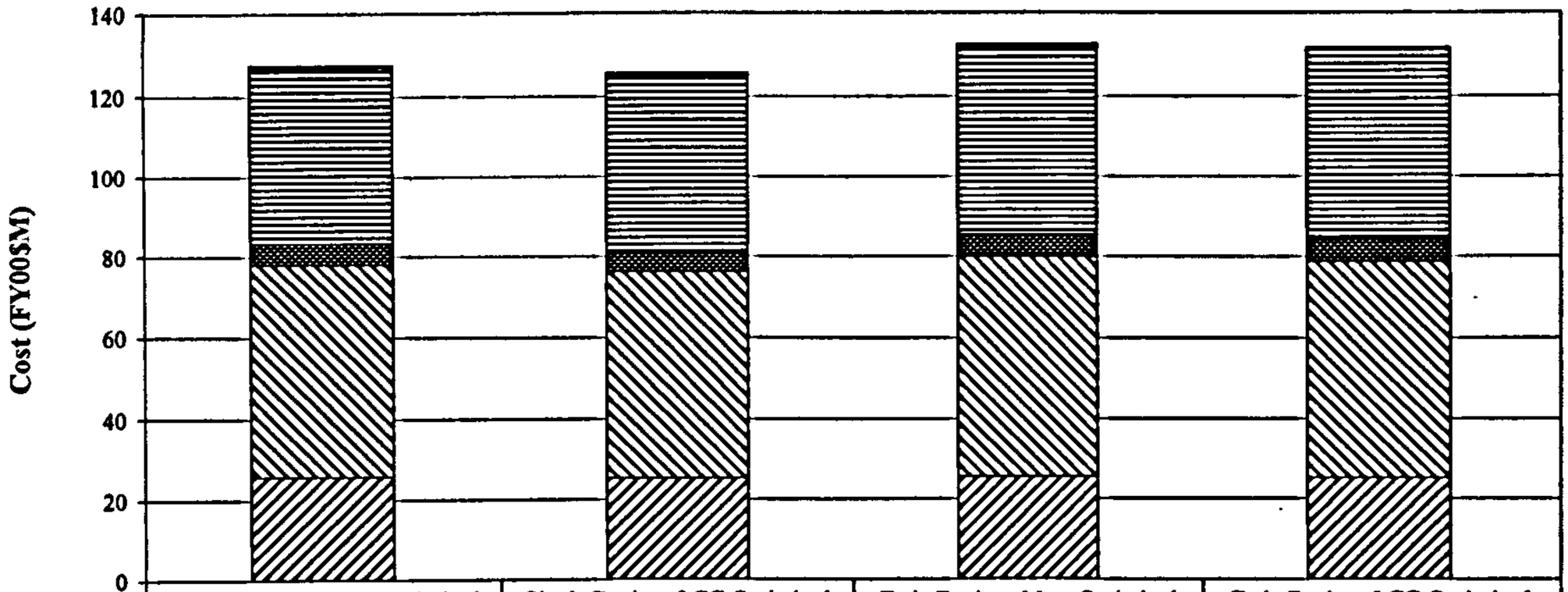
Table 8-4 shows some of the major aircraft parameters and component masses, and the changes that have taken place in changing the objective function from gross mass to LCC. For both the single and twin engine options, the main changes that have occurred are the reduction in airframe mass, and an increase in the engine air mass flow rate. The majority of the airframe mass savings for the single engine aircraft were the result of wing mass reductions. They were caused by a lower aspect ratio, increased thickness/chord ratio, a minor reduction in ¼ chord sweep, and the increase in the fuselage diameter, with only a small increase in gross wing area. A fuselage mass reduction was also achieved, mainly because of a decrease in fuselage length. The major airframe mass saving for the twin engine aircraft came from the reduction in fuselage length and mass. There was also a small decrease in wing mass, resulting from an increase in the thickness/chord ratio, and a small drop in wing loading, although wing area did rise.

Aircraft Parameter	Single Engine			Twin Engine		
	Mass-Optimised	LCC-Optimised	% Change	Mass-Optimised	LCC-Optimised	% Change
Wing Area (m^2)	51.294	51.769	0.93%	46.824	47.841	2.17%
Aspect Ratio	2.288	2.197	-3.98%	2.350	2.529	7.62%
¼ Chord Sweep (°)	41.437	40.244	-2.88%	39.501	39.578	0.19%
Thickness/Chord	0.0457	0.0525	14.88%	0.0411	0.0511	24.33%
Fuselage Length (m)	17.545	16.526	-5.81%	16.764	15.491	-7.59%
Fuselage Diameter (m)	1.890	1.951	3.23%	1.918	1.974	2.92%
Fuselage Height (m)	2.743	2.662	-2.95%	2.347	2.224	-5.24%
Canard Area (m^2)	2.322	2.375	2.28%	2.189	2.276	3.97%
Fin Area (m^2)	5.059	4.962	-1.92%	4.952	4.781	-3.45%
Advanced Material (%)	33.48	31.08	-7.17%	35.36	33.71	-4.67%
Eng. Mass Flow (kg/s)	150.875	160.840	6.60%	70.128	74.264	5.90%
Wing Mass (kg)	1911.8	1729.6	-9.53%	1687.7	1644.9	-2.54%
Fuselage Mass (kg)	1565.7	1501.7	-4.09%	1672.7	1554.5	-7.07%
Empennage Mass (kg)	181.1	179.7	-0.77%	174.7	172.1	-1.49%
U/carriage Mass (kg)	846.2	856.6	1.23%	802.8	807.7	0.61%
Engine Mass (kg)	2571.4	2807.4	9.18%	1928.9	2067.1	7.16%
Systems Mass (kg)	3853.2	3898.5	1.18%	3730	3754.7	0.66%
Fuel Mass (kg)	4608	4857	5.40%	4324	4457	3.08%

Table 8-4. Aircraft Parameters - Mass and LCC-Optimised Solutions.

The table and the discussion above highlight the complexities of the interactions between the aircraft design variables, and the power of the optimisation tool in reaching a converged solution, with so many competing factors to resolve. The existence of local minima is perhaps not surprising, given the weakness of the objective function with respect to some variables, although the optimisation tool has been found to be robust.

Figure 8-18 contains the LCC breakdown for the mass and LCC-optimised options, and clearly shows that the LCC-optimised aircraft do indeed have lower through-life costs, although the totals are similar. The single engine aircraft has a discounted LCC reduction of FY00$M1.764 per aircraft (1.38%), which gives a saving for the total buy of 620 aircraft of FY00$M1094. Despite an engine production cost increase of M$154, the drop in AMPR mass and consequential reduction in the advanced materials fraction results in acquisition cost savings of FY00M$1254. The savings for the twin engine design are not as large as for the single engine option, but are still significant, being FY00$M0.956 for a single aircraft (0.72%), or FY00M$592.7 for the total procurement. Again, acquisition costs are lower than for the mass-optimised design, but by the lower amount of FY00M$778.5. The lower acquisition costs resulting from the LCC optimisation process could prove very beneficial to the aircraft programme, as this money immediately becomes free to be used for further development work, component improvement programmes, or could be returned to the Treasury. As in-year spending is often capped to a maximum value, the lower acquisition costs may be more important from an economic viewpoint than from purely LCC-based considerations.



	Single Engine Mass-Optimised	Single Engine LCC-Optimised	Twin Engine Mass-Optimised	Twin Engine LCC-Optimised
Total	127.597	125.833	132.745	131.789
Disposal	0.194	0.184	0.193	0.186
O&S	44.039	44.280	47.091	47.381
GSE&IS	5.306	5.166	5.521	5.436
Production	52.298	50.918	54.415	53.587
RDT&E	25.760	25.285	25.525	25.199

Figure 8-18. Cost Breakdown – Mass and LCC-Optimised Solutions.

The only major cost component to increase in changing from the gross mass to LCC objective function is Operation and Support cost, which increases by 0.55% and 0.62% for the single and twin engine options, respectively. These increases are dominated by the growth in engine contract costs, due to the increase in engine air mass flow rate and thrust. All other changes in individual O&S cost contributors represent less than 0.1% of the total O&S cost, with the exception of Sustaining Support costs, which reduce by 0.21% and 0.11% for the two options, due to lower aircraft acquisition costs. The reductions in AMPR mass and advanced materials fraction cause relatively large drops in disposal costs, but, again, the effect on total LCC is negligible.

Although the designs are clearly different for the LCC and mass-optimised solutions, the differences are sufficiently small to warrant explanation of the relevance of optimisation for minimum LCC. It is felt that the small differences between the two designs are mostly due to the large dependency of the R&M model (and hence a major part of O&S cost) on aircraft mass. Reducing the dependency of the R&M models on mass, and increasing the emphasis on more representative R&M drivers, the final aircraft designs may become more varied. However, the discounting process effectively reduces the advantages of downstream savings, preventing the design from becoming significantly different to the current solutions.

8.3.8 Final Configuration Selection

As previously discussed, the choice between single and twin engine configurations is extremely difficult. The final decision essentially comes down to a choice between larger, single engine, cheaper aircraft that are more likely to be lost to peacetime attrition, and smaller, twin engine, more expensive aircraft that will probably suffer fewer peacetime attrition losses, and will possibly be more survivable in wartime.

The author's personal judgement would be to select the twin engine aircraft, because of the reduced risk of aircraft (and aircrew) losses, and superior wartime survivability that it offers. However, the results and discussion above point to the beginning of the design choices available to the design team, and there are many more decisions to be made before the 'winning' design is finalised. Whichever design proves to be the one chosen, it is certain that it is unlikely to be the 'best' choice for every set of circumstances. It is hoped that in future, COMBAT, or tools like it, will be used to guide designers from the start of programmes, so that informed choices can be made, and effort can be directed in the right direction at an early stage in the design process.

8.3.9 More Electric Aircraft Study

Now that a final configuration type has been selected, the final demonstration of the capability of COMBAT is a study to investigate the cost savings available from the introduction of advanced technologies. The systems considered for the study are 'More Electric Aircraft' (MEA) technologies, which aim to replace conventional aircraft systems with electric, or electric-based systems. The replacement electrical components are assumed to have R&M benefits, and in some cases, lower mass. The systems to be replaced include the hydraulic flight controls, airframe hydraulics and pneumatics, engine power off-take, electrical generation, and engine start. In order to provide a simple labelling system for the changes, the following incremental system changes are presented, where all changes are considered relative to the systems that would be fitted to an equivalent non-electric aircraft. All of the aircraft are twin engine canard designs that have been re-optimised for minimum LCC; the results are presented with the twin-engine canard concept selected at the end of the last section.

Level 1: Replacement of hydraulic flight control system with either electrohydrostatic actuators (EHAs) or electromechanical actuators (EMAs). EHAs would consist of an actuator, reservoir, and hydraulic pump driven by an electric motor, all within one unit. EMAs would consist of an electric motor providing direct drive through a clutch and gearbox assembly. EHAs are assumed to have a mass 50% higher than a standard unit, whilst the mass of EMAs remains unchanged. The total aircraft scheduled maintenance figure is assumed to reduce by 10%, and the arising maintenance values for both more-electric systems are assumed to reduce by 50%. Aircraft electrical system mass increases to 150%, electrical arising rate increases by 10%, and the advanced technology cost factor is increased by 2% for EHAs, and 3% for EMAs, to reflect the increased cost in developing and producing the new components. It has also been assumed that the electrical systems will be more efficient than the purely mechanical and hydraulic systems, leading to a 10% reduction in engine power off-take.

Level 2: Level 1 plus the total replacement of all aircraft hydraulic and pneumatic systems with electrical systems. Hydraulic and Pneumatic arising maintenance and mass drops to zero, but electrical mass is now increased to 200%, and electrical arising maintenance is increased by 20%. Total aircraft scheduled maintenance is reduced to 80% of the nominal level, and the advanced technology factor is increased by an extra 1%. A 10% reduction is applied to the ground support equipment costs for acquisition (GSE&IS), and elements of O&S costs (replacement support equipment), to account for the lack of hydraulic and pneumatic support equipment.

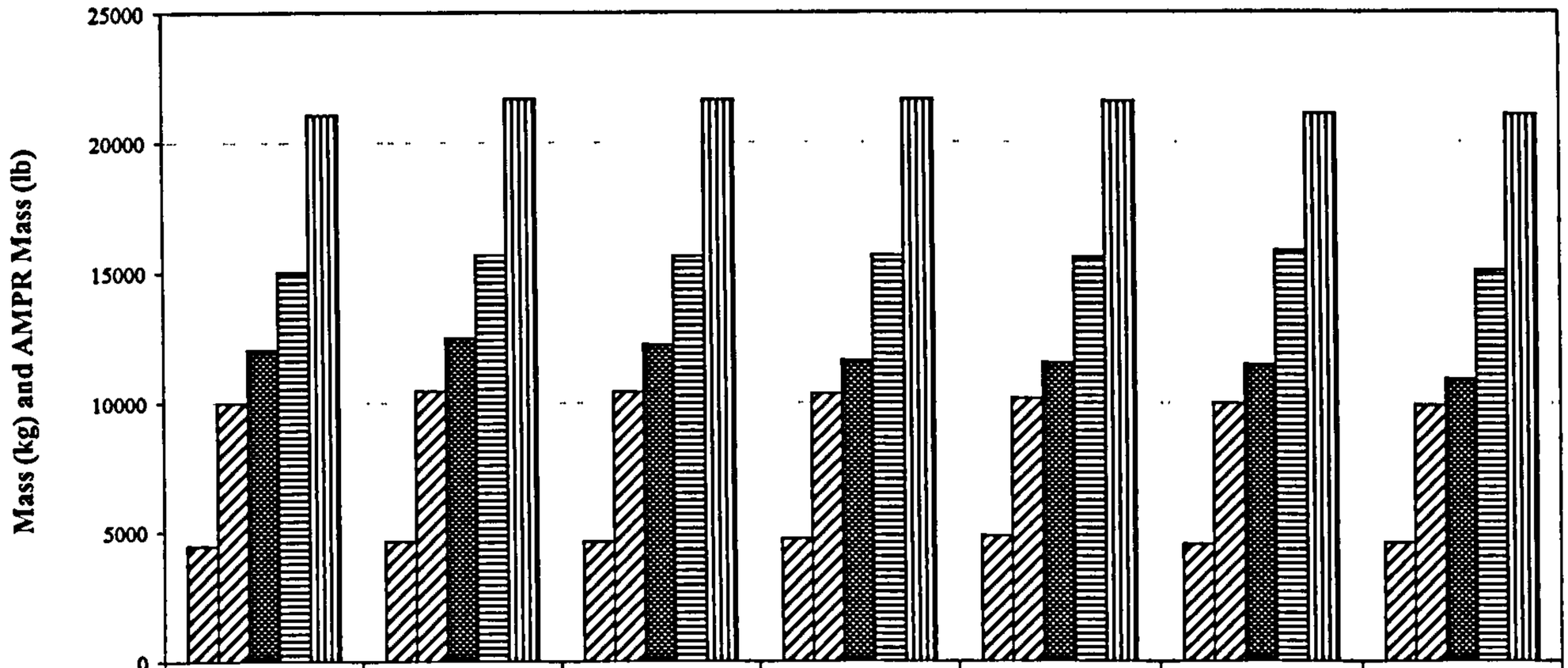
Level 3: Level 2 plus the integration of an all-electric starter/generator system into the engine, increasing electrical mass to 250% of the nominal amount. The electrical starting and off-take reduces the installed engine mass by 10% because of the removal of the external engine gearboxes and separate engine start system. Total aircraft scheduled maintenance drops to 70% of the nominal value because of the reduced maintenance required by having the combined engine start and generation system, which will have very few (if any) extra moving parts. The advanced technology factor is increased by an extra 4% in this instance, to allow for the cumulative effect of including all of the new technology systems. All other values remain as for Level 2.

Table 2-1 shows a summary of the system mass, maintenance effort, and cost factors applied for the electromechanical and electrohydrostatic replacement applications.

	Electrohydrostatic System			**Electromechanical System**		
Variable	**Level 1**	**Level 2**	**Level 3**	**Level 1**	**Level 2**	**Level 3**
Flight Control Mass	1.5	1.5	1.5	1.0	1.0	1.0
Engine Installation Mass	1.0	1.0	0.9	1.0	1.0	0.9
Hydraulic/Pneumatic Mass	0.5	0.0	0.0	0.5	0.0	0.0
Electrical System Mass	1.5	2.0	2.5	1.5	2.0	2.5
Advanced Tech. Factor	1.02	1.03	1.07	1.03	1.04	1.08
Scheduled Maintenance	0.9	0.8	0.7	0.9	0.8	0.7
Flight Control Maintenance	0.5	0.5	0.5	0.5	0.5	0.5
Hyd./Pneu. Maintenance	1.0	0.0	0.0	1.0	0.0	0.0
Electrical Maintenance	1.1	1.2	1.2	1.1	1.2	1.2
GSE/O&S Factor	1.0	0.9	0.9	1.0	0.9	0.9
Engine Power Off-take	0.9	0.9	0.9	0.9	0.9	0.9

Table 8-5. Mass, Maintenance, and Cost Factors for MEA Systems.

Mass totals for the various MEA configurations are shown in Figure 8-19. Most changes in the mass components are not large, with the exception of fuel mass for the EMA Level 1 aircraft, which increases by 8.09% over the values for the twin engine canard, and AMPR mass for the EMA level 3 aircraft, which drops by 9.47%.



	Twin Engine Canard	EHA Level 1	EHA Level 2	EHA Level 3	EMA Level 1	EMA Level 2	EMA Level 3
Fuel Mass	4457	4631	4617	4714	4818	4508	4574
Empty Mass	10001	10466	10431	10358	10163	10004	9911
AMPR Mass	12068	12515	12262	11638	11553	11498	10925
Mission Mass	15033	15672	15623	15646	15555	15807	15060
Gross Mass	21058	21697	21648	21671	21580	21112	21085

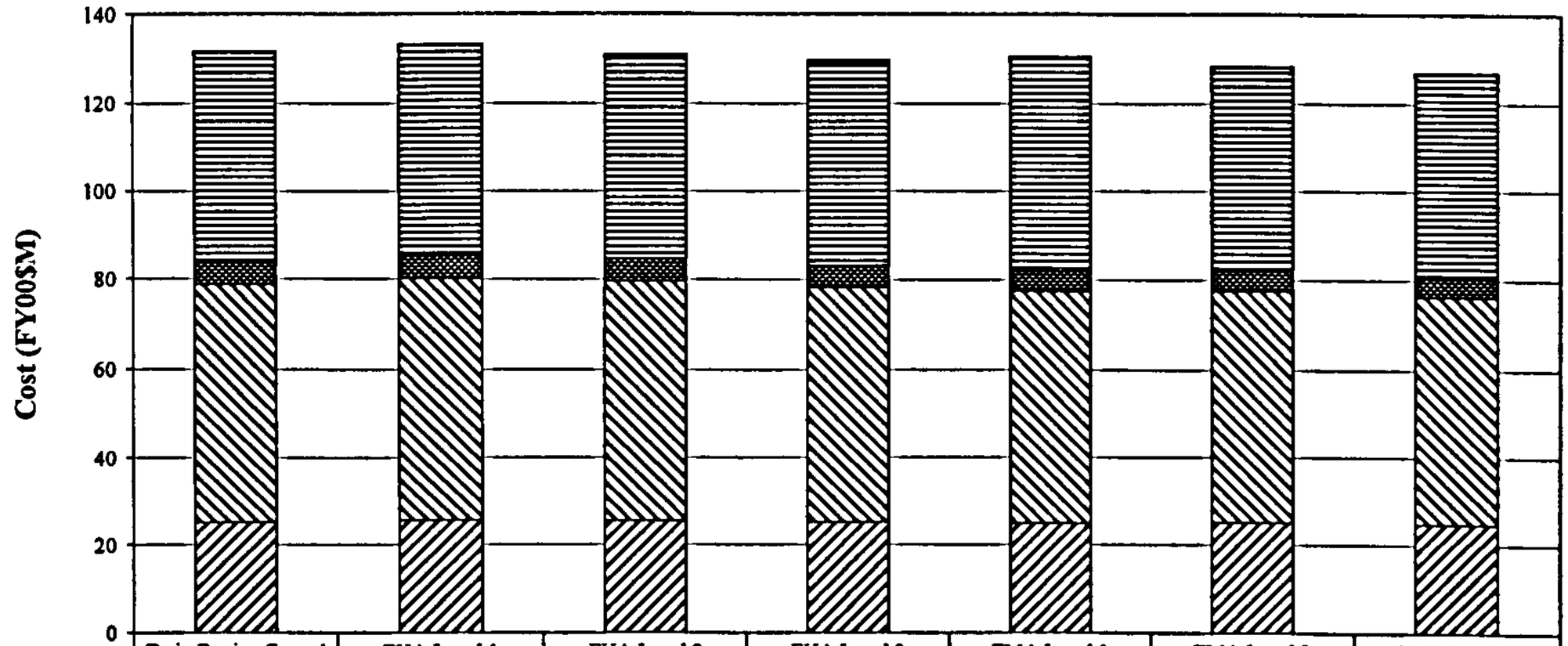
Figure 8-19. Mass Totals – More Electric Aircraft Options.

Table 8-6 shows the mass values for the 'baseline' twin engine canard configuration, and percentage changes for the aircraft utilising MEA technologies. The trends are seen to be similar for both EHA and EMA replacement flight control systems, although the lower mass of the EMAs is reflected in lower empty and AMPR masses across all levels of implementation. Direct mass comparisons of the effects of EHA and EMA implementation are not possible because of the differing effects of the mass changes on the basic configuration, and because the objective function is LCC, and not mass.

Mass Changes	Twin Engine Canard	EHA			EMA		
		Level 1	Level 2	Level 3	Level 1	Level 2	Level 3
Fuel Mass (kg)	4457	3.90%	3.57%	5.75%	8.09%	1.13%	2.61%
Empty Mass (kg)	10001	4.65%	4.30%	3.56%	1.62%	0.03%	-0.90%
Mission Mass (kg)	15033	4.25%	3.92%	4.08%	3.47%	5.14%	0.18%
Gross Mass (kg)	21058	3.03%	2.80%	2.91%	2.48%	0.25%	0.13%
AMPR Mass (lb.)	12068	3.71%	1.61%	-3.56%	-4.27%	-4.72%	-9.47%

Table 8-6. Mass Changes – More Electric Aircraft Options.

The cost figures, presented in Figure 8-20, show a general downward trend with increasing use of MEA technologies, although the initial increase in size caused by the EHA mass penalties negates the cost savings for the EHA level 1 aircraft. Replacement of the hydraulic and pneumatic systems with EHAs (Level 2) gives savings for GSE&IS and O&S that allow cost benefits to be realised, with a total LCC saving of 0.45%. Level 3 EHA replacement systems produce greater savings, with total a LCC reduction of 1.54%, equivalent to FY00$M1254 for the total procurement of 620 aircraft.



	Twin Engine Canard	EHA Level 1	EHA Level 2	EHA Level 3	EMA Level 1	EMA Level 2	EMA Level 3
Total	131.789	133 418	131 201	129.766	130 632	128 432	127.038
Disposal	0.186	0.190	0 188	0.180	0 177	0.179	0.172
O&S	47.381	47.427	46 422	46.352	47.585	45 891	45.754
GSE&IS	5.436	5.547	4 947	4 854	5.340	4 800	4.714
Production	53.587	54 672	54 176	53.164	52 641	52.575	51.632
RDT&E	25 199	25.582	25 469	25.215	24 890	24 987	24.765

Figure 8-20. Cost Breakdown - More Electric Aircraft Options.

The lower mass of the EMA components yields immediate benefits, with LCC reductions of 0.88%, 2.55%, and 3.61% for the three levels of MEA technology implementation. The final saving of 3.61% corresponds to total LCC savings of FY00$M2946, which is a significant amount, and would certainly be worth investigating in detail. The figures for both EHA and EMA replacements include the increases in the advanced technology cost factor, which would normally increase the development cost of the twin engine canard aircraft by FY00$M250, with production costs increasing by FY00$M276.

Even if the estimates for MEA technology costs were optimistic, the implementation costs would need to increase to more than five times the original estimates before LCC savings were removed from the EMA replacement programme. Cost saving for the EHA systems is less certain, but implementation costs would need to rise to 235% of the original estimate before EHA LCC savings were negated. This suggests that some level of LCC saving could be accomplished for both aircraft, although the accuracy of the cost estimates becomes more significant for electrohydrostatic systems.

Table 8-7 shows the costs for the twin engine canard aircraft, and the percentage changes for the MEA configurations. As with the mass variations, the trends are seen to be fairly consistent, with only minor variations taking place. Again, the utilisation of EMAs gives larger savings due to the lower masses of those components. With the exception of EMA Level 3 aircraft, the usual cost/mass trend has been broken, with new concepts having empty mass increases and LCC reductions. The EMA Level 3 aircraft has both empty mass and cost reductions, although the cost saving (3.61%) is significantly larger than then empty mass reduction (0.90%). This last option gives the largest LCC reduction, showing the benefits available when technologies present both R&M improvements and reduced size/mass.

Cost Changes	Twin Engine Canard	EHA			EMA		
		Level 1	Level 2	Level 3	Level 1	Level 2	Level 3
RDT&E	25.199	1.52%	1.07%	0.06%	-1.23%	-0.84%	-1.72%
Production	53.587	2.02%	1.10%	-0.79%	-1.77%	-1.89%	-3.65%
GSE&IS	5.436	2.04%	-9.00%	-10.71%	-1.77%	-11.70%	-13.28%
O&S	47.381	0.10%	-2.02%	-2.17%	0.43%	-3.14%	-3.43%
Disposal	0.186	2.15%	1.08%	-3.23%	-4.84%	-3.76%	-7.53%
Total	131.789	1.24%	-0.45%	-1.54%	-0.88%	-2.55%	-3.61%

Table 8-7. Cost Changes – More Electric Aircraft Options.

The MEA study has shown that replacing conventional aircraft systems with EHAs and EMAs will produce LCC savings, despite increases in aircraft mass and slightly higher acquisition costs. Although the cost penalties of EMAs have been assumed to be slightly higher than EHAs, they will return greater savings, due to their lower mass. Increasing the usage of more electric components reduces costs for both EHA and EMA systems, with fully electric aircraft offering the greatest savings.

8.4 Summary

Chapter 8 has:

- Demonstrated the robustness of the optimisation procedure, and given an example of the convergence trace achieved for one of the aircraft solutions.
- Presented a validation of the synthesis and cost methodologies for the F-16C and Eurofighter 2000 aircraft; non-optimised results were shown to be very good.
- Described the example aircraft specifications, including detailed point and mission performance requirements. Procurement details have been given, together with engine parameters, and indicators of engine performance relative to current designs.
- Outlined the procedure and methodology for selecting suitable candidate aircraft designs for a given set of performance, procurement, and operational requirements, based on LCC-optimised solutions.
- Investigated the suitability of single and twin engine versions of aft-tail, delta, and canard configurations for the specified mission. Two-crew and twin-fin options for the canard configurations were also considered, but were rejected, as the full benefits of these options could not be explored within this research.
- Shown that major configuration changes can break the mass-cost trend, in particular for single vs. twin engine options.
- Concluded that canard aircraft are probably the most suitable solution in cost and performance terms for the example specification. This is due to their smaller size and cost, relative to aft-tail designs, and because of their greater agility than delta configurations, despite slightly higher through-life costs.
- Demonstrated size and cost increases as the effects of internal stores carriage on single and twin engine canard designs. These options were rejected, as the current research cannot provide proof of the RCS benefits of internal carriage of stores on an aircraft not making use of VLO technology.
- Verified that aircraft optimised for minimum LCC have lower life cycle costs than aircraft optimised for minimum gross mass. LCC reductions were shown to be significantly larger than percentage changes in the empty mass of the LCC and mass-optimised designs.
- Demonstrated the flexibility and capability of the design tool in modelling and optimising many configurations for direct comparisons of configuration, mass, performance, and cost.
- Investigated the effects of More Electric Aircraft Technologies on the twin engine canard concept. The study concluded that MEA technologies should result in significant cost savings, increasing with greater utilisation, and showed that electromechanical components will return greater savings than electrohydrostatic units, despite their higher initial costs.

9 Discussion and Conclusions

The purpose of this chapter is to provide a retrospective overview of the research study, provide additional discussion of the research aims, and compare the achievements with the original objectives. It also lists improvements that could be made to some of the models, suggest areas for further work, and draws some overall conclusions for the research study.

9.1 Introduction

To help the procurement of more cost-effective military aircraft, a research study was initiated to reduce the through-life costs of combat aircraft by providing cost data during the conceptual design phase. In order to allow the rapid comparison of many competing aircraft concepts, the process was automated, using an existing optimisation tool to close the design iteration loop. In this way, conceptual aircraft designs can be produced to deliver a specified level of performance and minimum through-life costs, which can be considered as one step towards maximising cost effectiveness.

The overall aim of the research study was formalised in section 1.3:

"To produce a useable, working model to optimise the design of combat aircraft for minimum peacetime life cycle cost."

The fulfilment of these aims required a large piece of work, which included; design effort for the conceptual design of combat aircraft, the use of optimisation processes, formulation of a plausible life cycle cost estimating methodology, and the amalgamation of all of these models into a coherent software tool.

The aims were ambitious, particularly as the aircraft design methodology was to include complex thermodynamic algorithms for propulsion modelling. The progress detailed in previous chapters would not have been possible without the use of existing tools (such as ONX and OFFX for the engine models), and help and advice from those people mentioned in the Acknowledgements. Even with their help, some areas proved particularly problematic, especially the collection of cost data. Real cost data that was made available cannot be released in an open publication, such as this, for commercial and security reasons. Even the sensitive data that was obtained fell short of providing enough data points for all of the individual cost elements. To compound matters, cost data will always suffer from 'variability' due to errors in reporting, economic fluctuations, competition, political influence, and others factors discussed in section 1.2.

Without ambitious aims, though, the research may not have made a contribution to knowledge, as it would have been all too easy to create a basic aircraft conceptual design tool, add some very simple mass-based cost algorithms, and perform trade-off studies based purely on aircraft mass. This has been done before, and the standard conceptual design texts present both aircraft design algorithms and some very simple costing methods. Although the algorithms in the cost methodology, described in Chapter 7, rely on mass as an aircraft size descriptor, Chapter 8 has shown that certain configuration changes can reduce aircraft mass, but increase cost. This kind of result would not be possible with simple mass-based algorithms, and begins to demonstrate some of the potential of the tool, particularly if some of the developments suggested in section 9.3 are implemented.

9.2 Fulfilment of Project Aims

This section will review each of the major pieces of work that have contributed to this thesis (normally corresponding to chapters), and discuss how well each of them met their particular aims.

9.2.1 Literature Review

The aims of the literature review were: familiarisation with the study area; to determine the state of the art in the subject; to gather data and models that would be useful in the development of the tool; to prevent effort being wasted in repeating documented research; to identify other researchers looking at LCC; and, to determine how the thesis would contribute to existing knowledge.

The literature search proved an extremely useful activity at the start of the project, as not only did achieve all of the aims identified above, but it also helped to clarify terminology, which is vitally important when communicating with industrial contacts. The literature review process also helped to introduce the author into the costing community, as detailed questions regarding the work that other people are performing automatically makes them aware that new research in the subject is taking place. This often prompted interest in the current work, and offers of advice, pointers to other references, suggestions of 'rules of thumb', etc.

It is fair to say that the Literature review achieved all of its original aims, and, with the benefit of hindsight, highlighted some that were not fully appreciated when work began.

9.2.2 Optimisation

The optimisation tool, RQPMIN, has formed a significant part in achieving the overall goals of this work. Although not given an exhaustive description, a basic understanding of the processes and mechanisms utilised by RQPMIN was felt to be beneficial for both the reader and Author, who had no significant prior experience of optimisation.

It is felt that the basic principles of operation have been presented simply and concisely, but with sufficient detail to allow visibility of many of the basic and advanced functions to those with experience of optimisation tools. Descriptions of some of the major problems affecting gradient-based searches were highlighted, and the measures taken to reduce the risk of these occurring have been explained. In giving a brief, but useful, overview of optimisation tool used for this research, this chapter is felt to have been a useful addition to the thesis, completing the aim for its inclusion.

In terms of its operation, RQPMIN has been found to be an efficient and powerful tool for producing optimised conceptual aircraft designs. Few problems with convergence have been encountered, although the risk remains that a local minimum will halt the process before the global minimum can be reached. However, this is probably more a characteristic of the objective function than a failing of the optimisation tool, and can be overcome by starting the same investigation from different points. The production of many results in a relatively short space of time is one of the strengths of COMBAT, and would not be possible without the use of the optimiser to close the design iteration loop.

The optimiser has also allowed the introduction of many constraints to increase realism, such as the checks of airframe dimension ratios required by the aerodynamic modules. This has ensured that designs are feasible, in terms of real-world constraints, and constraints can be added easily should weaknesses be identified. An extra constraint that might be added in future developments is the limitation of taper ratio to prevent tip stall, as the synthesis currently always drops taper ratio to the minimum allowable value.

9.2.3 Program Structure and Airframe Definition

Before coding took place for the production of the software tool, a detailed description of the operation and structure of the code had been formalised to allow effective planning and efficient coding to take place. The first section of Chapter 4 describes the code structure, the modular approach adopted, and gives a brief explanation of the major sections of the model (optimiser, aircraft synthesis, LCC model). The second section gives visibility of the airframe definition methodology, including input variables, fuselage, wing, and empennage sizing, mass estimation and volume accounting, and demonstrates the accuracy of the methods used.

For a conceptual design synthesis, the sizing procedures were seen to be very accurate, and sizing and mass correlations of calculated and quoted values for a number of combat aircraft were presented. The method of fuel volume control was explained, and has been found to work well, especially as there are no discrete fuel tank volume limits to introduce discontinuities to any of the constraints, or objective functions.

Very few convergence/global design problems have been traced to the aircraft synthesis routines, although there were initially some discontinuities in the wing and overall mass calculations. The wing mass discontinuities were caused by step changes in the empirical wing mass factors at the critical mass values given in Table 4-1, and were due to a small mathematical error in calculating the gradient between the critical points. Once this problem was identified, the gradient functions were re-programmed and the problems have not reoccurred. Discontinuities in the gross mass were caused by too high a tolerance being allowed on the individual internal mass iterations, as described in section 3.2.1.

The airframe sizing methodology has been demonstrated to deliver realistic results for a wide range of combat aircraft using fairly simple methods. This suggests that most of the major sizing drivers are being captured, and have been scaled correctly, relative to one another. Overall, the aircraft design and sizing methodology gives a sound base from which to continue with the remainder of the aircraft analysis procedure.

9.2.4 Aerodynamics and Propulsion

The purpose of these two modules is to provide accurate values for lift and drag coefficients, angle of attack, thrust and thrust specific fuel consumption values for all flight conditions. To enable the rapid production of design configuration options the aerodynamic and engine-modelling routines must return the calculated values in an efficient manner. The models chosen have the advantage of good accuracy, without the need for computationally 'expensive' calculations.

The aerodynamic modules are based on parametric equations, or tabular data in the case of the drag model. Realistic results are produced, which have been validated with real data, where possible. The models rely on historical data, and novel shapes, such as forward-swept wings, blended-wing-body, flying wing, and 'stealthy' shapes cannot currently be modelled. Lack of correctly processed data is one of the major problems in implementing these changes, and one that will be difficult to overcome without access to sensitive commercial data, or extensive wind-tunnel testing/CFD programmes.

The engine modelling routines are a major contribution to this design synthesis, as they allow the simultaneous analysis of airframes with different engine models, which can be created for specific aircraft investigations. The ONX and OFFX thermodynamic models are capable of performing detailed calculations for a wide range of engine types and technologies for all flight ranges and Mach numbers. The results have been shown to be realistic, although sufficient information is not available to allow detailed comparison of real and theoretical engine characteristics. The engine dimension and mass prediction methods developed during this research have produced excellent results, although there is a question over the accuracy of the mass estimation of engines in the 150kg/s air mass flow rate region.

There are felt to be two main problems with the engine modelling routines; run times are much longer than for tabular interpolation methods, and the models are very sensitive to some of the input values during the engine design phase. Program run times could be shortened by generating an engine model using ONX, and then running OFFX to generate a matrix of data for many flight conditions. This file could then be interpolated in the same manner as the aerodynamic tables, which would probably be a more efficient method. However, it does not allow the engine variables to become part of the optimisation process, which may be possible in future versions of COMBAT.

The engine models are sensitive to changes in the input variables, causing OFFX to enter infinite loops or return computationally illegal values (division by zero, etc.). It is for this reason that the major engine variables (bypass ratio, overall pressure ratio, turbine inlet temperature, etc.) have not yet been given to RQPMIN for control. To improve the robustness of the engine models would require a large amount of work, and it may prove more efficient to re-write the models in a more 'optimiser-friendly' manner. OFFX, in particular, is highly iterative, and the convergence criteria for the calculations had to be tightened significantly during the implementation phases, to prevent discontinuities from occurring.

Despite the shortcomings of the engine models, and the lack of detailed data with which to validate them, the results would appear to be realistic, and correspond well for the data points that have been obtained. Flexibility is excellent for both aerodynamic and engine models, being able to model a wide range of airframe configurations, engine types, and technology options. With further work, run-times could be reduced, the robustness of the models could be improved, and new technologies, such as aerodynamic prediction for LO aircraft, and variable-cycle engines could be included.

9.2.5 Aircraft Point and Mission Performance

The point and mission performance calculation modules form a very important part of the aircraft design synthesis, as it is these two modules that essentially drive the size of the aircraft. The point performance constraints effectively determine the thrust/weight (engine air mass flow rate) and wing loading values (wing area), and the mission calculations drive the fuel mass and volume requirements.

The methods used are a combination of classical performance calculations, updated with corrections to account for real-world effects. Many of the fuel burn calculations have changed from simple mass fraction techniques, to take advantage of the extra information available from the mass and engine performance models. Calculations that would appear as one-step algorithms in simpler conceptual methodologies (such as cruise fuel burn) have been split into smaller segments to improve accuracy, although this does slightly increase program run time. It is felt that the additional accuracy that these improvements offer are worth small increases in run times, which would be negated if the engine model iteration times could be reduced.

The novel approaches used to predict ground performance of the aircraft are felt to be a useful addition, as they provide accuracy equivalent to more complex methods in a single parametric equation. Results are shown to be very good for both take-off and landing correlations, although complications with retarding devices increase the spread of results for the landing case. Corrections for instantaneous and sustained turn rates have been included, and are different for the differing configurations, based on the total lift being produced by the aircraft, not the wing alone (i.e. trimmed lift). The calculation of climb and acceleration performance is one area where significant differences exist between this synthesis and simple conceptual methods. Thrust and drag data is updated throughout the change in energy height, producing results that will be more responsive to conceptual design changes.

The mission performance calculations make use of many of the equations from the point performance estimation, although the emphasis changes from calculating absolute aircraft performance to estimating fuel used per flight phase. This depends heavily on the accuracy of the engine and aerodynamic drag models, which have been shown to produce realistic results. Several extra functions have been built into the performance routines, such as the ability to check sustained turn performance part way through missions, when the exact fuel fraction is not known. Another useful feature is the control of cruise Mach numbers by the optimiser, giving a limited level of mission profile optimisation, increasing the realism of the design process.

Where it has been possible to validate point performance data, the results have been shown to be impressive for a conceptual design methodology. There will always be room for improvement, but without significant increases in the number of data points available, it is difficult to suggest specific modifications. Validation of mission performance calculations has not been possible, as true mission performance data for combat aircraft is classified, due to the estimation of aircraft military effectiveness that it would allow. These points aside, the results of the mission performance calculations are certainly realistic, increasing confidence in the final size of the designs produced using COMBAT.

9.2.6 Life Cycle Cost Model

The life cycle cost model has proven to be by far the most difficult section of the study, because the overall structure and many of the models had to be initiated, and because cost data is variable and difficult to obtain. Although problems were expected, they were underestimated, mainly because there are many publications and people suggesting LCC expertise, but few have made valuable contributions to this work. A fundamental difference between the methodology used in this research and the majority of other cost models is the emphasis on activity-based costing. ABC has many benefits, including extra visibility and flexibility in the costing algorithms, and the use of data that is commercially important, thus being captured by company commercial sectors.

The basic structure of the LCC model follows the real-world processes in the life of any product; development, production, entry-into-service, operation, and disposal. Existing development and production cost models were utilised, with relatively minor changes being required to convert them to have to multi-partner capability, and to introduce the concept of advanced design tools. Modifications to the original models[21] were made to the section for advanced materials, algorithms for engine development cost were added, and engine production cost equations were replaced with more complex expressions. A simple fraction of production was included to make some allowance for the cost of ground support equipment and initial spares.

The accuracy of the production cost models has been shown to be very good for a number of modern combat aircraft. For the few data points available, development costs also appear to be representative. Both models include many intuitively appealing cost drivers, and the accuracy obtained in representing the costs of a wide range of aircraft suggests that the scaling for each is of the right order. The cost elements that have been most difficult to deal with have been avionics and software costs, although Boehm's[17] COCOMO has been found to deliver realistic results for the F-22 software development programme, once lines of source code are known. Avionics and software cost estimating has been notoriously difficult in the past, and no doubt will continue to be so with the rapid development of avionics systems and architectures. GSE&IS costs have not been obtained, and without some data for existing designs, it is difficult to know whether the current fraction is realistic. Those with costing experience who have commented on the model have not indicated that it is particularly erroneous.

As discussed in section 7.6, the prediction of reliability and maintainability is currently felt to be a weak part of the design and optimisation procedure. This is mainly due to the age of the current methods, poor grouping of some R&M components, selection of only basic R&M drivers, and their high dependence on mass. A new method for the prediction of arising maintenance was produced by Whittle[104], although the new method was found to require data unavailable at the conceptual design stage, and there is some question over the compatibility of the two data sets used to develop the model. The second part of the R&M model, that for scheduled maintenance, is dated, and is heavily mass dominated. This will cause conceptual designs to be driven towards a minimum mass solution, as mass and cost are difficult to discriminate with increases in the number of costing algorithms dependent on mass.

Despite these criticisms, the results of the R&M prediction methodology correspond fairly well with quoted values of aircraft total MMH/FH, and provide a useful driver for the manning levels that will be required during the operation and support phases. However, this area of the cost methodology is one of the areas where the first part of any future development effort should be spent, as it is operation and support that is currently being targeted to deliver the largest LCC savings.

The O&S cost model is based on the structure suggested by the US Cost Analysis and Improvement Group (CAIG) report[9], although a great deal of effort was spent in changing it into a 'generic' US/UK model, so that cost data from both nations could be utilised. The algorithms used are based on several existing models[5,22], although all of them required some modification to conform to the new structure, and to correlate with RAF and VAMOSC costs. Although real data cannot be included here, for security and commercial reasons, the models are fairly accurate, but cannot capture all of the variations that may occur in the operation of individual main operating bases.

One section of the O&S model that could be improved is the calculation of aircraft contract costs, especially for engines and avionics. Avionics O&S costs present the same problems as for development, i.e. lack of data, and rapid advances that prevents one generation of electronics being compared with its predecessor. Engine contract costs are very difficult to estimate because of the shortage of data points, and lack of visibility of the relative component costs within the contracts. This prevents further breakdown of the contracts into personnel and materials costs (as required for ABC), which would certainly be beneficial. During the development of the engine cost models, it was originally intended that the effects of engine de-rating (reducing ultimate engine performance to improve reliability) could be included, with the level of de-rating controlled by the optimiser. Because of problems with the sensitivity of the engine models, as mentioned previously, and difficulty in defining and implementing the concept of de-rating, this was not possible.

The O&S cost model is an extremely useful addition in the estimation of LCC, and, as far as the author is aware, is unique in the public domain. The values it produces appear to be consistent with those from RAF cost reports[11] and USAF VAMOSC data, and include, in a consistent structure, all of the cost components that will be driven by conceptual aircraft design. As has been suggested, many of the models could be improved, but this should be much easier now that the structure, input variables, and output values are better understood.

The final section of the LCC model is the disposal module. It is assumed that the aircraft will be broken up and disposed of, rather than being sold to a third party. The model is based on the production effort values, factored to produce disposal effort. Materials are disposed of separately, incurring either positive or negative costs, depending on whether they have a residual scrap value, or will incur disposal costs. It has not been possible to validate the disposal cost model against real data, although materials disposal values are known to be representative. As disposal costs make up such a small fraction of overall LCC, because of the discounting process, very significant errors would be needed before a change would be seen in the total.

9.2.7 Validation

Validation of the individual models has been discussed with the presentation of the model methodologies, and in previous sections of this discussion. With the exception of the engine thermodynamic cycles, aircraft mission performance, and the disposal cost model, all of the models have been validated, although the extent to which this has taken place was determined by the data available. Avionics and software cost models have proven particularly difficult to check, although it is felt that the values returned by the models are representative. Increased data collection and validation activities would be beneficial, to improve accuracy of the models, and user confidence in them.

9.2.8 Generation of Results

The results of Chapter 8 have shown that the COMBAT aircraft synthesis and optimisation tool developed is working, and is capable of generating meaningful results with varying input parameters. The optimisation tool, RQPMIN, has been linked with the aircraft synthesis and LCC models produced for this study, and drives the available conceptual design parameters to produce converged concepts with minimised values of the objective function being investigated. It has been shown that the optimised design corresponds with the relevant objective function, i.e. optimising an aircraft for minimum LCC results in a different solution to that optimised for minimum mass. A wide variety of design options can be studied, including configuration, procurement, engine, technology, and basing/operational changes; some of these have been demonstrated.

The investigations produced some interesting data, especially the differences between single and twin engine designs, the effects of composite materials on aircraft costs, and the impact of internal weapons carriage. In terms of demonstrating the capability and operation of COMBAT, the aims of the chapter were fulfilled. The results generated/conclusions drawn should provide a basis for debate, and for more detailed studies to be performed on the preferred configurations identified.

9.3 Areas for Further Work

From individual chapters in this thesis, and the discussion above, the reader will already have identified many areas of the work that could be improved. This was one of the intentions of this research, as it was always realised that the issues surrounding aircraft design for minimum LCC would not be resolved in the course of one Doctoral study. Hopefully, readers' interest in the subject will have been stimulated by this research, and that will investigate some of the problems identified. To help focus future development effort, this following sections include a prioritised list of suggested improvements for the COMBAT aircraft design synthesis and optimisation tool.

9.3.1 Reliability and Maintainability Models

As discussed previously, the current R&M models are weak in several respects. To overcome these, new R&M data needs to be collected from any available source, although care should be taken to ensure that only data sets containing consistent maintenance figures are combined. A new R&M model needs to be generated with a

similar number, but better grouping, of aircraft systems, and a more representative set of R&M drivers. It is recommended that two sets of equations be developed - one to predict defect rates, and the other to estimate mean time to repair for the given system. In this way, the number of data points can be significantly increased, by using any available data for defect rate prediction, and making some allowance for variations in the maintenance procedures of different nations to estimate system mean times to repair.

The improvement of this model should increase confidence in the methodology as a whole, and will probably result in larger differences between aircraft optimised for mass and LCC, increasing the value of the tool. This process would have been undertaken during the current study, but for limits of timescales and budgets.

9.3.2 Operation and Support Contract Costs

O&S contract costs make up approximately 40% of all of the in-service costs, by far the largest fraction of all the O&S cost components, but the breakdown and visibility of these costs is the weakest of all O&S expenditure. They also suffer by not including some of the variables that would allow influential trade-off studies to be performed, such as engine de-rating.

The problem with contract costs is that because they are only now being introduced as the preferred form of military aircraft maintenance, little or no data exists for comparison. In addition, because contracts are wholly controlled by commercial organisations that must bid competitively, it is very unlikely that visibility of costs will be achieved without an independent party performing all of the analysis. This approach does not help to disseminate data and encourage discussion of drivers, models and techniques, which is one of the problems identified during this research.

9.3.3 Improvement of LCC Model

Many of the models used in this research are developments of tried and trusted existing algorithms and methodologies. The one main exception to this is the LCC model, much of which was assembled during this research study. Although most of the main elements of LCC have been captured and included, there are other factors that should be built-in before the LCC model can be considered as complete. The two main examples of this are the estimation of risk, and the prediction of timescales.

Risk was mentioned briefly in section 8.3.6, with regard to composite materials. No decision on major expenditure should take place without proper assessment of risk, which is often a measure of the level of technology being utilised. The concept of timescales is also important, as it gives an indication of project lead-times, and allows the calculation of in-year spending. Lead-times are important for project planners, as it allows adequate provision of personnel, facilities, and budgets, to ensure that components are available for inclusion when required, and to reduce risk by allowing sufficient development times. In-year spending is significant for Treasury budgeting, and calculation of these values would allow the capping of annual spend, increasing the realism of the models. Both of these metrics are factors that would add significantly to the usefulness of the LCC tool, and should be included in future development plans.

9.3.4 Investigation of Avionics and Software LCC

The estimation of avionics and software LCC is one of the weakest areas in this study, and is generally recognised as an area requiring further investigation. Many of the avionics life cycle components are based only on mass, and software costs rely on the number of source lines of code (SLOC) being known. Avionics cost development has not been placed higher up this list of improvements because it is unclear to what extent the design of a combat aircraft affects the cost of the avionics suite, which is mainly driven by avionics functionality. That said, many 'mechanical' aspects of avionics cannot currently be modelled, such as the effects of avionics cooling on reliability and engine bleed and size, and the impact of fuselage packing density on avionics maintainability. These modifications would make valuable contributions to the tool.

9.3.5 Upgrade of the Engine Modelling Routines

The ONX and OFFX engine models have proven to be very useful for this research, by allowing the analysis of the effects of engine design on aircraft size and costs, although this has not been demonstrated in the results. However, run times could be reduced, and sensitivity to the design inputs is preventing the optimiser from controlling the engine routines. If these issues could be resolved, and the changes suggested above for engine contract and LCC costs could be implemented, the tool would become much more powerful, by allowing the trade-off between engine technology, reliability, aircraft size, and LCC.

9.3.6 Aircraft 'Advanced Technologies'

COMBAT can currently model most existing combat aircraft designs, but is limited in the number of advanced technology application that can be studied. Applications that are of particular interest are blended wing-body shapes, implementation of low-observable configurations and technologies, thrust vectoring and tail-less designs, uninhabited vehicles, advances in avionics and control, and other emerging technologies identified in the Literature Search of Chapter 2.

Some of these will require significant modifications to the aircraft synthesis, drag, performance, and cost models, but the modular nature of the code should allow these modifications to be made as additions, without the need for a major re-write of large sections of code. One addition that would be particularly useful is the inclusion of agility metrics, as these would allow the comparison of combat effectiveness for competing designs. It should be noted that these changes are suggested as improvements, rather than corrections of perceived deficiencies in the overall model.

9.3.7 Aircraft Synthesis Modifications

Some minor modifications to the aircraft synthesis code might improve the robustness and speed of convergence. One such modification is the removal of the iteration to calculate gross mass, by including empty mass or gross mass as a design variable, and adding an equality constraint to ensure that the calculated empty mass is the same as the input empty mass value. This would reduce the number of internal iterations, and overall run time. Another possible improvement would be the replacement of the 'fuel

fraction' variable with the actual fuel mass value. This would stop the fuel mass changing with variations in the airframe dimensions or engine size, allowing the optimiser more detailed control. This modification is a purely theoretical improvement, as no problems with convergence have been identified as being caused by the use of the fuel fraction variable. As mentioned in section 9.2.2, the synthesis currently always drops taper ratio to the lowest possible value, which is a valid attempt at reducing mass and LCC, but would normally be limited by real-world constraints. However, a constraint to prevent tip stall may be a useful addition to increase user confidence in the aircraft synthesis methodology.

9.4 Conclusions

A software tool has been developed that is capable of generating conceptual design data for various configurations of combat aircraft, and optimising those designs for functions of either minimum mass or cost. This distinguishes the current research study from previous published work by allowing the consistent investigation of alternative design configurations, where previous MVO studies were limited to a single aircraft type. The development and inclusion of a detailed life cycle cost model for military aircraft is a unique feature of this research.

A series of results has been generated to show the effects of changing the aircraft configuration, and the impact of composite materials on the size and cost of competing designs. The influence of changing from a mass to a cost-based objective function, the effect of internal weapons carriage, and the benefits of More Electric Technologies have also been studied. Rational explanations for the changes occurring from one design to the next have been given, and all of the converged designs appear realistic. Many more such analyses are capable of being performed without modification to the code, and the potential for development to generate new studies is excellent. Some of the suggestions in the previous section would improve the ability to investigate the design and cost effects of advanced technologies, allowing conceptual design and balance of investment studies.

It has been shown that aircraft optimised for mass and LCC do not produce the same result, with the LCC optimisation producing savings over the mass-optimised designs in the order of 1.4% and 0.7% of total LCC for single and twin engine aircraft, respectively. For the examples generated, this is equivalent to monetary savings of FY00$M1094 or FY00M$592.7. With development of the R&M and cost models, the effects of mass on operation and support costs should reduce, further increasing the differences in the aircraft designs and cost savings.

In terms of the overall objectives, the research has been very successful. There are many modifications and improvements that can now be implemented to improve the flexibility and utility of the tool, and it is hoped that effort and funding may be become available to allow this to take place. There is still some way to go in the transformation of Life Cycle Costing from an art to a science. Hopefully this thesis will prove a useful addition to the pool of knowledge, and improve the procurement decision process for future air combat systems.

References

1. Abell, J B, Kirkwood, T F, Petruschell, R L, and Smith, G K. (1988). The Cost and Performance Implications of Reliability Improvements in the F-16A/B Aircraft. RAND N-2499-AF. RAND Corporation, Santa Monica, CA.

2. Alexander, A J. (1988). The Costs and Benefits of Reliability in Military Equipment. RAND P-7515. RAND Corporation, Santa Monica, CA.

3. Andrews, H. (1993). ASTOVL - Technology, Capability, Weight, and Cost. AIAA 93-4852.

4. Annear, G G. (1984). Simple Methods for the Prediction of Aero Gas Turbine Weight. Tech. Memo P-1037. Royal Aircraft Establishment. CONFIDENTIAL.

5. Anon. (1980). Naval Aircraft Operating and Support Cost-Estimating Model FY-1979. ASC-R-126. Administrative Sciences Corporation, Falls Church, VA.

6. Anon. (1984). *Military Handbook. Life Cycle Cost Model for Defense Material Systems - Data Collection Workbook.* MIL-HDBK-276-1 (MC). US DoD.

7. Anon. (1991). Economic Appraisal in Central Government - A Technical Guide for Government Departments. Her Majesty's Treasury.

8. Anon. (1991). 'Near-net-shape Production Techniques in Aircraft Construction - Innovative Production Methods Reduce Costs and Enhance Competitiveness'. New-Tech News, no. 2, p. 13-15.

9. Anon. (1992). Operating and Support Cost-Estimating Guide. Office of the Secretary of Defense - Cost Analysis and Improvement Group. The Pentagon, Washington DC.

10. Anon. (1992). *Military Handbook. Cost Engineering: Policy and Procedures.* MIL-HDBK-1010A. US DoD.

11. Anon. (1994). Cost of Support Spreadsheet (COSS) Study - Tornado Fleet. LC/165752/5/1/LSS2/SSMW. Cost of Support Flight, Central Servicing Development Establishment. RESTRICTED.

12. Apgar, H. (1993). Design-to-Life-Cycle-Cost in Aerospace. AIAA 93-1181.

13. Augustine, N R. (1983). *Augustine's Laws and Major System Development Programs.* American Institute of Aeronautics and Astronautics, New York.

14. Bailey, M W, and Overton, K S. (1990). Automated Aircraft Engine Costing Using Artificial Intelligence. AIAA 90-1887.

15. Belbin, G R. (1985). Thermoplastic Structural Composites: A Challenging Opportunity. IN: *Materials and Design*, Vol. 6, No. 1.

References

16. Birkler, J L, Garfinkle, J B, and Marks K E. (1982). Development and Production Cost Estimating Relationships for Aircraft Turbine Engines. N-1882-AF. RAND Corporation, Santa Monica, CA.

17. Boehm, B W. (1981). *Software Engineering Economics*. Prentice-Hall Inc., New Jersey, USA.

18. Bruce, P, and Pederson, S M. (1982). *The Software Development Project*. John Wiley and Sons, New York.

19. Brundick, W. (ed.). (1995). *Parametric Cost Estimating Handbook*. Naval Sea Systems Command, US DoD.

20. Burns, J W. (1991). Weight Estimating Guide for Advanced Composites. SAWE 2020. Society of Allied Weight Engineers.

21. Burns, J W. (1994). Aircraft Cost Estimation Methodology and Value of a Pound Derivation for Preliminary Design Development Applications. SAWE 2228. Society of Allied Weight Engineers.

22. Chaffey, W K T. (1992). The Organisation and Facilities of a Main Operating Base. Working Paper OS34-92-WP-27. Operational Studies Business Sector, Total Air Systems Studies Department, Defence Research Agency Farnborough. RESTRICTED.

23. Chester, R J *et al.* (1993). 'Research into Materials Aspects of Aircraft Maintenance and Life Extension'. Aircraft Engineering, January 1993.

24. Cohen, H, Rogers, G F C, and Saravanamuttoo, H I H. (1996). *Gas Turbine Theory*. Fourth edition. Longman, Harlow, UK.

25. Cole, P A. (1991). The Impact of Manpower, Personnel and Training (MPT) on Life Cycle Cost. In: *NAECON 91; Proceedings of the IEEE National Aerospace and Electronics Conference*, Dayton, OH, May 20-24, 1991. Vol. 2 p. 842-848.

26. Collingbourne, J. (1984). Multivariate Optimisation Applied to the Initial Design of Transport Aircraft. RAE TR 84044.

27. Comey, D H, Culy, D G, and Cassidy, J E. (1978). Trade-Off Studies with an Interactive Engine/Airframe Life-Cycle-Cost Model. SAE 781033.

28. Comptroller and Auditor General (1995). Ministry of Defence: Eurofighter 2000. HC 724 Session 1994-95. National Audit Office. HMSO.

29. Comptroller and Auditor General (1998). Ministry of Defence: Major Projects Report. HC 695 Session 1997-98. National Audit Office. HMSO

30. Daniel, D W. (1987). Reliability and life cycle cost of military aircraft - The vital link. I - The context. In: *Reliability '87; Proceedings of the Sixth Conference*, Birmingham, England, 14-16 April 1987. Volume 1.

References

31. Dean, E B. (1996). Activity Based Cost from the Perspective of Competitive Advantage. http://mijuno.larc.nasa.gov/dfc/abc.html. Design for Competitive Advantage Homepage, Langley Research Center, NASA.

32. Dhillon, B S. (1989). *Life Cycle Costing: Techniques, Models and Applications.* Gordon and Breach, New York.

33. Eberl, R J. (1982). Life Cycle Cost Applications to Conceptual Designs. SAWE 1479. Society of Allied Weight Engineers

34. Feagin, R C, and Morrison, W D. (1978). Delta Method, An Empirical Drag Build-up Technique. NASA-CR-151971. Lockheed-California Co., Burbank, CA.

35. Foye, R L. (1983). A Survey of Reported Weight and Cost Savings for Composite vs. Metal Airframes. SAWE 1547. Society of Allied Weight Engineers.

36. Gerend, R P, and Roundhill, J P. (1970). Correlation of Gas Turbine Engine Weights and Dimensions. AIAA 70-669.

37. Goldsmith, E L, and Seddon, J. (ed.) (1993). *Practical Intake Aerodynamic Design.* Blackwell Scientific Publications, Oxford.

38. Green, C. (1999). 'The Lean Enterprise'. 3rd Sir Frank Whittle Lecture – Royal Aeronautical Society, 2 February 1999. Aerospace, Vol.26, No: 3, March 1999.

39. Gunston, Bill (ed.). (1996). *Jane's Aero Engines.* Jane's Information Group Limited, Coulsden, Surrey, UK.

40. Harmon, D F, Pates, P A, and Gregor, D. (1975). Maintainability Estimating Relationships. In: *Proceedings: 1975 Annual Reliability and Maintainability Symposium.*

41. Herrick, P W. (1989). Fighter Aircraft Affordability, Survivability and Effectiveness Through Multi-Function Nozzles. AIAA 89-2815.

42. Herrmann, O, and Biehl, W. (1985). 'Effect of Engine Technology on Advanced Fighter Design and Cost'. Journal of Aircraft, vol. 23, no. 6, 472-476.

43. Hess, R W, and Romanoff, H P. (1987). Airframe Cost Estimating Relationships: All Mission Types. RAND N-2283/1-AF. RAND Corporation, Santa Monica, CA.

44. Hodder, S D, and Simm, S E. (1995). The Impact of Engine Technology on Combat Aircraft Performance. IN: *AGARD Advanced Aero-Engine Concepts and Control.* 25-29 September 1995, Seattle, USA.

45. Howe, D H, *et al.* (1992-1997). Aircraft Mass Prediction: Powerplants, Systems, and Equipment; & Structural Components. DAeT Lecture Notes 9218, 9317, and Revisions. Cranfield College of Aeronautics, Cranfield University.

References

46. Howe, D H. (1992). Climb and Manoeuvre, Take-Off and Landing Conditions. DAeT Lecture Note 9250 and Revisions. Cranfield College of Aeronautics, Cranfield University.

47. Howe, D H. (1994). Rapid Estimation of Drag Coefficient for Initial Design Work. DAeT 9471. Cranfield College of Aeronautics, Cranfield University.

48. Howe, D H. (1995). Lift Prediction Model. Un-numbered. Cranfield College of Aeronautics, Cranfield University.

49. Howe, D H. (1994). The prediction of Aircraft Wing Mass. Un-numbered. Cranfield College of Aeronautics, Cranfield University.

50. Huenecke, K. (1987). *Modern Combat Aircraft Design.* Airlife Publishing Ltd., Shrewsbury, UK.

51. Jhanjee, P. (1987). 'The Impact of "Double-R / Half-M" on Design'. IN: *IEEE Transactions on Reliability*, Vol. R-36, No. 3, p. 317-320.

52. Johnson, V. S. (1989). Optimising Conceptual Aircraft Designs for Minimum Life Cycle Cost. In: *Recent Advances in Multidisciplinary Analysis and Optimisation*, Part 3 p. 1195-1217, April 1989.

53. Kalviste, J. (1990). Measures of Merit for Aircraft Dynamic Maneuvring. SAE 901005.

54. Kandebo, S W. (1992). 'GE, Pratt Ground Tests Validate Nozzle Concepts'. Aviation Week and Space Technology, Vol. 136 No: 9, March 2, p.42-46.

55. Kandebo, S W, and Hughes, D. (1995). 'F-22 Software on Track with Standard Processes'. Aviation Week and Space Technology Special Report, 'F-22: Air Superiority for the 21st Century', July 24, 38-60.

56. Killingsworth, P S, Jarvaise, J M. (1990). VHSIC Electronics and the Cost of Air Force Avionics in the 1990's. RAND R-3843-AF. RAND Corporation, Santa Monica, CA.

57. Kirkpatrick, D L I, Collingbourne, J. (1973). Multivariate Analysis Applied to Aircraft Optimisation – A Second Progress Report. RAE TM Aero 1553.

58. Kitowski, J V. (1992). Fighter Airframe/Propulsion Integration: A General Dynamics Perspective. AIAA 92-3332.

59. Levenson, G S *et al.* (1972). *Cost-Estimating Relationships for Aircraft Airframes.* RAND RM-4845-PR. RAND Corporation, Santa Monica, CA.

60. Liston, G W, and Small, L L. (1992). Fighter Airframe/Propulsion Integration: A Wright Laboratory Perspective. AIAA 92-3337.

61. Logan, G T. (). Modular Avionics Systems Architecture (MASA).

References

62. Loughlan, J L, Brooks, W, Irving, P, Commander, M W. (1995). *The Stress Analysis and Design of Composite Structures - Short Course 24-28 April 1995.* Cranfield University.

63. Lovell, D A. (1988). The Application of Multivariate Optimisation to Combat Aircraft Design. RAE Technical Report 88003.

64. Mace, J, and Nyberg, G. (1992). Fighter Airframe/Propulsion Integration: A McDonnell Aircraft Perspective. AIAA 92-3333.

65. Mattingly, J D, Heiser, W H, and Daley, D H. (1987). *Aircraft Engine Design.* American Institute of Aeronautics and Astronautics, New York.

66. Mattingly, J D. (1996). *Elements of Gas Turbine Propulsion.* McGraw-Hill Inc.

67. Meitzler, T D. (1988). An Operating and Support Cost Model for Aircraft Engines: SOSCM. AIAA 88-3244.

68. Mishler, R, Wilkinson, T. (1992). Emerging Airframe/Propulsion Integration Technologies at General Electric. AIAA 92-3335.

69. Morris, A J. (ed.) (1982). *Foundations of Structural Optimization: A Unified Approach.* Wiley series in numerical methods in engineering, Chichester, UK.

70. Nemhauser, G L, Rinooy Kan, A H, and Todd, M J (eds.) (1989). *Optimization.* Handbooks in operations research and management science. North-Holland, Amsterdam.

71. Nicolai, L M. (1975). *Fundamentals of Aircraft Design.* METS, Inc. San Jose, California.

72. Palmer, J R. (1983). Turbomatch User Guide. School of Mechanical Engineering, Cranfield University.

73. Peckham, D H. (1971). Multivariate Analysis Applied to Aircraft Optimisation – A First Progress Report. RAE TM Aero 1352.

74. Perry, D H. (1970). Multivariate Analysis Applied to Aircraft Optimisation – Some Proposals and Data for an Exploratory Study. RAE TM Aero 1237.

75. PRICE Systems. PRICE H™ Computer Aided Parametric Estimating for Total Hardware Costs. PRICE Systems, Moorestown, NJ, USA.

76. Pugh, P G. *The Cost Estimator's Task in Aid of Decision-Making.* Lecture Notes, University College, London.

77. Pugh, P G. (1984). How Can It Cost So Much? - An Introduction to Life-Cycle Costs of Modern Military Aircraft. Lecture Note DES 8419. Cranfield College of Aeronautics, Cranfield University.

78. Raymer, D P. (1989). *Aircraft Design: A Conceptual Approach.* American Institute of Aeronautics and Astronautics, New York.

References

79. Resetar, S A, Rogers, J C, and Hess, R W. (1991). Advanced airframe structural materials: a primer and cost estimating methodology. RAND R-4016-AF. RAND Corporation, Santa Monica, CA.

80. Retterer, B L, Criscimagna, N H. (1987). 'Half-Maintainability - A Formidable Goal.' IN: *IEEE Transactions on Reliability*, Vol. R-36, No. 3, p. 324-330.

81. Rodrigues, L J. (1996). NAVY AVIATION. F/A-18E/F will Provide Marginal Operational Improvement at High Cost. GAO/NSIAD-96-98. US General Accounting Office, Report to Congressional Committees.

82. Roskam, J. (1990). *Airplane Design.* Roskam Aviation and Engineering Corporation, Kansas, USA.

83. Rowe, B H. (1995). 'Remembrances and Perspectives: 5 Decades with Aero Engines'. 39th R J Mitchell Lecture, Cranfield University 25 April 1995.

84. Schemensky, R T. (1973). Development of an Empirically Based Computer Program to Predict the Aerodynamic Characteristics of Aircraft. AFFDL-73-144. US Air Force Flight Dynamics Directorate.

85. Schwartz, P. (1987). Use of a Detail Cost Model to Perform Conceptual Phase Cost Analysis. SAWE 1784. Society of Allied Weight Engineers.

86. Serghides, V C. (1985). Development of a Reliability and Maintainability Prediction Methodology for the Aircraft Conceptual Design Process. M.Sc. Thesis, Cranfield University.

87. Serghides, V C, Fielding, J P (1987). 'A Maintainability-Prediction Methodology for Use in Aircraft Design.' IN: *Maintenance Management International*, No. 6, p. 265-272.

88. Siegers, F. (1996). Conceptual Design Synthesis And Optimization For New Generations Of Combat Aircraft. Ph.D. Thesis, College of Aeronautics, Cranfield University.

89. Skira, C A. (1995). Cost Reduction of Advanced Turbine Engines. AIAA 95-3024.

90. Skow, A M. (1990). Agility as a Contribution to Design Balance. AIAA 90-1305.

91. Skrobanski, J J. (1988). The method of RQPMIN (A Program for Constrained Optimization). RAE Working Paper AE (88) WP15. COMMERCIAL-IN-CONFIDENCE.

92. Skrobanski, J J. (1994). RQPMIN Version 3.0 User Guide. The MVA Consultancy.

93. Sieron, T R *et al.* (1993). Procedures and Design Data for the Formulation of Aircraft Configurations. WL-TR-93-3068. Wright Laboratory, Wright-Patterson Air Force Base, OH.

References

94. Smith, H W, and Burnham, R. (1980). The Outside has to be bigger than the inside. AIAA 80-0726R.

95. Spitzer, C R. (1993). *Digital Avionics Systems: Principles and Practice*. Second Edition, McGraw-Hill, USA.

96. Sternberger, N L, *et al.* (1980). Modular Life Cycle Cost Model for Advanced Aircraft Systems, Phase III. AFFDL-TR-78-40. USAF Flight Dynamics Laboratory, Wright-Patterson Air Force Base.

97. Sternberger, N L. (1980). The Modular Life Cycle Cost Model for Advanced Aircraft Systems: An Overview. SAWE 1351. Society of Allied Weight Engineers.

98. Various (ed.). (1975-1998). *Jane's All The World's Aircraft*. Jane's Information Group Limited, Coulsden, Surrey, UK.

99. Vinh, N G. (1993). *Flight mechanics of high-performance aircraft*. Cambridge University Press (Cambridge Aerospace Series; 4).

100. Wanstall, B. (1986). JAS 39 Gripen: A Swedish Solution to a Multi-Role Need. Interavia, August 1986.

101. Warwick, G. (1992). Counting the Cost of Composites. Flight International, 14-20 October 1992, p. 40-42.

102. Whitford, R. (1987). *Design for air combat*. Jane's Publishing Group, London.

103. Whittle, R G, and Serghides, V C. (1996). A Method for the Estimation of Aircraft Powerplant Dimensions. IC-AERO Report 96-03. Imperial College, London.

104. Whittle, R. (1998). Development of the Low Support Vehicle. Ph.D. Thesis. Imperial College, London.

105. Williams, C D. (1990). What is WAT? Wrap Around Test: Maximizing Avionics BIT Utilization to Minimize Flightline Armament System Test Equipment Requirements. SAE 901787.

106. Wilson, R A L. (1997). The Introduction of Laminar Flow to The Design and Optimisation of Transport Aircraft. Ph.D. Thesis, College of Aeronautics, Cranfield University.

107. Yelverton, J N. (1995). Technology-Insertion Life Cycle Cost Model. AIAA 95-0961.

References

This page is intentionally blank.

Aircraft	'macchi MB-339 C	EF 2000	F-4E Phantom	F-5E Tiger II	F-14A Tomcat	F-15E Eagle	F-16C Falcon	F-18C Hornet	F-20 T'shark	F-22 Raptor	FMA 63 Pampa	Hawk 100	Hawk 200	Sepecat Jaguar	MiG-23 Flogger K	MiG-25 Foxbat P	MiG-29 Fulcrum K	MiG-31 Foxh'nd D	Mirage 2000C	Rafale D	Sea Harrier II	SU-27 Flanker B	Tornado GR4	Tornado F3
External Dimensions																								
Length (m)	11.24	15.08	19.20	14.45	19.10	19.43	15.03	17.07	14.42	18.92	10.93	11.84	11.34	15.52	16.71	23.82	16.28	22.69	14.36	15.30	14.17	21.94	16.72	18.68
Span - swept (m)	10.85	10.50	11.77	8.13	11.65	13.05	9.14	11.43	8.13	13.56	9.69	9.39	9.39	8.69	7.78	14.02	11.36	13.46	9.13	10.40	7.70	14.70	8.60	8.60
Span - spread (m)					19.54										13.97								13.91	13.91
% length to cockpit rear		29.4%	37.7%	41.2%	38.1%	35.9%	31.2%	31.5%	41.7%	25.8%				30.4%	29.3%				32.1%				30.0%	36.1%
Height (m)	3.994	6.40	5.02	4.07	4.88	5.63	5.09	4.66	4.22	5.00	4.29	3.98	4.13	4.89	4.82	6.10	4.73	6.15	5.20	5.34	3.71	5.93	5.95	5.95
Effective Diameter	1.393	2.08	2.34	1.57	2.75	2.41	1.90	2.08	1.81	3.28	1.86	1.84	1.90	1.78	1.85	2.68	2.32	2.73	1.60	2.08	2.11	2.84	2.214	2.21
Tailplane span (m)	4.08			4.31	9.97			6.58	4.72	8.84	4.58			4.53	5.51		7.78							
Wheel track (m)	2.483	3.83	5.45	3.80	5.00	2.75	2.36	3.11	3.80	3.28	2.66	3.47	3.47	2.41	2.66	3.85	3.09		3.40		6.76	4.36	3.10	3.10
Wheel base (m)	4.369	3.74	7.42	5.17	7.02	5.42	4.00	5.42	5.30	6.04	4.42	4.50	3.56	5.69	5.77	5.14	3.65		5.00		3.45	5.88	6.20	6.20
Ultimate load factor	11.1	13.5	9.8	9.8	9.8	11.0	11.0	11.3	13.7	13.7	9.2	12.2	12.2	12.0	12.9	6.9	13.7	7.7	13.5	13.5	11.9	13.7	11.4	11.4
Number of Crew	2	1	2	1	2	2	1	1	1	1	2	2	1	1	1	1	1	1	1	1	1	1	2	2
Number of Engines	1	2	2	2	2	2	1	2	1	2	1	1	1	2	1	2	2	2	1	2	1	2	2	2
Areas (m^2)																								
Wing Area Swept	19.3	50.00	49.24	17.28	52.49	56.50	27.87	37.16	18.58	78.04	15.63	16.69	16.69	24.18	34.16	61.40	38.00	61.60	41.00	46.00	18.68	62.00	26.60	26.60
Ailerons (total)	1.328			0.86		2.46	5.82	2.27			0.89	0.97	0.97			2.72								
Leading-edge flaps				1.14	4.29		6.82	4.50		4.76				1.05										
Trailing-edge flaps	2.21			1.95	9.87	3.33	5.82	5.75		5.10	2.93	2.50	2.50	4.12	5.90	4.30								
Vertical tail(s) (total)	2.37	4.78	5.70	3.85	7.90	9.78	4.00	9.68	3.3	16.54	1.86	3.19	3.19	3.90	4.83	16.00	11.80	16.40	4.54	5.50	2.59	15.40	9.65	9.65
Rudders (total)	0.61			0.57	3.06	1.85	1.08	1.45		5.09	0.655	0.58	0.58			2.12					0.49			
Horizontal surfaces	4.36	2.40	7.85	5.48	13.01	10.34	5.92	8.18	3.99	12.63	4.35	4.33	4.33	7.80	6.01	9.81	7.26	9.20	0.06	2.42	4.41	12.30	9.45	9.45
Spoilers (total)					1.97									0.90	1.20									
Aspect Ratio (swept)	6.10	2.21	2.81	3.83	2.59	3.01	3.00	3.52	3.56	2.36	6.00	5.28	5.28	3.12	1.77	3.20	3.40	2.94	2.03	2.35	3.17	3.49	2.78	2.78
Aspect Ratio (spread)					7.27										5.22								7.27	7.27
Taper Ratio	0.592	0.150	0.178	0.238	0.324	0.290	0.235	0.365	0.250	0.100	0.570	0.366	0.311	0.244	0.444	0.373	0.264	0.285	0.079	0.214	0.285	0.304	0.520	0.520
Leading-edge sweep	10.84	53	45	31.22	68	44.53	40	26.39	32.41	42	2.61	26	26	45.95	74	41	42	40	58	46	40.36	42	67	67
Quarter-chord Sweep	8.50	44.77	36.9	24	66.3	38.7	32.31	20	25	28.95	0.00	21.5	21.5	40	73.00	32	36.09	32.00	49.73	37.24	34	36.77	65.96	65.96
Average thickness/chord	0.135	0.050	0.051	0.048	0.109	0.048	0.040	0.053	0.048	0.051	0.135	0.100	0.100	0.063	0.070	0.050	0.048	0.058	0.045	0.057	0.075	0.066	0.157	0.157
Masses (kg)																								
Empty	3310	9999	13757	4410	18951	14515	8663	10455	5965	14580	2821	4400	4450	7000	10200	19960	10900	21825	7500	9060	6374	15000	13890	14500
Ext. Weapons (max.)	1814	6500	7250	3175	6577	11113	5443	7031	4080	9071	1550	3000	3000	4763	3000	1900	3000	1900	6300	8000	3630	3420	9000	8500
Take-off (max.)	6350	20250	28030	11214	33724	36741	19187	22328	12700	27216	5000	9100	9100	15700	17800	36720	18500	41000	17000	21500	11880	30500	27950	27986
Take-off (normal)	4884		23424		29072		15759	16651			3800			10954			15240		13930			22000	20411	20411
Landing (max.)		16850	20865	5745	22895	20094	10455	13775	7265		3630	7650	7650	10954	11800	30260	13665	31300	10161	12000	7770	19250	19935	19935
Thrust - Max. (kN)	19.6	180.0	159.2	44.5	240.2	258.8	122.8	142.4	80.0	310.0	15.6	26.2	26.2	74.8	127.5	220.0	162.8	303.8	95.1	145.8	95.6	245.2	143.0	147.0
Thrust - Mil. (kN)	13.0	120.0	106.2	32.0	143.1	164.6	87.7	94.4	53.3	206.7	10.4	23.1	23.1	49.2	78.4	182.0	98.8	196.0	64.3	97.4	95.5	163.0	77.0	81.0
Fuel - Internal (kg)	1388	4000	5900	1996	7348	5280	3162	4926	2290	9979	1118	1304	1360	3360	4500	14570	4640	14520	3160	4000	2295	6500	5103	5670
Fuel - External (kg)	520			2415	1724	5310	3066	3053	2919		1050	932	932	960	1860	4370	907	3120	3720		2404	3310	3600	6530
Fuel Fraction (Unloaded)	0.30	0.29	0.30	0.31	0.28	0.27	0.27	0.32	0.28	0.41	0.28	0.23	0.23	0.32	0.31	0.42	0.30	0.40	0.30	0.31	0.26	0.30	0.27	0.28
T/W (Take-off)	0.314	0.906	0.579	0.404	0.726	0.718	0.652	0.650	1.122	1.161	0.317	0.293	0.293	0.486	0.730	0.611	0.897	0.755	0.570	0.691	0.820	0.952	0.522	0.535
T/W (Combat)	0.271	0.802	0.532	0.466	0.563	0.739	0.690	0.586	0.594	0.874	0.255	0.360	0.356	0.453	0.573	0.658	0.684	0.665	0.536	0.659	1.043	0.832	0.375	0.383
Performance																								
M_{MAX} - high altitude	0.77	2.00	2.10	1.64	1.88	2.50	2.00	1.80	2.00	2.00	0.66	0.88	0.88	1.60	2.35	2.83	2.30	2.83	2.20	2.00	1.25	2.35	2.20	2.20
M_{MAX} - low altitude	0.74	1.20	1.20	0.95	1.20	1.20	1.20	1.01	1.20	1.20	0.61	0.82	0.83	1.10	1.10	0.98	1.06	1.23	1.20	1.14	0.97	1.10	1.20	1.21
Take-off speed (m/s)	100															100.3	61.2							82.3
Landing speed (m/s)	98	51.4			55.9	59.4		57.4			61.7	59.3	59.3	59.2	64.7	67.3	60.0		64.3	49.3			59.2	59.2
Ceiling (m)	14240		16750	15600	17100	19800	15250	15240	17315	15240	12900	13565	13720		18500	20700	17000	22900	16460		15600	18000	15250	21335
Climb rate (m/s)	36		163	174	152	254	254	254	273.3		26	60	60		240	208	330	208	284	300	254	305	203	203
Take-off Roll (m)	490	500	1338	1737	762	275	533	427	1082	1067	424	640	640	565	500	1250	250	1200		600	305	500	760	760
Landing Roll (m)	455	500	1152		732		808		655	1067	461	605	605	680	750	800 chute	600 chute	800	410		0	600	370	370
First Flight	Dec-85	Mar-94	Feb-67	Aug-72	Dec-70	Jul-72	Jun-84	Jun-74	Aug-82	Oct-90	Oct-84	May-86	May-86		1965	Aug-64	1977	1975	Mar-78	Jul-86	Aug-78	1977	Oct-79	Oct-79

Appendix A

Aircraft	'macchi MB-339 C	EF2000	F-4E Phantom	F-5E Tiger II	F-14A Tomcat	F-15E Eagle	F-16C Falcon	F-18C Hornet	F-20 T'shark	F-22 Raptor	FMA 63 Pampa	Hawk 100	Hawk 200	Sepecat Jaguar	MiG-23 Flogger K	MiG-25 Foxbat P	MiG-29 Fulcrum K	MiG-31 Foxh'nd D	Mirage 2000C	Rafale D	Sea Harrier II	SU-27 Flanker B	Tornado GR4	Tornado F3
Take-off Ground Run Estimate																								
Take-off Mass	4884	20250	28030	11214	33724	20557		16651	12700		3700	5794	5900	10954	17800	36720	15240	46200		19500				27986
W/S for take-off	253.1	405.0	569.3	649.0	642.5	363.8		448.1	683.5		236.7	347.2	353.5	453.0	476.6	598.0	401.1	750.0		423.9				1052.1
T/W for take-off	0.408	0.906	0.579	0.404	0.726	1.283		0.872	0.642		0.429	0.461	0.453	0.696	0.730	0.611	1.089	0.670		0.762				0.535
Aspect Ratio	6.10	2.21	2.81	3.83	7.27	3.01	3.00	3.52	3.56	2.36	6.00	5.28	5.28	3.12	5.22	3.20	3.40	2.94	2.03	2.35	3.17	3.49	7.27	7.27
Useable lift coefficient	1.596	0.991	0.957	1.368	1.385	0.964	1.182	1.360	1.325	1.124	1.670	1.433	1.433	1.274	1.577	0.971	1.109	1.098	0.716	1.056	1.186	1.191	1.768	1.566
Actual Ground Run	490	500	1338	1737	762	275		427	1082		424	640	630	565	500	1250	250	1200		600				1525
Calculated Ground Run	544	478	1274	1649	732	276		406	960		455	705	736	594	473	1227	330	1200		593				1599
Real % Error	11.0%	-4.4%	-4.8%	-5.1%	-4.0%	0.3%		-4.9%	-11.2%		7.3%	10.2%	16.8%	5.0%	-5.4%	-1.8%	32.0%	0.0%		-1.1%				4.8%
Landing Run Estimate																								
Landing Mass (kg)	4884	16850	20865	5230	22895	20244	10455	13775	8409		3500	4600	4600	7762	11800	30260	13665	33000	10161	12000	7770		14500	14500
Landing W/S (kg/m^2)	253.1	337.0	423.7	302.7	436.2	358.3	375.1	370.7	302.2		223.9	275.6	275.6	321.0	315.9	492.8	359.6	535.7	247.8	260.9	416.0		545.1	545.1
Reverse Thrust Efficiency	0	0.5	0	0	0	0	0	0	0		0	0	0	0	0	0.48	0.4	0.5	0	0.5	0.5		0.800	0.8
Landing C_L	2.387	1.310	1.488	1.803	2.290	1.498	1.786	2.021	1.975	1.490	2.532	2.172	2.172	1.962	2.008	1.433	1.615	1.601	0.946	1.408	1.582		2.347	2.347
Actual Landing Run	455	500	1152	762	732	1067			655		461	605	605	680	750	800	600	800					370	370
Calculated Landing Run	467	537	1255	739	839	1054			674		390	559	559	721	693	730	509	698					396	396
Real % Error	2.6%	7.4%	8.9%	-3.0%	14.7%	-1.3%			2.9%		-15.5%	-7.6%	-7.6%	6.0%	-7.6%	-8.7%	-15.2%	-12.7%					7.2%	7.2%
Tailplane Sizing Correlation																								
Tailplane Size Factor	1	0.2535	1	1	1	1	1	1	1	1	1	1	1		1	1	1	1		0.2535	1	1	1	1
Length (m)	11.24	15.08	19.20	14.45	19.10	19.43	15.03	17.07	14.42	18.92	10.93	11.84	11.34		16.71	23.82	16.28	22.69		15.30	14.17	19.85	16.72	18.68
Tailplane Area (m^2)	4.36	2.4	7.85	5.48	13.01	10.34	5.92	8.18	3.99	12.63	4.354	4.33	4.33		6.01	9.81	7.26	9.2		2.42	4.41	12.3	9.45	9.45
Calc. Tailplane Area (m^2)	4.83	2.37	8.65	4.13	13.00	9.60	5.93	7.17	4.37	12.45	4.12	4.25	4.30		6.75	9.68	7.40	9.84		2.22	4.41	10.26	9.48	9.30
Real % Error	10.7%	-1.1%	10.2%	-24.7%	-0.1%	-7.1%	0.2%	-12.3%	9.5%	-1.4%	-5.3%	-2.0%	-0.8%		12.4%	-1.3%	1.9%	7.0%		-8.5%	0.0%	-16.5%	0.3%	-1.6%
Fin Sizing Correlation																								
Number of fins	1	1	1	1	2	2	1	2	1	2	1	1	1	1	1	2	2	2	1	1	1	2		
Total Fin Area (m^2)	2.98	4.78	5.7	3.85		11.63	5.08	11.13	3.3	16.54	2.518	3.77	3.77	3.9		18.12	11.8	16.4	4.54	5.5		15.4		
Calculated Fin Area (m^2)	3.00	5.03	6.25	3.89		13.51	4.52	11.14	4.28	16.46	3.60	3.77	3.74	4.46		16.77	11.64	16.43	3.92	4.92		15.40		
Real % Error	0.7%	5.2%	9.7%	1.0%		16.2%	-10.9%	0.1%	29.7%	-0.5%	42.9%	0.0%	-0.7%	14.3%		-7.4%	-1.3%	0.2%	-13.5%	-10.6%		0.0%		
Mass Estimation																								
Design Diving Speed	350	408	408	323	408	408	408	408	408	408	208	279	282	374	374	333	361	419	408	388	330	374	408	412
C_1 Wing Mass Constant	8.60E-4	5.72E-4	6.00E-4	5.80E-4	5.95E-4	5.80E-4	5.80E-4	6.00E-4	5.80E-4	5.80E-4	9.75E-4	6.27E-4	6.27E-4	5.80E-4	7.86E-4	5.80E-4	5.80E-4	5.80E-4	6.21E-4	5.54E-4	5.80E-4	5.80E-4	6.65E-4	6.64E-4
Comp. Wing Mass Saving	5.0%	16.0%	0.0%	0.0%	2.0%	5.0%	0.0%	2.0%	0.0%	10.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	15.0%	0.0%	0.0%	0.0%	0.0%
Comp. Fuse. Mass Saving	0.0%	22.0%	0.0%	2.0%	2.0%	5.0%	0.0%	2.0%	2.0%	15.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	2.0%	15.0%	0.0%	0.0%	0.0%	0.0%
Comp. Tail Mass Saving	0.0%	10.0%	0.0%	0.0%	20.0%	20.0%	20.0%	30.0%	20.0%	30.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	30.0%	30.0%	0.0%	0.0%	0.0%	0.0%
Wing Mass	651.1	1723.3	2693.4	707.2	4891.2	1891.9	1665.3	1590.2	936.6	2318.4	379.1	547.8	527.9	1175.3	3974.2	3910.3	1930.5	4136.2	1863.8	1554.8	608.1	2272.6	3297.5	3311.5
Fuselage Mass	567.2	1513.1	3098.9	916.0	4121.6	3118.8	1543.7	2280.9	1341.9	4270.6	569.0	782.9	778.3	1401.5	1648.1	4939.7	2169.4	5607.6	1113.4	1509.1	1429.3	4075.3	2314.7	2750.5
Empennage Mass	149.3	225.3	363.4	183.9	276.9	519.3	226.4	388.8	138.2	638.4	82.2	133.9	135.6	74.7	125.6	707.0	483.5	799.3	113.1	186.3	76.4	785.7	235.8	237.8
Undercarriage Mass	264.9	778.9	1053.9	449.5	1251.6	1355.4	740.8	852.9	504.7	1025.4	212.1	370.2	370.2	614.7	690.8	1354.7	716.1	1501.0	661.9	823.5	474.3	1140.0	1051.1	1052.3
Engine Mass	326.2	2350.0	3450.0	612.0	3200.0	3582.0	1791.0	2070.0	990.0	3000.0	295.7	582.0	603.0	1476.0	2005.0	4400.0	2110.0	4832.0	1500.0	1794.0	1615.0	3060.0	2123.0	2147.2
Auxiliary Power Unit		235.0						207.0		300.0							211.0			179.4			212.3	214.7
Fuel Tanks & System	146.5	449.8	626.1	248.0	756.0	782.3	417.0	503.1	280.7	671.8	116.8	197.5	198.3	353.7	410.7	921.2	426.1	1001.8	375.4	473.5	265.1	682.0	612.6	621.8
Power & Flying Controls	121.2	320.5	415.7	191.1	482.0	516.2	293.6	346.6	211.1	388.4	100.1	161.7	161.7	261.5	289.1	516.0	298.2	563.5	278.6	321.6	200.1	444.8	414.8	415.2
Hydraulics & Pneumatics	127.5	227.7	267.9	169.4	293.8	306.7	221.6	239.1	180.3	264.0	113.1	152.6	152.6	200.5	213.5	306.6	217.6	324.0	208.6	234.6	174.4	279.4	267.5	267.7
Electrics & Batteries	234.7	248.6	326.7	151.3	381.6	410.1	237.6	269.9	168.0	318.7	192.0	317.5	317.5	200.8	223.1	409.9	230.4	449.7	214.6	261.5	158.9	350.7	325.9	326.3
Environmental Control	90.6	204.2	341.7	122.0	404.4	437.3	194.8	222.5	135.9	265.0	73.5	124.5	101.9	163.4	182.4	345.8	188.6	381.5	175.2	215.3	128.2	293.2	340.8	341.2
Equip., Controls, Furn.	314.3	334.1	484.2	263.3	520.0	540.2	293.5	340.7	249.8	391.0	304.3	329.6	226.8	294.0	288.6	440.8	317.9	466.3	286.8	339.6	244.9	403.4	471.8	472.5
Avionics	381.8	606.9	1010.0	460.6	825.2	810.6	589.7	697.6	484.6	719.8	359.9	426.3	426.3	533.2	567.3	873.8	578.6	943.1	554.3	627.2	471.3	773.0	1100.0	1100.0
Fixed Armament Items	181.4	650.0	725.0	317.5	657.7	1111.3	544.3	703.1	408.0	907.1	155.0	300.0	300.0	476.3	300.0	190.0	300.0	190.0	630.0	800.0	363.0	342.0	900.0	850.0
Actual Empty Mass	3310.0	9999.0	13757.0	4410.0	18951.0	14515.0	8663.0	10455.0	5965.0	14580.0	2821.0	4400.0	4450.0	7000.0	10200.0	19960.0	10900.0	21825.0	7500.0	9060.0	6374.0	15000.0	13890.0	14500.0
Calculated Empty Mass	3556.5	9867.3	14856.9	4791.8	18062.0	15382.0	8759.3	10712.5	6029.7	15478.5	2952.7	4426.3	4299.9	7225.6	10918.3	19315.8	10178.0	21196.0	7975.9	9320.5	6209.1	14902.0	13667.8	14108.7
Real % Error	7.4%	-1.3%	8.0%	8.7%	-4.7%	6.0%	1.1%	2.5%	1.1%	6.2%	4.7%	0.6%	-3.4%	3.2%	7.0%	-3.2%	-6.6%	-2.9%	6.3%	2.9%	-2.6%	-0.7%	-1.6%	-2.7%
% Advanced Materials	5.7%	33.6%	0.0%	1.7%	6.5%	10.7%	1.7%	7.4%	3.6%	32.3%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	0.0%	2.7%	28.7%	0.0%	0.0%	0.0%	0.0%

Manufacturer	Engine	Mk No:	A/B	No: Shafts	Max. S.L.S Thrust (kN)	Mil. S.L.S Thrust (kN)	TSFC Max. Power (mg/Ns)	TSFC Mil. Power (mg/Ns)	Max. Fan Press. Ratio	Max. Comp. Press. Ratio	Basic Length (m)	Max. Length (m)	D_I (m)	D_{MAX} (m)	Mass Flow Rate (kg/s)	BPR	Max. TIT (°K)	Mass (kg)	IOC Year	Calc D_I (m)	% Error	Calc. D_{MAX} (m)	% Error	Calc. Basic Length (m)	% Error	Calc. Max. Length (m)	% Error	Calc. Mass (kg)	% Error	Application
Allied Signal	TFE1042	6	0	2	21.5	21.5	X	19.40		14.0	1.625	X		0.686	40.8	0.84		395.0		0.557		0.723	5.4%	1.681	3.5%	1.681	3.5%	430.1	8.9%	
Allied Signal	TFE1042	70	1	2	41.2	22.7		22.66		14.0	1.925	2.880	0.591	0.782	43.3	0.40		617.0	1989	0.574	-2.9%	0.742	-5.2%	1.838	-4.5%	3.234	12.3%	626.9	1.6%	Ching Kuo IDF
Allied Signal	TFE109	1	0	2	5.9	5.9	X	11.10			1.130	X		0.597				199.0	1982											Jet Squalus 1300
Aviadvigatel	D30	F6	1	2	186.1	93.2	53.80	20.40	3.00	21.2	3.499	7.040	1.020		150.0	0.57	1660	2416.0	1976	1.068	4.7%	1.282		3.253	-7.0%	6.186	-12.1%	2546.8	5.4%	MiG-31
Eurojet	EJ200		1	2	88.6	58.4	47.68	22.59		26.0	2.096	4.026	0.719	0.851	73.8	0.41	1830	1011.5	1994	0.749	4.2%	0.938	10.2%	2.216	5.7%	3.913	-2.8%	1026.9	1.5%	EF 2000
General Electric	F101	GE-100	1	2	133.5	75.7				26.5		4.596		1.397	159.0	2.01	1644	1814.0	1974	1.100		1.317	-5.7%	2.681		4.594	0.0%	1791.7	-1.2%	B-1
General Electric	F101	GE-102	1	2	136.9	75.6	69.70	15.92	2.31	26.8		4.590		1.402	161.5	1.91	1672	2017.6	1975	1.108		1.326	-5.4%	2.734		4.719	2.8%	1857.2	-8.0%	B-1B
General Electric	F103	GE-100-E1	0	2	233.5	233.5	X	10.65		30.3	4.394	X	2.195	2.390	673.0	4.40	1639	3851.0	1973	2.262	3.1%	2.489	4.1%	4.102	-6.7%	4.102	-6.7%	3829.4	-0.6%	747-200
General Electric	F110	GE-100	1	2	124.6	78.1	58.90	19.06	2.98	30.4	2.685	4.630		1.180	122.4	0.76		1492.0	1984	0.965		1.172	-0.6%	2.698	0.5%	4.831	4.3%	1652.0	10.7%	F-15, F-16
General Electric	F110	GE-129	1	2	138.0	75.7	48.16	18.13		30.7	2.685	4.620		1.180	122.5	0.76		1791.0	1988	0.965		1.173	-0.6%	2.695	0.4%	4.838	4.7%	1648.9	-7.9%	F-16C/D, F-15E
General Electric	F110	GE-400	1	2	119.3	71.6				30.4	2.685	5.900		1.180	120.2	0.76		1599.0	1987	0.956		1.163	-1.4%	2.672	-0.5%	4.782	-18.9%	1614.1	0.9%	F-14 D
General Electric	F118	GE-100	0	2	84.5	84.5	X	18.98		35.1	2.553	X		1.180				1452.0	1987											B-2, TR-1
General Electric	F404	GE-400	1	2	71.2		52.40		3.90	25.0	2.413	4.030	0.704	0.876	64.4	0.34	1645	989.0	1978	0.700	-0.6%	0.883	0.8%	2.093	-13.2%					F-18A/B
General Electric	F404	GE-400-D	0	2	48.0	48.0	X	22.66		25.0	2.413	X	0.704	0.876				814.2	1986											A-6F
General Electric	F404	GE-402	1	2	78.8	53.3	49.29	23.79	4.30	27.0	2.413	4.030	0.704	0.876	66.2	0.27	1778	1035.0	1990	0.710	0.8%	0.894	2.0%	2.119	-12.2%	3.728	-7.5%	941.3	-9.1%	F-18C/D
General Electric	J79	GE-19	1	1	79.2	52.5	55.65	24.08	X	13.4		5.301	0.978	0.992	77.1	0.00	1261	1742.5	1970	0.766	-21.7%	1.004	1.2%	2.868		5.247	-1.0%	1356.6	-22.1%	F-4J, RA-5C
General Electric	J85	21	1	1	22.2	15.6	60.30	23.80	X	8.3	1.186	2.858	0.449	0.533	23.8	0.00	1250	306.0	1972	0.425	-5.2%	0.599	12.3%	1.658		2.817	-1.4%	353.5	15.5%	F-5E/F
General Electric	TF34	GE-100	0	2	40.3	40.3	X	10.50	1.50	21.0	2.565	X	1.150	1.298	153.0	6.23	1498	653.2	1970	1.079	-6.2%	1.298	0.0%	1.559		1.559		845.9		A-10
General Electric	TF39	GE-1	0	2	181.5	181.5	X	16.43	1.56	26.0	4.785	X	2.400	2.540	702.6	8.00	1589	3316.0	1968	2.311	-3.7%	2.540	0.0%	2.670		2.670		3316.0	0.0%	C-5A, C-5B
Klimov	RD33		1	2	81.4	49.4	58.06	20.95		20.0		4.229		1.000	76.0	0.40	1670	1055.0	1975	0.760		0.950	-5.0%	2.344		4.229	0.0%	1135.9	7.7%	MiG-29
Lyulka Saturn	AL21	F3A	1	1	110.5	76.5	52.50	21.47	X	14.8		5.160	0.885		104.0	0.00	1385	2005.0	1973	0.889	0.5%	1.146		3.309		6.180	19.8%	1992.0	-0.6%	Su-24
Lyulka Saturn	AL31	F	1	2	122.6	79.4		18.98		23.0		4.950		1.220	110.0	0.60	1700	1530.0	1983	0.915		1.118	-8.3%	2.716		4.950	0.0%	1630.0	6.5%	Su-27, 30, 35
Pratt & Whitney	F100	PW-100	1	2	106.0	65.2	72.23	19.26	3.10	23.0	2.699	4.855	0.928	1.194	101.2	0.70	1672	1375.0	1973	0.877	-5.5%	1.078	-9.7%	2.565	-5.0%	4.615	-5.0%	1417.1	3.1%	F-15A
Pratt & Whitney	F100	PW-220	1	2	104.3	63.9		20.40	3.10	24.8	2.699	4.855		1.181	103.4	0.70	1672	1444.0	1985	0.887		1.089	-7.8%	2.566	-4.9%	4.604	-5.2%	1430.1	-1.0%	F-15C/D
Pratt & Whitney	F100	PW-229	1	2	129.4	79.2		20.11	3.80	32.0	2.699	4.855	0.884	1.181	112.5	0.36	1755	1681.0	1988	0.925	4.6%	1.129	-4.4%	2.700	0.0%	4.909	1.1%	1682.1	0.1%	F-15, F-16
Pratt & Whitney	J57	P-59W	1	2	61.1	51.1		28.16		12.5		4.289		0.987	91.0	0.00		1959.0	1950	0.832		1.028	4.1%	2.916		5.622		1946.5	-0.6%	F-100
Pratt & Whitney	TF33	P-3	0	2	75.6	75.6	X	14.73	1.70	13.0	3.462	X		1.349	204.1	1.40	1144	1769.0	1960	1.246		1.469	9.0%	3.690	6.6%	3.690	6.6%	1871.5	5.8%	B-52H
Pratt & Whitney	TF33	P-7	0	2	93.4	93.4	X	15.85		15.6	3.607	X		1.372	209.0	1.36	1203	2088.0	1971	1.261		1.485	8.2%	3.654	1.3%	3.654	1.3%	1880.1	-10.0%	C-141A/B
Pratt & Whitney	J75	P-13B	0	2	75.9	75.9	X				4.858	X		1.092	136.0	0.00		2381.0		1.017		1.226	12.3%							U-2, TR-1
Pratt & Whitney	J75	P-17	1	1	109.0	71.6	60.90	23.23	X	11.9		6.035		1.092	114.8	0.00	1150	2665.0	1958	0.934		1.196	9.5%	3.600		6.893	14.2%	2454.6	-7.9%	F-106A
Pratt & Whitney	J75	P-19W	1	2	118.3							6.502		1.092	136.0	0.00		2703.4		1.017		1.226	12.3%							F105D & later
Pratt & Whitney	J52	P-8	0	2	41.4	41.4	X	24.35		12.6	3.018	X		0.766		0.00		961.0												A-6,EA-6A,A-4E
Pratt & Whitney	J52	P408	0	1	49.8	49.8	X	25.21	X	14.6	3.020	X		0.815		0.00	1283	1051.4	1970											A-4M, EA-6B
Pratt & Whitney	J58	P-1	1	1	153.5		61.57			8.8		5.377		1.407	166.0	0.00		2869.5	1959	1.124		1.407	0.0%	4.577						SR-71
Pratt & Whitney	J60	P-4	0	1	10.4	10.4	X	28.15	X	6.5	1.981	X		0.556		0.00		212.0												
Pratt & Whitney	PW1120		1	2	91.7	60.3	52.65	22.70		26.8				1.021	80.9			1292.0	1984	0.784										Lavi
Pratt & Whitney	F119		1	2	157.5														1991										F-22
Pratt & Whitney	TF30	P-100	1	2	111.7	64.8	69.40	19.55	2.40	17.1		6.139		1.242	118.0	0.70	1589	1807.0	1971	0.947		1.154	-7.1%	2.909		5.417	-11.8%	1870.4	3.5%	F-111A,C,E
Pratt & Whitney	TF30	P-111	1	2	111.7		69.40		2.40	17.1		6.139		1.242	118.0		1513	1824.0	1971	0.947										F-111F
Pratt & Whitney	TF30	P-414A	1	2	93.0		78.75		2.40	19.8		5.987		1.293	112.0		1513	1928.0		0.923										F-14A
Pratt & Whitney	TF30	P-9,9A,109	1	2	92.7	55.3	74.10	19.83	2.20	18.1		6.137		1.245	110.2	0.80	1411	1862.0	1968	0.916		1.120	-10.1%	2.746		5.026	-18.1%	1631.8	-12.4%	FB-111D
Rolls-Royce	Adour	811	1	2	37.4	24.6	60.30	24.55	2.50	11.4	1.956	2.970	0.559	0.762	43.3	0.73		738.0	1981	0.574	2.7%	0.742	-2.6%	1.815	-7.2%	3.153	6.2%	590.2	-20.0%	Sepecat Jaguar
Rolls-Royce	Adour	861	0	2	25.4	25.4	X	21.16		11.3	1.956	X	0.559	0.762	43.6	0.75		582.0	1981	0.576	3.0%	0.744	-2.3%	1.820	-7.0%	1.820	-7.0%	482.3	-17.1%	BAe Hawk 100
Rolls-Royce	Adour	871	0	2	26.0	26.0	X	21.95	2.80	11.2	1.956	X	0.559	0.762	44.0	0.80	1550	603.0	1988	0.578	3.5%	0.747	-1.9%	1.819	-7.0%	1.819	-7.0%	481.8	-20.1%	BAe Hawk 200
Rolls-Royce	Pegasus	11	0	2	95.7	95.7	X	20.96	2.30	14.0	2.515	X		1.219	196.0	1.40		1462.9	1971	1.221		1.443		3.571		3.571		1513.1	3.4%	Harrier, +Sea
Rolls-Royce	Pegasus	11-21	0	2	97.9	97.9	X	20.67	2.30	15.2	2.515	X		1.219	206.0	1.40		1467.4		1.252		1.475		3.622		3.622		1570.7	7.0%	AV-8B
Rolls-Royce	Pegasus	11-61	0	2	105.9	105.9	X	19.80		15.6	2.515	X		1.219	209.0	1.40		1614.8	1990	1.261		1.485		3.635		3.635		1587.2	-1.7%	BAe Harrier II
Rolls-Royce	Spey	101	0	2	49.1	49.1	X	17.84		16.5	2.908	X	0.826		91.6	0.96		1120.0	1961	0.835	1.0%	1.032		2.473	-15.0%	2.473	-15.0%	900.8	-19.6%	Buccaneer
Rolls-Royce	Spey	202	1	2	92.7	54.5		17.85		20.0	2.985	5.207	0.826		92.5	0.62		1856.6	1965	0.839	1.6%	1.036		2.525	-15.4%	4.575	-12.1%	1344.7	-27.6%	Phantom FG.1
Rolls-Royce	Viper	531	0	1	13.9	13.9	X	28.30	X	5.6	1.806	X		0.749	25.0	0.00		345.0		0.436		0.612	-18.3%	1.806	0.0%	1.806	0.0%	321.6	-6.8%	
Rolls-Royce	Viper	632	0	1	17.8	17.8	X	27.48	X	6.0	1.806	X		0.749	26.3	0.00	1240	358.0	1968	0.447		0.626	-16.5%	1.836	1.6%	1.836	1.6%	334.7	-6.5%	MB 326K
Rolls-Royce	Viper	680-43	0	1	19.4	19.4	X	27.76	X	6.7	1.960	X		0.749	28.8	0.00	1260	360.0		0.468		0.651	-13.1%	1.894	-3.4%	1.894	-3.4%	360.0	0.0%	MB 339B/K

Manufacturer	Engine	Mk No:	A/B	No: Shafts	Max. S.L.S Thrust (kN)	Mil. S.L.S Thrust (kN)	TSFC Max. Power (mg/Ns)	TSFC Mil. Power (mg/Ns)	Max. Fan Press. Ratio	Max. Comp. Press. Ratio	Basic Length (m)	Max. Length (m)	D_I (m)	D_{MAX} (m)	Mass Flow Rate (kg/s)	BPR	Max. TIT (°K)	Mass (kg)	IOC Year	Calc D_I (m)	% Error	Calc. D_{MAX} (m)	% Error	Calc. Basic Length (m)	% Error	Calc. Max. Length (m)	% Error	Calc. Mass (kg)	% Error	Application
R-R/SNECMA	Olympus	593-610	1	2	167.8					15.5	3.810	7.112	1.100	1.206	186.0	0.00	1420	3075.0	1974	1.189	8.1%	1.407	16.7%	4.116	8.0%					Concorde
R-R/SNECMA	Olympus	593-621	1	2	177.6					15.5	3.810	7.112	1.100	1.206	186.0	0.00		3386.0		1.189	8.1%	1.407	16.7%	4.116	8.0%					
SNECMA	Atar	9C	1	1	60.8	43.2	57.03	28.33	X	5.8	3.195	5.944		1.120	68.0	0.00	1163	1450.0		0.719		0.950	-15.2%	3.045	-4.7%	5.807	-2.3%	1457.7	0.5%	Mirage IIIE/R
SNECMA	Atar	9K50	1	1	70.5	49.2	55.72	27.47	X	6.2	3.195	5.944	0.786	1.028	72.0	0.00	1203	1590.0	1969	0.740	-5.9%	0.974	-5.2%	3.110	-2.7%	5.944	0.0%	1551.2	-2.4%	
SNECMA	M53	5	1	1	88.3	55.0		24.64	X	9.8		4.853		1.055	86.2	0.35		1450.0	1979	0.810		1.056	0.0%	3.044		5.620	15.8%	1453.2	0.2%	Mirage 2000
SNECMA	M53	P2	1	1	95.2	64.5	58.89	25.55	X	9.8	2.585	5.070	0.796	1.055	94.0	0.36	1806	1500.0	1984	0.845	6.2%	1.097	3.9%	3.182	23.1%	5.906	16.5%	1631.0	8.7%	Mirage-2000
SNECMA	M88	2	1	2	75.0	50.0	50.00	22.20	3.98	24.5	2.035	3.538	0.669	0.900	65.0	0.30	1850	897.0	1989	0.703	5.1%	0.887	-1.5%	2.121	4.2%	3.743	5.8%	930.2	3.7%	Rafale
Turbo-Union	RB199	103	1	3	71.2	40.5		18.38		23.5	1.910	3.251	0.719	0.870	74.6	1.08	1650	965.0	1983	0.753	4.8%	0.916	5.3%	1.910	0.0%	3.385	4.1%	953.7	-1.2%	Tornado IDS
Turbo-Union	RB199	104	1	3	73.0	40.5		18.38		23.5	1.910	3.607	0.719		74.6	1.08	1650	976.0	1985	0.753	4.8%	0.916		1.910	0.0%	3.387	-6.1%	953.7	-2.3%	Tornado ADV
Turbo-Union	RB199	105	1	3	75.2	42.2	64.40	18.50	2.85	24.4	1.930	3.302	0.752	0.914	74.2	1.10	1670	975.7		0.751	-0.1%	0.914	0.0%	1.889	-2.1%	3.335	1.0%	935.7	-4.1%	Tornado ECR
Turbo-Union	RB199	105 Dry	0	3	42.2	42.2	18.50	18.50	2.85	24.4	1.930	1.930	0.752	0.914	74.2	1.10	1670	720.0		0.751	-0.1%	0.914	0.0%	1.889	-2.1%	1.889	-2.1%	754.3	4.8%	Tornado (Dry)
T'meca-SNECMA	Larzac	04-C6	0	2	13.2	13.2	X	20.10		10.6	1.187	X	0.452	0.602	28.0	1.13	1405	290.0	1972	0.461	2.2%	0.613	1.8%	1.384	16.6%	1.384	16.6%	291.9	0.7%	
T'meca-SNECMA	Larzac	04-C20	0	2	14.1	14.1	X	21.11		11.1	1.187	X	0.452	0.602	28.6	1.04	1433	302.0	1982	0.466	3.3%	0.619	2.8%	1.406	18.4%	1.406	18.4%	301.3	-0.2%	Alpha-Jet
Volvo Flygmotor	RM8	A	1	2	115.6	65.6	69.96	17.60		16.5	3.121	5.873	1.030	1.349	146.0	1.00		2100.0	1968	1.054	2.3%	1.267	-6.0%	3.145	0.8%	5.873	0.0%	2293.6	9.2%	AJ-37, SF-37
Volvo Flygmotor	RM8	B	1	2	125.0	72.1	71.38	18.13		18.5	3.214	5.979	1.030	1.349	146.0	1.00	1470	2220.0	1974	1.054	2.3%	1.267	-6.0%	3.091	-3.8%	5.727	-4.2%	2219.7	0.0%	JA-37 Viggen
Volvo Flygmotor	RM12		1	2	80.5	54.0	50.60	23.90	4.20	27.5	2.132	4.040	0.709	0.884	68.0	0.31	1733	1050.5	1988	0.719	1.4%	0.905	2.3%	2.132	0.0%	3.747	-7.3%	953.0	-9.3%	JAS-39 Gripen
Williams Research	F107	WR-101	0	2	2.83	2.83	X	19.40	2.10	13.8	1.232	X		0.305	6.169	1.00	1373	66.2	1979	0.217		0.315	3.3%	0.609		0.609		66.2	0.0%	ALCM
Williams-Rolls	FJ44	82-004	0	2	8.01	8.01	X	12.68		12.8	1.024	X	0.479	0.602	28.3	3.24	1252	203.0	1985	0.464	-3.2%	0.617	2.5%	1.024	0.0%	1.024	0.0%	217.4	7.1%	Squalus - others
No: of Data Points																				33		52		39		47		53		
Average |%| Error																				4.038		5.905		5.721		6.671		6.358		
< 5% Error																				72.7%		50.0%		59.0%		51.1%		50.9%		
< 10% Error																				97.0%		78.8%		82.1%		72.3%		83.0%		
< 15% Error																				97.0%		90.4%		89.7%		85.1%		86.8%		
< 20% Error																				97.0%		100.0%		97.4%		100.0%		92.5%		
< 25% Error																				100.0%		100.0%		100.0%		100.0%		98.1%		
R^2 Value																				0.9829		0.9565		0.9344		0.9278		0.9741		

Appendix C - COMBAT Output Data

C.1 Example Aircraft Configurations

F-16C Example Configuration Data

No Optimisation of Input Variables

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER
NUMBER OF CREW	1
MAX HI-ALT MACH NO:	2.00
MAX LO-ALT MACH NO:	1.20
DESIGN MAX G LOAD	9.00
ULTIMATE LOAD FACTOR	13.65
MASS OF EXT WEAPONS	175.0 kg
MASS OF INT WEAPONS	0.0 kg
VOL. OF INT WEAPONS	0.00 m^3
VOL. OF GUN & AMMO.	0.25 m^3
% ADVANCED MATERIALS	1.72

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1715.4
TOTAL FUSELAGE GROUP	1604.3
TOTAL EMPENNAGE GROUP	203.7
TOTAL UNDERCARRIAGE	695.3
ENGINE & INSTALLATION	1477.1
SYSTEMS	
FUEL SYSTEM & TANKS	394.5
POWER & FLYING CONTROLS	278.1
HYDRAULICS & PNEUMATICS	214.2
ELECTRICS & BATTERIES	224.4
AIR SYSTEM (AIRCON/ICE)	183.5
COCKPIT INSTRUMENTS ETC	285.9
TOTAL AVIONICS SYSTEM	569.4
FIXED ARMAMENT ITEMS	800.0
AUXILIARY POWER UNIT	0.0
AIRCRAFT EMPTY MASS	8645.7

WING PARAMETERS

WING TYPE	CONV. & LEX
GROSS AREA	27.870 m^2
TOTAL SPAN	9.144 m
MEAN AERO CHORD	3.438 m
LDG EDGE SWEEP	40.000 Deg
1/4 CHORD SWEEP	32.318 Deg
ASPECT RATIO	3.000
TAPER RATIO	0.235
THCK/CHRD RATIO	0.0400

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.
AIR MASS FLOW RATE	103.400 kg/s
MAXIMUM THRUST	105.502 kN
MILITARY THRUST	66.513 kN
INLET DIAMETER	0.887 m
MAXIMUM DIAMETER	1.088 m
BARE ENGINE LENGTH	2.600 m
OVERALL LENGTH	4.691 m
FAN BYPASS RATIO	0.600
COMP. PRESS. RATIO	24.799

FUSELAGE PARAMETERS

OVERALL LENGTH	15.030 m
EFFECTIVE DIA.	1.900 m
MAXIMUM HEIGHT	1.734 m
MAXIMUM WIDTH	2.081 m
ENG INLET AREA	0.457 m^2
BASE DRAG AREA	0.644 m^2
CENTRELINE HT.	2.478 m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.182
LANDING CL	1.786
HIGH SPEED CL	1.560
BUFFET MAX CL	0.237
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	AFT-TAIL
NET TAIL AREA	5.933 m^2
TAIL MEAN AERO CHRD	1.572 m
TAIL ASPECT RATIO	2.400
TAIL THK/CHRD RATIO	0.0360
NUMBER OF FINS	1
TOTAL FIN AREA	4.559 m^2
FIN MEAN AERO CHRD	2.466 m
FIN ASPECT RATIO	0.750
FIN THCK/CHRD RATIO	0.0360

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							12098.5	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	12006.1	0.9924
2 TAKE-OFF	15.0	0.27	0.05	1.20	1.0	1.3	11914.0	0.9923
3 CLIMB / ACCEL.	9144.0	0.80	0.00	0.00	0.0	7.3	11443.4	0.9605
4 SUBSON. CRUISE	9144.0	0.80	255.00	0.00	0.0	24.8	11083.1	0.9685
5 DESCEND/DECEL.	50.0	0.92	0.00	0.00	1.0	25.4	11083.1	1.0000
6 SUBSON. CRUISE	50.0	0.92	60.00	0.00	1.0	28.6	10833.3	0.9775
7 COMBAT TURNS	50.0	0.92	180.00	0.00	1.0	28.8	10740.4	0.9914
8 SUBSON. CRUISE	50.0	0.92	60.00	0.00	1.0	32.0	10491.0	0.9768
9 CLIMB / ACCEL.	9144.0	0.80	0.00	0.00	1.0	33.8	9706.9	0.9253
10 SUBSON. CRUISE	9144.0	0.80	255.00	0.00	0.0	51.3	9373.8	0.9657
11 DESCEND/DECEL.	610.0	0.40	0.00	0.00	0.0	51.9	9373.8	1.0000
12 LOITER/C.A.P.	610.0	0.40	10.00	0.00	0.0	61.9	9149.7	0.9761
13 LANDING	0.0	0.18	0.00	0.00	0.0	62.1	9104.0	0.9950
14 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	63.1	9011.6	0.9899

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.98	1.00	1.0	1000.0	434.91
2 LANDING DIST.	0.0	0.00	0.4	0.15	0.36	0.0	1000.0	648.68
3 ATTAINED TURN	9140.0	1.20	1.0	0.50	0.05	0.0	9.0	8.82
4 ATTAINED TURN	9140.0	0.80	1.0	0.50	0.05	0.0	5.0	5.68
5 SUSTAINED TURN	6900.0	0.90	1.0	0.50	0.05	1.0	5.0	4.40
6 SUSTAINED TURN	12800.0	1.60	1.0	0.50	0.05	1.0	3.0	1.98
7 MACH XS THRUST	10975.0	2.00	0.0	0.50	0.05	1.0	0.0	-17 93
8 MACH XS THRUST	0.0	1.20	0.0	0.50	0.05	1.0	0.0	-4.84

AIRCRAFT EMPTY MASS	8645.7 kg
CREW INCLUDING EQPT	100.0 kg
TOTAL WEAPONS MASS	175.0 kg
INTERNAL FUEL MASS	3177.8 kg
EXTERNAL FUEL MASS	0.0 kg
AIRCRAFT ALL-UP MASS	12098.5 kg
AIRCRAFT GROSS MASS	17923.5 kg

WING LOADING	434.11 kg/m^2
THRUST/WEIGHT	0.600
FUEL FRACTION	0.177
XS FUEL VOL.	-0.000 m^3
THRUST RATIO	1.910

F-16C Example Cost Data

No Optimisation of Input Variables

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	11853.5	lb.	No. DEVELOPMENT AIRCRAFT		8	No. PRODUCTION AIRCRAFT		3763		
MAX. TRUE AIRSPEED	1143.7	kt.	No. STATIC AIRFRAMES		2	AIFRAME ENGINEERING	M$	1968.3	AIR SERVICES	0.076
UNINST. AVION. MASS	1129.7	lb.	No. DEVELOPMENT ENGINES		12.	PRODUCTION TOOLING	M$	1063.0	FLYING CONTROLS	0.539
NUMBER OF PARTNERS	1.000		AIRFRAME ENGINEERING COST	M$	850.9	MANUFACTURING LABOUR	M$	16143.3	FUEL SYSTEM	0.228
COLLABORATION FACTOR	1.000		DEVELOPMENT SUPPORT COST	M$	497.0	QUALITY CNTRL LBR COST	M$	2446.5	HYD. & PNEU.	0.439
ADV. MATERIAL FACTOR	1.006		FLIGHT TESTING COST	M$	233.6	MANUFACTURING MATERIAL	M$	7583.9	UNDERCARRIAGE	0.696
ADV. TECH. FACTOR	1.003		DEVELOPMENT TOOLING COST	M$	524.1	PRODUCTION ENGINE COST	M$	19435.1	OXYGEN SYSTEM	0.057
ADV. DESIGN TOOLS	1.100		MANUFACTURING LABOUR	M$	551.5	AVIONICS PURCHASE COST	M$	29840.2	MISC. MECH	0.020
ADV. TECH TESTING	1.100		QUALITY CONTROL LABOUR	M$	83.6				TOTAL STRUCTURE	1.120
LOW OBSERVABLES	1.000		MANUFACTURING MATERIAL	M$	54.5	TOTAL PRODUCTION COST	M$	78480.4	POWERPLANT SYST.	0.533
SECURITY FACTOR	1.100		ENGINE DEVELOPMENT COST	M$	2150.2				ARMAMENT SYSTEM	0.920
FIRST FLIGHT YEAR	1977.		DEVELOPMENT ENGINE COST	M$	98.9	G.S.E. & INITIAL SPARES	M$	7848.0	AVIONICS SYSTEM	0.437
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$	1714.1				NAV. & COMMS	0.278
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$	47.6	ENGINEERING LABOUR RATE		108.6 $/hr	ELEC. & INSTR.	0.350
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$	1741.3	TOOLING/QUALITY LABOUR RT		79.6 $/hr	SCHEDULED MAINT.	4.702
			TOTAL DEVELOPMENT COST	M$	8547.1	MANUFACTURING LABOUR RATE		68.3 $/hr	TOTAL MAINTENANCE	10.392

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	150.	
AVG. YRLY FLIGHT TIME	320.0	
No. OPS. PERSONNEL	2961.6	
No. SUPP. PERSONNEL	1519.7	
OPS. PERSONNEL COST	M$	122.751
SUPP. PERSONNEL COST	M$	62.207
SUPPORT & TRAINING	M$	23.418
UNIT LVL CONSUMPTION	M$	55.922
ENGINE CONTRACT COST	M$	40.567
MECH. CONTRACT COST	M$	67.815
AVIONICS CNTRCT COST	M$	48.182
SUPPLY CONTRACT COST	M$	5.393
SUSTAINING SUPPORT	M$	21.546
INSTALLATION SUPPORT	M$	17.874
1st YR BASE O&S COST	M$	465.675
TOTAL 20.YR O&S COST	M$	5341.249

DISPOSAL		
DISPOSAL LABOUR COST	M$	807.2
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-390.3
TOTAL DISPOSAL COST	M$	416.8
DEFLATED DISPOSAL COST	M$	130.0

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	2.267
PROD.	M$	20.812
GSE&IS	M$	2.086
O & S	M$	35.608
DISP.	M$	0.034
TOTAL	M$	60.806

Appendix C - COMBAT Output Data

C.1 Example Aircraft Configurations

F-16C Example Configuration Data

No Optimisation of Input Variables

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.00	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.65	
MASS OF EXT WEAPONS	175.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.25	m^3
% ADVANCED MATERIALS	1.72	

WING PARAMETERS

WING TYPE	CONV. & LEX	
GROSS AREA	27.870	m^2
TOTAL SPAN	9.144	m
MEAN AERO CHORD	3.438	m
LDG EDGE SWEEP	40.000	Deg
1/4 CHORD SWEEP	32.318	Deg
ASPECT RATIO	3.000	
TAPER RATIO	0.235	
THCK/CHRD RATIO	0.0400	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.030	m
EFFECTIVE DIA.	1.900	m
MAXIMUM HEIGHT	1.734	m
MAXIMUM WIDTH	2.081	m
ENG INLET AREA	0.457	m^2
BASE DRAG AREA	0.644	m^2
CENTRELINE HT.	2.478	m

EMPENNAGE PARAMETERS

TAIL PLACEMENT	AFT-TAIL	
NET TAIL AREA	5.933	m^2
TAIL MEAN AERO CHRD	1.572	m
TAIL ASPECT RATIO	2.400	
TAIL THK/CHRD RATIO	0.0360	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.559	m^2
FIN MEAN AERO CHRD	2.466	m
FIN ASPECT RATIO	0.750	
FIN THCK/CHRD RATIO	0.0360	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1715.4
TOTAL FUSELAGE GROUP	1604.3
TOTAL EMPENNAGE GROUP	203.7
TOTAL UNDERCARRIAGE	695.3
ENGINE & INSTALLATION	1477.1
SYSTEMS	
FUEL SYSTEM & TANKS	394.5
POWER & FLYING CONTROLS	278.1
HYDRAULICS & PNEUMATICS	214.2
ELECTRICS & BATTERIES	224.4
AIR SYSTEM (AIRCON/ICE)	183.5
COCKPIT INSTRUMENTS ETC	285.9
TOTAL AVIONICS SYSTEM	569.4
FIXED ARMAMENT ITEMS	800.0
AUXILIARY POWER UNIT	0.0
AIRCRAFT EMPTY MASS	8645.7

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	103.400	kg/s
MAXIMUM THRUST	105.502	kN
MILITARY THRUST	66.513	kN
INLET DIAMETER	0.887	m
MAXIMUM DIAMETER	1.088	m
BARE ENGINE LENGTH	2.600	m
OVERALL LENGTH	4.691	m
FAN BYPASS RATIO	0.600	
COMP. PRESS. RATIO	24.799	

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.182
LANDING CL	1.786
HIGH SPEED CL	1.560
BUFFET MAX CL	0.237
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							12098.5	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	12006.1	0.9924
2 TAKE-OFF	15.0	0.27	0.05	1.20	1.0	1.3	11914.0	0.9923
3 CLIMB / ACCEL.	9144.0	0.80	0.00	0.00	0.0	7.3	11443.4	0.9605
4 SUBSON. CRUISE	9144.0	0.80	255.00	0.00	0.0	24.8	11083.1	0.9685
5 DESCEND/DECEL.	50.0	0.92	0.00	0.00	1.0	25.4	11083.1	1.0000
6 SUBSON. CRUISE	50.0	0.92	60.00	0.00	1.0	28.6	10833.3	0.9775
7 COMBAT TURNS	50.0	0.92	180.00	0.00	1.0	28.8	10740.4	0.9914
8 SUBSON. CRUISE	50.0	0.92	60.00	0.00	1.0	32.0	10491.0	0.9768
9 CLIMB / ACCEL.	9144.0	0.80	0.00	0.00	1.0	33.8	9706.9	0.9253
10 SUBSON. CRUISE	9144.0	0.80	255.00	0.00	0.0	51.3	9373.8	0.9657
11 DESCEND/DECEL.	610.0	0.40	0.00	0.00	0.0	51.9	9373.8	1.0000
12 LOITER/C.A.P.	610.0	0.40	10.00	0.00	0.0	61.9	9149.7	0.9761
13 LANDING	0.0	0.18	0.00	0.00	0.0	62.1	9104.0	0.9950
14 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	63.1	9011.6	0.9899

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.98	1.00	1.0	1000.0	434.91
2 LANDING DIST.	0.0	0.00	0.4	0.15	0.36	0.0	1000.0	648.68
3 ATTAINED TURN	9140.0	1.20	1.0	0.50	0.05	0.0	9.0	8.82
4 ATTAINED TURN	9140.0	0.80	1.0	0.50	0.05	0.0	5.0	5.68
5 SUSTAINED TURN	6900.0	0.90	1.0	0.50	0.05	1.0	5.0	4.40
6 SUSTAINED TURN	12800.0	1.60	1.0	0.50	0.05	1.0	3.0	1.98
7 MACH XS THRUST	10975.0	2.00	0.0	0.50	0.05	1.0	0.0	-17 93
8 MACH XS THRUST	0.0	1.20	0.0	0.50	0.05	1.0	0.0	-4.84

AIRCRAFT EMPTY MASS	8645.7	kg	WING LOADING 434.11 kg/m^2
CREW INCLUDING EQPT	100.0	kg	THRUST/WEIGHT 0.600
TOTAL WEAPONS MASS	175.0	kg	FUEL FRACTION 0.177
INTERNAL FUEL MASS	3177.8	kg	XS FUEL VOL. -0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.910
AIRCRAFT ALL-UP MASS	12098.5	kg	
AIRCRAFT GROSS MASS	17923.5	kg	

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER
NUMBER OF CREW	1
MAX HI-ALT MACH NO:	2.00
MAX LO-ALT MACH NO:	1.20
DESIGN MAX G LOAD	9.00
ULTIMATE LOAD FACTOR	13.50
MASS OF EXT WEAPONS	475.0 kg
MASS OF INT WEAPONS	0.0 kg
VOL. OF INT WEAPONS	0.00 m^3
VOL. OF GUN & AMMO.	0.31 m^3
% ADVANCED MATERIALS	33.68

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1562.2
TOTAL FUSELAGE GROUP	1685.4
TOTAL EMPENNAGE GROUP	177.0
TOTAL UNDERCARRIAGE	819.1
ENGINE & INSTALLATION SYSTEMS	2052.1
FUEL SYSTEM & TANKS	483.6
POWER & FLYING CONTROLS	334.7
HYDRAULICS & PNEUMATICS	233.9
ELECTRICS & BATTERIES	260.2
AIR SYSTEM (AIRCON/ICE)	214.2
COCKPIT INSTRUMENTS ETC	340.8
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	205.2
AIRCRAFT EMPTY MASS	10073.4

WING PARAMETERS

WING TYPE	DELTA WING
GROSS AREA	50.000 m^2
TOTAL SPAN	10.500 m
MEAN AERO CHORD	5.629 m
LDG EDGE SWEEP	53.000 Deg
1/4 CHORD SWEEP	44.765 Deg
ASPECT RATIO	2.205
TAPER RATIO	0.150
THCK/CHRD RATIO	0.0500

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.
AIR MASS FLOW RATE	73.820 kg/s
MAXIMUM THRUST	179.831 kN
MILITARY THRUST	120.280 kN
INLET DIAMETER	0.749 m
MAXIMUM DIAMETER	0.937 m
BARE ENGINE LENGTH	2.215 m
OVERALL LENGTH	3.910 m
FAN BYPASS RATIO	0.408
COMP. PRESS. RATIO	26.001

FUSELAGE PARAMETERS

OVERALL LENGTH	15.080 m
EFFECTIVE DIA.	2.080 m
MAXIMUM HEIGHT	1.125 m
MAXIMUM WIDTH	3.262 m
ENG INLET AREA	0.659 m^2
BASE DRAG AREA	1.023 m^2
CENTRELINE HT.	2.179 m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.991
LANDING CL	1.310
HIGH SPEED CL	1.484
BUFFET MAX CL	0.524
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD
NET TAIL AREA	2.374 m^2
TAIL MEAN AERO CHRD	1.160 m
TAIL ASPECT RATIO	1.764
TAIL THK/CHRD RATIO	0.0450
NUMBER OF FINS	1
TOTAL FIN AREA	4.867 m^2
FIN MEAN AERO CHRD	2.573 m
FIN ASPECT RATIO	0.735
FIN THCK/CHRD RATIO	0.0450

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							15351.1	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15185.4	0.9892
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15088.7	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	14655.1	0.9713
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	13856.0	0.9455
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	13856.0	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	13507.5	0.9749
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	13292.3	0.9841
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	13162.8	0.9903
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	13001.1	0.9877
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	12188.7	0.9375
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	12029.6	0.9869
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.5	11825.9	0.9831
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.4	11698.5	0.9892
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.4	11698.5	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.5	10972.5	0.9379
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.9	10972.5	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	85.1	10917.7	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.1	10751.9	0.9848

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	240.67
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	502.99
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.70
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.12
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.08
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	2.81
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	24.08
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	32.12
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.85

AIRCRAFT EMPTY MASS	10073.4 kg	WING LOADING	307.02 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.858
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.220
INTERNAL FUEL MASS	4702.7 kg	XS FUEL VOL.	0.008 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	0.965
AIRCRAFT ALL-UP MASS	15351.1 kg		
AIRCRAFT GROSS MASS	21376.1 kg		

Eurofighter 2000 Example Cost Data

No Optimisation of Input Variables

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12225.7	lb.
MAX. TRUE AIRSPEED	1143.7	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.271	
ADV. TECH. FACTOR	1.277	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT		9
No. STATIC AIRFRAMES		2
No. DEVELOPMENT ENGINES		21.
AIRFRAME ENGINEERING COST	M$	1828.6
DEVELOPMENT SUPPORT COST	M$	865.2
FLIGHT TESTING COST	M$	464.0
DEVELOPMENT TOOLING COST	M$	971.1
MANUFACTURING LABOUR	M$	1101.3
QUALITY CONTROL LABOUR	M$	166.9
MANUFACTURING MATERIAL	M$	92.7
ENGINE DEVELOPMENT COST	M$	3273.4
DEVELOPMENT ENGINE COST	M$	157.3
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	14978.5

PRODUCTION		
No. PRODUCTION AIRCRAFT		620
AIFRAME ENGINEERING	M$	3014.3
PRODUCTION TOOLING	M$	1114.6
MANUFACTURING LABOUR	M$	10478.4
QUALITY CNTRL LBR COST	M$	1588.0
MANUFACTURING MATERIAL	M$	2713.9
PRODUCTION ENGINE COST	M$	6625.9
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	32486.4
G.S.E. & INITIAL SPARES	M$	3248.6
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.046
FLYING CONTROLS	0.561
FUEL SYSTEM	0.143
HYD. & PNEU.	0.267
UNDERCARRIAGE	0.707
OXYGEN SYSTEM	0.067
MISC. MECH	0.004
TOTAL STRUCTURE	0.612
POWERPLANT SYST.	0.486
ARMAMENT SYSTEM	0.973
AVIONICS SYSTEM	0.174
NAV. & COMMS	0.225
ELEC. & INSTR.	0.312
SCHEDULED MAINT.	4.768
TOTAL MAINTENANCE	9.345

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	673.0	
No. SUPP. PERSONNEL	395.6	
OPS. PERSONNEL COST	M$	28.109
SUPP. PERSONNEL COST	M$	16.321
SUPPORT & TRAINING	M$	5.396
UNIT LVL CONSUMPTION	M$	13.850
ENGINE CONTRACT COST	M$	26.561
MECH. CONTRACT COST	M$	16.846
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.925
SUSTAINING SUPPORT	M$	12.689
INSTALLATION SUPPORT	M$	4.262
1st YR BASE O&S COST	M$	142.564
TOTAL 25.YR O&S COST	M$	1822.451

DISPOSAL

DISPOSAL LABOUR COST	M$	523.9
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-45.3
TOTAL DISPOSAL COST	M$	478.7
DEFLATED DISPOSAL COST	M$	111.5

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	23.813
PROD.	M$	51.648
GSE&IS	M$	5.240
O & S	M$	46.730
DISP.	M$	0.177
TOTAL	M$	127.607

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.65	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	39.01	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	4344.5
TOTAL FUSELAGE GROUP	1862.3
TOTAL EMPENNAGE GROUP	366.6
TOTAL UNDERCARRIAGE	1125.7
ENGINE & INSTALLATION	4597.3
SYSTEMS	
FUEL SYSTEM & TANKS	678.5
POWER & FLYING CONTROLS	439.9
HYDRAULICS & PNEUMATICS	277.5
ELECTRICS & BATTERIES	346.7
AIR SYSTEM (AIRCON/ICE)	289.7
COCKPIT INSTRUMENTS ETC	383.7
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	459.7
AIRCRAFT EMPTY MASS	16877.1

WING PARAMETERS

WING TYPE	CONV. & LEX	
GROSS AREA	77.060	m^2
TOTAL SPAN	15.598	m
MEAN AERO CHORD	5.533	m
LDG EDGE SWEEP	44.701	Deg
1/4 CHORD SWEEP	38.646	Deg
ASPECT RATIO	3.157	
TAPER RATIO	0.250	
THCK/CHRD RATIO	0.0397	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	227.429	kg/s
MAXIMUM THRUST	277.017	kN
MILITARY THRUST	185.282	kN
INLET DIAMETER	1.314	m
MAXIMUM DIAMETER	1.538	m
BARE ENGINE LENGTH	4.009	m
OVERALL LENGTH	7.855	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.161	m
EFFECTIVE DIA.	2.305	m
MAXIMUM HEIGHT	2.104	m
MAXIMUM WIDTH	2.525	m
ENG INLET AREA	1.015	m^2
BASE DRAG AREA	1.607	m^2
CENTRELINE HT.	2.784	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.130
LANDING CL	1.717
HIGH SPEED CL	1.514
BUFFET MAX CL	0.219
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	AFT-TAIL	
NET TAIL AREA	12.885	m^2
TAIL MEAN AERO CHRD	2.259	m
TAIL ASPECT RATIO	2.526	
TAIL THK/CHRD RATIO	0.0357	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.486	m^2
FIN MEAN AERO CHRD	2.636	m
FIN ASPECT RATIO	0.789	
FIN THCK/CHRD RATIO	0.0357	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							24061.9	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	23806.6	0.9894
2	TAKE-OFF	15.0	0.23	0.05	1.20	1.0	1.2	23662.1	0.9939
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.7	22987.8	0.9715
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.8	21881.6	0.9519
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.8	21881.6	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.8	21400.0	0.9780
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	21100.4	0.9860
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	20954.9	0.9931
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	20765.2	0.9909
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	19513.8	0.9397
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	19354.7	0.9918
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	19062.4	0.9849
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	18919.2	0.9925
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	18919.2	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	17872.8	0.9447
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	17872.8	1.0000
17	LANDING	0.0	0.16	0.00	0.00	0.0	85.0	17783.5	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.0	17528.1	0.9856

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	220.55
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	391.02
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	7.0	6.99
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	7.49
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	7.27
6	SUSTAINED TURN	12925.0	1.60	1.0	0.50	1.00	1.0	3.0	4.24
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	85.08
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	112.45
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.51

AIRCRAFT EMPTY MASS	16877.1 kg	WING LOADING	312.25 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.939
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.220
INTERNAL FUEL MASS	6609.8 kg	XS FUEL VOL.	-0.001 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.285
AIRCRAFT ALL-UP MASS	24061.9 kg		
AIRCRAFT GROSS MASS	30086.9 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	20201.2	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.323	
ADV. TECH. FACTOR	1.288	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	3316.0
DEVELOPMENT SUPPORT COST	M$	1650.2
FLIGHT TESTING COST	M$	963.9
DEVELOPMENT TOOLING COST	M$	1620.2
MANUFACTURING LABOUR	M$	1770.1
QUALITY CONTROL LABOUR	M$	268.2
MANUFACTURING MATERIAL	M$	145.2
ENGINE DEVELOPMENT COST	M$	3622.5
DEVELOPMENT ENGINE COST	M$	186.2
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	19600.8

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	5466.2
PRODUCTION TOOLING	M$	1859.7
MANUFACTURING LABOUR	M$	16841.7
QUALITY CNTRL LBR COST	M$	2552.3
MANUFACTURING MATERIAL	M$	4251.9
PRODUCTION ENGINE COST	M$	6009.2
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	43932.2
G.S.E. & INITIAL SPARES	M$	4393.2
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.059
FLYING CONTROLS	0.653
FUEL SYSTEM	0.199
HYD. & PNEU.	0.406
UNDERCARRIAGE	0.835
OXYGEN SYSTEM	0.083
MISC. MECH	0.005
TOTAL STRUCTURE	0.832
POWERPLANT SYST.	0.697
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.360
NAV. & COMMS	0.322
ELEC. & INSTR.	0.454
SCHEDULED MAINT.	6.089
TOTAL MAINTENANCE	12.053

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	807.5	
No. SUPP. PERSONNEL	418.6	
OPS. PERSONNEL COST	M$	33.454
SUPP. PERSONNEL COST	M$	17.211
SUPPORT & TRAINING	M$	6.403
UNIT LVL CONSUMPTION	M$	15.439
ENGINE CONTRACT COST	M$	19.328
MECH. CONTRACT COST	M$	23.929
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.965
SUSTAINING SUPPORT	M$	16.717
INSTALLATION SUPPORT	M$	4.890
1st YR BASE O&S COST	M$	155.943
TOTAL 25.YR O&S COST	M$	1993.470

DISPOSAL		
DISPOSAL LABOUR COST	M$	842.1
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-68.9
TOTAL DISPOSAL COST	M$	773.3
DEFLATED DISPOSAL COST	M$	180.2

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	31.162
PROD.	M$	69.844
GSE&IS	M$	7.086
O & S	M$	51.115
DISP.	M$	0.286
TOTAL	M$	159.493

Aircraft Configuration Data

Single Engine, Delta Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	30.80	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1826.1
TOTAL FUSELAGE GROUP	1344.5
TOTAL EMPENNAGE GROUP	118.0
TOTAL UNDERCARRIAGE	839.6
ENGINE & INSTALLATION	2693.1
SYSTEMS	
FUEL SYSTEM & TANKS	494.1
POWER & FLYING CONTROLS	341.9
HYDRAULICS & PNEUMATICS	237.1
ELECTRICS & BATTERIES	266.1
AIR SYSTEM (AIRCON/ICE)	219.2
COCKPIT INSTRUMENTS ETC	324.6
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	269.3
AIRCRAFT EMPTY MASS	10678.6

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	56.820	m^2
TOTAL SPAN	11.159	m
MEAN AERO CHORD	6.228	m
LDG EDGE SWEEP	51.511	Deg
1/4 CHORD SWEEP	41.487	Deg
ASPECT RATIO	2.192	
TAPER RATIO	0.100	
THCK/CHRD RATIO	0.0477	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	156.055	kg/s
MAXIMUM THRUST	190.080	kN
MILITARY THRUST	127.135	kN
INLET DIAMETER	1.089	m
MAXIMUM DIAMETER	1.303	m
BARE ENGINE LENGTH	3.287	m
OVERALL LENGTH	6.210	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.585	m
EFFECTIVE DIA.	1.922	m
MAXIMUM HEIGHT	1.755	m
MAXIMUM WIDTH	2.106	m
ENG INLET AREA	0.696	m^2
BASE DRAG AREA	1.103	m^2
CENTRELINE HT.	2.548	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.945
LANDING CL	1.249
HIGH SPEED CL	1.347
BUFFET MAX CL	0.500
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	NONE	
NET TAIL AREA	0.000	m^2
TAIL MEAN AERO CHRD	0.000	m
TAIL ASPECT RATIO	1.753	
TAIL THK/CHRD RATIO	0.0429	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.714	m^2
FIN MEAN AERO CHRD	2.540	m
FIN ASPECT RATIO	0.731	
FIN THCK/CHRD RATIO	0.0429	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15927.8	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15752.6	0.9890
2	TAKE-OFF	15.0	0.24	0.05	1.20	1.0	1.2	15653.6	0.9937
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	15215.8	0.9720
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14418.6	0.9476
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14418.6	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	14068.3	0.9757
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13867.5	0.9857
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13759.4	0.9922
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13597.6	0.9882
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12738.9	0.9369
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12579.8	0.9875
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12386.3	0.9846
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	12280.0	0.9914
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	12280.0	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11550.0	0.9406
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11550.0	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.8	11492.3	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	11317.1	0.9848

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	225.09
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	481.95
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.13
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	7.50
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.06
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.33
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	51.43
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	68.48
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10678.6 kg	WING LOADING	280.32 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.883
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.213
INTERNAL FUEL MASS	4674.2 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.176
AIRCRAFT ALL-UP MASS	15927.8 kg		
AIRCRAFT GROSS MASS	21952.8 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	11984.0	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.242	
ADV. TECH. FACTOR	1.272	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	2028.6
DEVELOPMENT SUPPORT COST	M$	1016.7
FLIGHT TESTING COST	M$	517.8
DEVELOPMENT TOOLING COST	M$	1018.7
MANUFACTURING LABOUR	M$	1117.6
QUALITY CONTROL LABOUR	M$	169.4
MANUFACTURING MATERIAL	M$	95.0
ENGINE DEVELOPMENT COST	M$	3574.5
DEVELOPMENT ENGINE COST	M$	155.4
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15751.8

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3343.9
PRODUCTION TOOLING	M$	1169.3
MANUFACTURING LABOUR	M$	10634.0
QUALITY CNTRL LBR COST	M$	1611.5
MANUFACTURING MATERIAL	M$	2781.7
PRODUCTION ENGINE COST	M$	5013.3
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	31505.0
G.S.E. & INITIAL SPARES	M$	3150.5
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.050
FLYING CONTROLS	0.570
FUEL SYSTEM	0.162
HYD. & PNEU.	0.374
UNDERCARRIAGE	0.718
OXYGEN SYSTEM	0.076
MISC. MECH	0.005
TOTAL STRUCTURE	0.666
POWERPLANT SYST.	0.534
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.352
NAV. & COMMS	0.237
ELEC. & INSTR.	0.352
SCHEDULED MAINT.	4.853
TOTAL MAINTENANCE	7

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	710.9	
No. SUPP. PERSONNEL	402.0	
OPS. PERSONNEL COST	M$	29.617
SUPP. PERSONNEL COST	M$	16.572
SUPPORT & TRAINING	M$	5.676
UNIT LVL CONSUMPTION	M$	14.333
ENGINE CONTRACT COST	M$	14.543
MECH. CONTRACT COST	M$	17.861
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.547
SUSTAINING SUPPORT	M$	12.136
INSTALLATION SUPPORT	M$	4.439
1st YR BASE O&S COST	M$	133.329
TOTAL 25.YR O&S COST	M$	1704.391

DISPOSAL

DISPOSAL LABOUR COST	M$	531.7
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-46.4
TOTAL DISPOSAL COST	M$	485.4
DEFLATED DISPOSAL COST	M$	113.1

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.043
PROD.	M$	50.087
GSE&IS	M$	5.081
O & S	M$	43.702
DISP.	M$	0.180
TOTAL	M$	124.094

Aircraft Configuration Data

Single Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	31.08	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1729.6
TOTAL FUSELAGE GROUP	1501.7
TOTAL EMPENNAGE GROUP	179.7
TOTAL UNDERCARRIAGE	856.6
ENGINE & INSTALLATION	2807.4
SYSTEMS	
FUEL SYSTEM & TANKS	506.0
POWER & FLYING CONTROLS	347.8
HYDRAULICS & PNEUMATICS	239.6
ELECTRICS & BATTERIES	270.9
AIR SYSTEM (AIRCON/ICE)	223.4
COCKPIT INSTRUMENTS ETC	325.1
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	280.7
AIRCRAFT EMPTY MASS	10973.7

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	51.679	m^2
TOTAL SPAN	10.656	m
MEAN AERO CHORD	5.733	m
LDG EDGE SWEEP	49.786	Deg
1/4 CHORD SWEEP	40.244	Deg
ASPECT RATIO	2.197	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0525	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	160.840	kg/s
MAXIMUM THRUST	195.908	kN
MILITARY THRUST	131.033	kN
INLET DIAMETER	1.105	m
MAXIMUM DIAMETER	1.321	m
BARE ENGINE LENGTH	3.340	m
OVERALL LENGTH	6.328	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.526	m
EFFECTIVE DIA.	1.951	m
MAXIMUM HEIGHT	1.781	m
MAXIMUM WIDTH	2.138	m
ENG INLET AREA	0.717	m^2
BASE DRAG AREA	1.137	m^2
CENTRELINE HT.	2.662	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.031
LANDING CL	1.363
HIGH SPEED CL	1.530
BUFFET MAX CL	0.545
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.375	m^2
TAIL MEAN AERO CHRD	1.162	m
TAIL ASPECT RATIO	1.758	
TAIL THK/CHRD RATIO	0.0473	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.962	m^2
FIN MEAN AERO CHRD	2.603	m
FIN ASPECT RATIO	0.732	
FIN THCK/CHRD RATIO	0.0473	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							16405.7	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	16225.1	0.9890
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	16121.7	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	15671.5	0.9721
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14835.0	0.9466
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14835.0	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	14466.4	0.9752
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	14259.2	0.9857
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	14145.7	0.9920
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13976.4	0.9880
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	13091.4	0.9367
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12932.3	0.9878
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12732.4	0.9845
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	12621.0	0.9913
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	12621.0	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11860.7	0.9398
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11860.7	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.8	11801.4	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	11620.8	0.9847

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	233.28
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.03
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.07
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.15
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	52.14
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	74.33
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10973.7 kg	WING LOADING	317.45 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.891
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.217
INTERNAL FUEL MASS	4857.0 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.154
AIRCRAFT ALL-UP MASS	16405.7 kg		
AIRCRAFT GROSS MASS	22430.7 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12311.3	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.245	
ADV. TECH. FACTOR	1.272	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	2078.1
DEVELOPMENT SUPPORT COST	M$	1042.2
FLIGHT TESTING COST	M$	534.7
DEVELOPMENT TOOLING COST	M$	1042.4
MANUFACTURING LABOUR	M$	1143.3
QUALITY CONTROL LABOUR	M$	173.3
MANUFACTURING MATERIAL	M$	97.0
ENGINE DEVELOPMENT COST	M$	3577.7
DEVELOPMENT ENGINE COST	M$	157.6
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15904.5

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3425.6
PRODUCTION TOOLING	M$	1196.4
MANUFACTURING LABOUR	M$	10878.4
QUALITY CNTRL LBR COST	M$	1648.6
MANUFACTURING MATERIAL	M$	2840.4
PRODUCTION ENGINE COST	M$	5086.6
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	32027.4
G.S.E. & INITIAL SPARES	M$	3202.7
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.051
FLYING CONTROLS	0.576
FUEL SYSTEM	0.169
HYD. & PNEU.	0.406
UNDERCARRIAGE	0.723
OXYGEN SYSTEM	0.073
MISC. MECH	0.005
TOTAL STRUCTURE	0.676
POWERPLANT SYST.	0.553
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.404
NAV. & COMMS	0.243
ELEC. & INSTR.	0.367
SCHEDULED MAINT.	4.911
TOTAL MAINTENANCE	10.215

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	722.2	
No. SUPP. PERSONNEL	404.0	
OPS. PERSONNEL COST	M$	30.064
SUPP. PERSONNEL COST	M$	16.647
SUPPORT & TRAINING	M$	5.760
UNIT LVL CONSUMPTION	M$	14.427
ENGINE CONTRACT COST	M$	14.864
MECH. CONTRACT COST	M$	18.317
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.577
SUSTAINING SUPPORT	M$	12.337
INSTALLATION SUPPORT	M$	4.492
1st YR BASE O&S COST	M$	135.090
TOTAL 25.YR O&S COST	M$	1726.907

DISPOSAL		
DISPOSAL LABOUR COST	M$	543.9
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-47.4
TOTAL DISPOSAL COST	M$	496.6
DEFLATED DISPOSAL COST	M$	115.7

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.285
PROD.	M$	50.918
GSE&IS	M$	5.166
O & S	M$	44.280
DISP.	M$	0.184
TOTAL	M$	125.833

Aircraft Configuration Data

Single Engine, Canard Configuration. 1 Crew, 2 Fins. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	31.72	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1806.6
TOTAL FUSELAGE GROUP	1569.7
TOTAL EMPENNAGE GROUP	352.5
TOTAL UNDERCARRIAGE	898.9
ENGINE & INSTALLATION SYSTEMS	3115.3
FUEL SYSTEM & TANKS	535.1
POWER & FLYING CONTROLS	362.6
HYDRAULICS & PNEUMATICS	245.9
ELECTRICS & BATTERIES	283.0
AIR SYSTEM (AIRCON/ICE)	233.9
COCKPIT INSTRUMENTS ETC	334.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	311.5
AIRCRAFT EMPTY MASS	11754.1

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	55.286	m^2
TOTAL SPAN	10.921	m
MEAN AERO CHORD	5.984	m
LDG EDGE SWEEP	49.706	Deg
1/4 CHORD SWEEP	39.923	Deg
ASPECT RATIO	2.157	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0507	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	173.345	kg/s
MAXIMUM THRUST	211.141	kN
MILITARY THRUST	141.221	kN
INLET DIAMETER	1.147	m
MAXIMUM DIAMETER	1.365	m
BARE ENGINE LENGTH	3.474	m
OVERALL LENGTH	6.630	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.396	m
EFFECTIVE DIA.	2.026	m
MAXIMUM HEIGHT	1.849	m
MAXIMUM WIDTH	2.219	m
ENG INLET AREA	0.773	m^2
BASE DRAG AREA	1.225	m^2
CENTRELINE HT.	2.682	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.034
LANDING CL	1.367
HIGH SPEED CL	1.527
BUFFET MAX CL	0.547
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.509	m^2
TAIL MEAN AERO CHRD	1.206	m
TAIL ASPECT RATIO	1.726	
TAIL THK/CHRD RATIO	0.0456	
NUMBER OF FINS	2	
TOTAL FIN AREA	10.660	m^2
FIN MEAN AERO CHRD	3.850	m
FIN ASPECT RATIO	0.719	
FIN THCK/CHRD RATIO	0.0456	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							17600.0	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17405.4	0.9889
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17294.3	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	16806.6	0.9718
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	15891.9	0.9456
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	15891.9	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	15489.5	0.9747
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	15266.9	0.9856
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	15143.8	0.9919
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	14958.9	0.9878
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	14005.0	0.9362
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	13845.9	0.9886
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	13632.2	0.9846
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	13511.1	0.9911
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	13511.1	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	12679.0	0.9384
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	12679.0	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.8	12615.6	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	12421.0	0.9846

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	231.78
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.01
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.09
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	55.37
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	78.87
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11754.1 kg	WING LOADING	318.35 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.911
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.223
INTERNAL FUEL MASS	5270.9 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.145
AIRCRAFT ALL-UP MASS	17600.0 kg		
AIRCRAFT GROSS MASS	23625.0 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	13158.5	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.251	
ADV. TECH. FACTOR	1.273	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	2204.4
DEVELOPMENT SUPPORT COST	M$	1107.9
FLIGHT TESTING COST	M$	578.8
DEVELOPMENT TOOLING COST	M$	1102.5
MANUFACTURING LABOUR	M$	1208.7
QUALITY CONTROL LABOUR	M$	183.2
MANUFACTURING MATERIAL	M$	102.1
ENGINE DEVELOPMENT COST	M$	3586.1
DEVELOPMENT ENGINE COST	M$	163.4
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16295.1

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3633.7
PRODUCTION TOOLING	M$	1265.5
MANUFACTURING LABOUR	M$	11500.1
QUALITY CNTRL LBR COST	M$	1742.8
MANUFACTURING MATERIAL	M$	2989.2
PRODUCTION ENGINE COST	M$	5273.2
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33355.9
G.S.E. & INITIAL SPARES	M$	3335.6
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY EFFORT	
AIR SERVICES	0.052
FLYING CONTROLS	0.590
FUEL SYSTEM	0.174
HYD. & PNEU.	0.405
UNDERCARRIAGE	0.737
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.705
POWERPLANT SYST.	0.578
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.394
NAV. & COMMS	0.258
ELEC. & INSTR.	0.383
SCHEDULED MAINT.	5.090
TOTAL MAINTENANCE	10.502

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	735.6	
No. SUPP. PERSONNEL	406.3	
OPS. PERSONNEL COST	M$	30.597
SUPP. PERSONNEL COST	M$	16.735
SUPPORT & TRAINING	M$	5.861
UNIT LVL CONSUMPTION	M$	14.606
ENGINE CONTRACT COST	M$	15.703
MECH. CONTRACT COST	M$	19.193
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.643
SUSTAINING SUPPORT	M$	12.842
INSTALLATION SUPPORT	M$	4.554
1st YR BASE O&S COST	M$	138.338
TOTAL 25.YR O&S COST	M$	1768.421

DISPOSAL		
DISPOSAL LABOUR COST	M$	575.0
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-50.2
TOTAL DISPOSAL COST	M$	524.9
DEFLATED DISPOSAL COST	M$	122.3

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.906
PROD.	M$	53.030
GSE&IS	M$	5.380
O & S	M$	45.344
DISP.	M$	0.194
TOTAL	M$	129.855

Aircraft Configuration Data

Single Engine, Canard Configuration. 2 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	2	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.53	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2130.2
TOTAL FUSELAGE GROUP	1678.0
TOTAL EMPENNAGE GROUP	191.6
TOTAL UNDERCARRIAGE	899.0
ENGINE & INSTALLATION	2885.1
SYSTEMS	
FUEL SYSTEM & TANKS	530.9
POWER & FLYING CONTROLS	362.6
HYDRAULICS & PNEUMATICS	245.9
ELECTRICS & BATTERIES	283.0
AIR SYSTEM (AIRCON/ICE)	292.6
COCKPIT INSTRUMENTS ETC	438.2
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	288.5
AIRCRAFT EMPTY MASS	11930.7

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	57.861	m^2
TOTAL SPAN	11.435	m
MEAN AERO CHORD	5.981	m
LDG EDGE SWEEP	52.993	Deg
1/4 CHORD SWEEP	44.990	Deg
ASPECT RATIO	2.260	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0468	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	164.045	kg/s
MAXIMUM THRUST	199.812	kN
MILITARY THRUST	133.644	kN
INLET DIAMETER	1.116	m
MAXIMUM DIAMETER	1.332	m
BARE ENGINE LENGTH	3.375	m
OVERALL LENGTH	6.406	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	17.621	m
EFFECTIVE DIA.	1.971	m
MAXIMUM HEIGHT	1.799	m
MAXIMUM WIDTH	2.159	m
ENG INLET AREA	0.732	m^2
BASE DRAG AREA	1.159	m^2
CENTRELINE HT.	2.788	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.988
LANDING CL	1.307
HIGH SPEED CL	1.489
BUFFET MAX CL	0.523
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.548	m^2
TAIL MEAN AERO CHRD	1.187	m
TAIL ASPECT RATIO	1.808	
TAIL THK/CHRD RATIO	0.0422	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.224	m^2
FIN MEAN AERO CHRD	2.633	m
FIN ASPECT RATIO	0.753	
FIN THCK/CHRD RATIO	0.0422	

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	249.99
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.66
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.09
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.35
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	58.16
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	86.45
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							17601.6	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17417.4	0.9895
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17307.7	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.7	16824.6	0.9721
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.8	15957.6	0.9485
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.8	15957.6	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.8	15575.8	0.9761
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	15359.9	0.9861
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	15254.6	0.9931
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	15088.5	0.9891
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	14185.8	0.9402
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	14026.7	0.9888
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.4	13829.1	0.9859
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	13725.7	0.9925
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	13725.7	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	12933.4	0.9423
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	12933.4	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	85.0	12868.7	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.0	12684.5	0.9857

AIRCRAFT EMPTY MASS	11930.7 kg	WING LOADING	304.21 kg/m^2
CREW INCLUDING EQPT	200.0 kg	THRUST/WEIGHT	0.862
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.211
INTERNAL FUEL MASS	4995.8 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.281
AIRCRAFT ALL-UP MASS	17601.6 kg		
AIRCRAFT GROSS MASS	23626.6 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	13811.5	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.269	
ADV. TECH. FACTOR	1.277	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	2329.9
DEVELOPMENT SUPPORT COST	M$	1160.7
FLIGHT TESTING COST	M$	613.1
DEVELOPMENT TOOLING COST	M$	1160.4
MANUFACTURING LABOUR	M$	1271.6
QUALITY CONTROL LABOUR	M$	192.7
MANUFACTURING MATERIAL	M$	107.1
ENGINE DEVELOPMENT COST	M$	3579.9
DEVELOPMENT ENGINE COST	M$	159.2
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16632.6

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3840.6
PRODUCTION TOOLING	M$	1331.9
MANUFACTURING LABOUR	M$	12099.3
QUALITY CNTRL LBR COST	M$	1833.6
MANUFACTURING MATERIAL	M$	3134.6
PRODUCTION ENGINE COST	M$	5135.2
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	34326.4
G.S.E. & INITIAL SPARES	M$	3432.6
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.053
FLYING CONTROLS	0.635
FUEL SYSTEM	0.175
HYD. & PNEU.	0.442
UNDERCARRIAGE	0.740
OXYGEN SYSTEM	0.076
MISC. MECH	0.005
TOTAL STRUCTURE	0.694
POWERPLANT SYST.	0.572
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.465
NAV. & COMMS	0.248
ELEC. & INSTR.	0.381
SCHEDULED MAINT.	5.033
TOTAL MAINTENANCE	10.576

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	786.3	
No. SUPP. PERSONNEL	418.4	
OPS. PERSONNEL COST	M$	33.592
SUPP. PERSONNEL COST	M$	17.259
SUPPORT & TRAINING	M$	6.261
UNIT LVL CONSUMPTION	M$	14.563
ENGINE CONTRACT COST	M$	15.079
MECH. CONTRACT COST	M$	19.277
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.623
SUSTAINING SUPPORT	M$	13.165
INSTALLATION SUPPORT	M$	4.805
1st YR BASE O&S COST	M$	142.229
TOTAL 25.YR O&S COST	M$	1818.166

DISPOSAL		
DISPOSAL LABOUR COST	M$	605.0
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-51.3
TOTAL DISPOSAL COST	M$	553.7
DEFLATED DISPOSAL COST	M$	129.0

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	26.443
PROD.	M$	54.573
GSE&IS	M$	5.537
O & S	M$	46.620
DISP.	M$	0.205
TOTAL	M$	133.377

Aircraft Configuration Data

Single Engine, Canard Configuration. 2 Crew, 2 Fins. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	2	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	34.42	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2262.5
TOTAL FUSELAGE GROUP	1749.4
TOTAL EMPENNAGE GROUP	374.1
TOTAL UNDERCARRIAGE	940.7
ENGINE & INSTALLATION	3158.1
SYSTEMS	
FUEL SYSTEM & TANKS	559.0
POWER & FLYING CONTROLS	377.0
HYDRAULICS & PNEUMATICS	252.0
ELECTRICS & BATTERIES	294.9
AIR SYSTEM (AIRCON/ICE)	305.9
COCKPIT INSTRUMENTS ETC	447.2
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	315.8
AIRCRAFT EMPTY MASS	12741.7

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	61.699	m^2
TOTAL SPAN	11.824	m
MEAN AERO CHORD	6.168	m
LDG EDGE SWEEP	52.923	Deg
1/4 CHORD SWEEP	44.918	Deg
ASPECT RATIO	2.266	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0446	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	175.042	kg/s
MAXIMUM THRUST	213.208	kN
MILITARY THRUST	142.604	kN
INLET DIAMETER	1.153	m
MAXIMUM DIAMETER	1.371	m
BARE ENGINE LENGTH	3.492	m
OVERALL LENGTH	6.670	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	17.539	m
EFFECTIVE DIA.	2.036	m
MAXIMUM HEIGHT	1.858	m
MAXIMUM WIDTH	2.230	m
ENG INLET AREA	0.781	m^2
BASE DRAG AREA	1.237	m^2
CENTRELINE HT.	2.809	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.989
LANDING CL	1.308
HIGH SPEED CL	1.491
BUFFET MAX CL	0.523
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.683	m^2
TAIL MEAN AERO CHRD	1.217	m
TAIL ASPECT RATIO	1.813	
TAIL THK/CHRD RATIO	0.0401	
NUMBER OF FINS	2	
TOTAL FIN AREA	11.209	m^2
FIN MEAN AERO CHRD	3.852	m
FIN ASPECT RATIO	0.755	
FIN THCK/CHRD RATIO	0.0401	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							18783.7	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	18587.2	0.9895
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	18470.1	0.9937
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.7	17948.4	0.9718
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.8	17011.3	0.9478
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.8	17011.3	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.8	16599.9	0.9758
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	16369.5	0.9861
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	16257.3	0.9931
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	16080.0	0.9891
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	15116.8	0.9401
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	14957.7	0.9895
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.4	14746.8	0.9859
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	14636.4	0.9925
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	14636.4	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	13778.6	0.9414
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	13778.6	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	85.0	13709.7	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.0	13513.2	0.9857

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	250.00
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.67
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.09
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.35
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	61.83
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	92.37
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	12741.7 kg	WING LOADING	304.44 kg/m^2
CREW INCLUDING EQPT	200.0 kg	THRUST/WEIGHT	0.876
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.216
INTERNAL FUEL MASS	5367.0 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.282
AIRCRAFT ALL-UP MASS	18783.7 kg		
AIRCRAFT GROSS MASS	24808.7 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	14806.6	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.278	
ADV. TECH.. FACTOR	1.279	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	14.	
AIRFRAME ENGINEERING COST	M$	2483.1
DEVELOPMENT SUPPORT COST	M$	1237.7
FLIGHT TESTING COST	M$	666.0
DEVELOPMENT TOOLING COST	M$	1232.6
MANUFACTURING LABOUR	M$	1349.9
QUALITY CONTROL LABOUR	M$	204.6
MANUFACTURING MATERIAL	M$	113.1
ENGINE DEVELOPMENT COST	M$	3587.3
DEVELOPMENT ENGINE COST	M$	164.2
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	17096.6

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	4093.1
PRODUCTION TOOLING	M$	1414.8
MANUFACTURING LABOUR	M$	12844.5
QUALITY CNTRL LBR COST	M$	1946.5
MANUFACTURING MATERIAL	M$	3312.0
PRODUCTION ENGINE COST	M$	5298.0
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	35860.2
G.S.E. & INITIAL SPARES	M$	3586.0
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.054
FLYING CONTROLS	0.648
FUEL SYSTEM	0.180
HYD. & PNEU.	0.442
UNDERCARRIAGE	0.755
OXYGEN SYSTEM	0.076
MISC. MECH	0.005
TOTAL STRUCTURE	0.719
POWERPLANT SYST.	0.594
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.458
NAV. & COMMS	0.261
ELEC. & INSTR.	0.396
SCHEDULED MAINT.	5.205
TOTAL MAINTENANCE	1

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	799.1	
No. SUPP. PERSONNEL	420.6	
OPS. PERSONNEL COST	M$	34.101
SUPP. PERSONNEL COST	M$	17.344
SUPPORT & TRAINING	M$	6.358
UNIT LVL CONSUMPTION	M$	14.729
ENGINE CONTRACT COST	M$	15.816
MECH. CONTRACT COST	M$	20.133
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.685
SUSTAINING SUPPORT	M$	13.737
INSTALLATION SUPPORT	M$	4.865
1st YR BASE O&S COST	M$	145.372
TOTAL 25.YR O&S COST	M$	1858.342

DISPOSAL		
DISPOSAL LABOUR COST	M$	642.2
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-54.3
TOTAL DISPOSAL COST	M$	588.0
DEFLATED DISPOSAL COST	M$	137.0

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	27.181
PROD.	M$	57.011
GSE&IS	M$	5.784
O & S	M$	47.650
DISP.	M$	0.218
TOTAL	M$	137.843

Aircraft Configuration Data

Single Engine, Canard Configuration. 1 Crew, 1 Fin. Internal Carriage

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	0.0	kg
MASS OF INT WEAPONS	600.0	kg
VOL. OF INT WEAPONS	2.90	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	34.90	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2322.9
TOTAL FUSELAGE GROUP	1601.5
TOTAL EMPENNAGE GROUP	191.6
TOTAL UNDERCARRIAGE	886.5
ENGINE & INSTALLATION	2834.2
SYSTEMS	
FUEL SYSTEM & TANKS	522.4
POWER & FLYING CONTROLS	358.3
HYDRAULICS & PNEUMATICS	244.1
ELECTRICS & BATTERIES	279.5
AIR SYSTEM (AIRCON/ICE)	230.8
COCKPIT INSTRUMENTS ETC	334.8
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	283.4
AIRCRAFT EMPTY MASS	11795.0

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	60.828	m^2
TOTAL SPAN	11.830	m
MEAN AERO CHORD	6.078	m
LDG EDGE SWEEP	57.369	Deg
1/4 CHORD SWEEP	51.127	Deg
ASPECT RATIO	2.301	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0530	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	161.950	kg/s
MAXIMUM THRUST	197.260	kN
MILITARY THRUST	131.937	kN
INLET DIAMETER	1.109	m
MAXIMUM DIAMETER	1.325	m
BARE ENGINE LENGTH	3.352	m
OVERALL LENGTH	6.355	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	17.191	m
EFFECTIVE DIA.	1.958	m
MAXIMUM HEIGHT	1.788	m
MAXIMUM WIDTH	2.145	m
ENG INLET AREA	0.722	m^2
BASE DRAG AREA	1.144	m^2
CENTRELINE HT.	2.736	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.928
LANDING CL	1.228
HIGH SPEED CL	1.422
BUFFET MAX CL	0.491
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.668	m^2
TAIL MEAN AERO CHRD	1.204	m
TAIL ASPECT RATIO	1.840	
TAIL THK/CHRD RATIO	0.0477	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.112	m^2
FIN MEAN AERO CHRD	2.582	m
FIN ASPECT RATIO	0.767	
FIN THCK/CHRD RATIO	0.0477	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							17374.5	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17192.6	0.9895
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17084.3	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	16615.4	0.9726
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	15786.8	0.9501
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	15786.8	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	15417.5	0.9766
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	15203.8	0.9861
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.3	15100.1	0.9932
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	14945.9	0.9898
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	14054.8	0.9404
11 INTRNL WEAPONS	8382.0	0.90	0.26	0.00	1.0	53.8	13895.8	0.9887
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	13685.2	0.9848
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	13582.4	0.9925
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	13582.4	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	12814.2	0.9434
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	12814.2	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	84.9	12750.2	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	12568.3	0.9857

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	250.00
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.81
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.21
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.31
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.63
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	58.98
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	82.78
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11795.0 kg	WING LOADING	285.63 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.864
TOTAL WEAPONS MASS	600.0 kg	FUEL FRACTION	0.210
INTERNAL FUEL MASS	4879.5 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.285
AIRCRAFT ALL-UP MASS	17374.5 kg		
AIRCRAFT GROSS MASS	23274.5 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	13968.8	lb.	No. DEVELOPMENT AIRCRAFT	9		No. PRODUCTION AIRCRAFT	620		AIR SERVICES	0.053
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES	2		AIFRAME ENGINEERING	M$ 3924.8		FLYING CONTROLS	0.577
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES	14.		PRODUCTION TOOLING	M$ 1357.8		FUEL SYSTEM	0.172
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$ 2381.0		MANUFACTURING LABOUR	M$ 12333.1		HYD. & PNEU.	0.428
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$ 1175.1		QUALITY CNTRL LBR COST	M$ 1869.0		UNDERCARRIAGE	0.737
ADV. MATERIAL FACTOR	1.282		FLIGHT TESTING COST	M$ 621.4		MANUFACTURING MATERIAL	M$ 3192.7		OXYGEN SYSTEM	0.079
ADV. TECH. FACTOR	1.280		DEVELOPMENT TOOLING COST	M$ 1183.0		PRODUCTION ENGINE COST	M$ 5103.5		MISC. MECH	0.005
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$ 1296.2		AVIONICS PURCHASE COST	M$ 6951.3		TOTAL STRUCTURE	0.674
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$ 196.4					POWERPLANT SYST.	0.563
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$ 109.1		TOTAL PRODUCTION COST	M$ 34732.2		ARMAMENT SYSTEM	1.058
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$ 3578.4					AVIONICS SYSTEM	0.441
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$ 158.2		G.S.E. & INITIAL SPARES	M$ 3473.2		NAV. & COMMS	0.245
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$ 3352.2					ELEC. & INSTR.	0.375
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$ 78.5		ENGINEERING LABOUR RATE	118.5 $/hr		SCHEDULED MAINT.	5.020
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$ 2627.4		TOOLING/QUALITY LABOUR RT	86.8 $/hr		TOTAL MAINTENANCE	10.425
			TOTAL DEVELOPMENT COST	M$16756.9		MANUFACTURING LABOUR RATE	74.5 $/hr			

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	732.5	
No. SUPP. PERSONNEL	405.7	
OPS. PERSONNEL COST	M$	30.474
SUPP. PERSONNEL COST	M$	16.715
SUPPORT & TRAINING	M$	5.837
UNIT LVL CONSUMPTION	M$	14.504
ENGINE CONTRACT COST	M$	14.939
MECH. CONTRACT COST	M$	19.002
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.606
SUSTAINING SUPPORT	M$	13.293
INSTALLATION SUPPORT	M$	4.540
1st YR BASE O&S COST	M$	137.515
TOTAL 25.YR O&S COST	M$	1757.896

DISPOSAL

DISPOSAL LABOUR COST	M$	616.7
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-50.8
TOTAL DISPOSAL COST	M$	565.9
DEFLATED DISPOSAL COST	M$	131.9

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	26.640
PROD.	M$	55.218
GSE&IS	M$	5.602
O & S	M$	45.074
DISP.	M$	0.210
TOTAL	M$	132.744

Aircraft Configuration Data

Single Engine, Canard Configuration. 1 Crew, 1 Fin. Mass-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.48	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1911.8
TOTAL FUSELAGE GROUP	1565.7
TOTAL EMPENNAGE GROUP	181.1
TOTAL UNDERCARRIAGE	846.2
ENGINE & INSTALLATION SYSTEMS	2571.4
FUEL SYSTEM & TANKS	496.6
POWER & FLYING CONTROLS	344.2
HYDRAULICS & PNEUMATICS	238.1
ELECTRICS & BATTERIES	268.0
AIR SYSTEM (AIRCON/ICE)	220.9
COCKPIT INSTRUMENTS ETC	323.3
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	257.1
AIRCRAFT EMPTY MASS	10929.3

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	51.294	m^2
TOTAL SPAN	10.833	m
MEAN AERO CHORD	5.597	m
LDG EDGE SWEEP	50.331	Deg
1/4 CHORD SWEEP	41.437	Deg
ASPECT RATIO	2.288	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0457	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	1.	
AIR MASS FLOW RATE	150.875	kg/s
MAXIMUM THRUST	183.771	kN
MILITARY THRUST	122.915	kN
INLET DIAMETER	1.070	m
MAXIMUM DIAMETER	1.284	m
BARE ENGINE LENGTH	3.229	m
OVERALL LENGTH	6.081	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	17.545	m
EFFECTIVE DIA.	1.890	m
MAXIMUM HEIGHT	1.725	m
MAXIMUM WIDTH	2.070	m
ENG INLET AREA	0.673	m^2
BASE DRAG AREA	1.066	m^2
CENTRELINE HT.	2.743	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.021
LANDING CL	1.350
HIGH SPEED CL	1.532
BUFFET MAX CL	0.540
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.322	m^2
TAIL MEAN AERO CHRD	1.126	m
TAIL ASPECT RATIO	1.830	
TAIL THK/CHRD RATIO	0.0411	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.059	m^2
FIN MEAN AERO CHRD	2.576	m
FIN ASPECT RATIO	0.763	
FIN THCK/CHRD RATIO	0.0411	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							16112.3	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15942.9	0.9895
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15842.3	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	15396.6	0.9719
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	14595.1	0.9479
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	14595.1	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	14242.5	0.9758
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	14044.2	0.9861
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.3	13946.5	0.9930
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	13790.9	0.9888
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	12960.7	0.9398
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	12801.6	0.9877
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	12620.4	0.9858
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	12524.6	0.9924
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	12524.6	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	11792.9	0.9416
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	11792.9	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	85.0	11734.0	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.0	11564.6	0.9856

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	248.37
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.65
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.07
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.29
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	54.58
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	79.26
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10929.3	kg	WING LOADING 314.12 kg/m^2
CREW INCLUDING EQPT	100.0	kg	THRUST/WEIGHT 0.847
TOTAL WEAPONS MASS	475.0	kg	FUEL FRACTION 0.208
INTERNAL FUEL MASS	4608.1	kg	XS FUEL VOL. 0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.269
AIRCRAFT ALL-UP MASS	16112.3	kg	
AIRCRAFT GROSS MASS	22137.3	kg	

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	12819.9	lb.	No. DEVELOPMENT AIRCRAFT	9		No. PRODUCTION AIRCRAFT	620		AIR SERVICES	0.052
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES	2		AIFRAME ENGINEERING	M$ 3617.7		FLYING CONTROLS	0.564
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES	14.		PRODUCTION TOOLING	M$ 1257.4		FUEL SYSTEM	0.168
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$ 2194.7		MANUFACTURING LABOUR	M$ 11428.8		HYD. & PNEU.	0.436
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$ 1085.1		QUALITY CNTRL LBR COST	M$ 1732.0		UNDERCARRIAGE	0.722
ADV. MATERIAL FACTOR	1.269		FLIGHT TESTING COST	M$ 561.1		MANUFACTURING MATERIAL	M$ 2975.9		OXYGEN SYSTEM	0.075
ADV. TECH. FACTOR	1.277		DEVELOPMENT TOOLING COST	M$ 1095.5		PRODUCTION ENGINE COST	M$ 4932.5		MISC. MECH	0.005
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$ 1201.2		AVIONICS PURCHASE COST	M$ 6951.3		TOTAL STRUCTURE	0.651
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$ 182.0					POWERPLANT SYST.	0.544
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$ 101.7		TOTAL PRODUCTION COST	M$ 32895.7		ARMAMENT SYSTEM	1.058
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$ 3571.0					AVIONICS SYSTEM	0.462
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$ 152.9		G.S.E. & INITIAL SPARES	M$ 3289.6		NAV. & COMMS	0.232
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$ 3352.2					ELEC. & INSTR.	0.362
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$ 78.5		ENGINEERING LABOUR RATE	118.5 $/hr		SCHEDULED MAINT.	4.842
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$ 2627.4		TOOLING/QUALITY LABOUR RT	86.8 $/hr		TOTAL MAINTENANCE	10.175
			TOTAL DEVELOPMENT COST	M$16203.1		MANUFACTURING LABOUR RATE	74.5 $/hr			

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	721.2	
No. SUPP. PERSONNEL	403.8	
OPS. PERSONNEL COST	M$	30.025
SUPP. PERSONNEL COST	M$	16.640
SUPPORT & TRAINING	M$	5.752
UNIT LVL CONSUMPTION	M$	14.352
ENGINE CONTRACT COST	M$	14.196
MECH. CONTRACT COST	M$	18.136
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.545
SUSTAINING SUPPORT	M$	12.617
INSTALLATION SUPPORT	M$	4.487
1st YR BASE O&S COST	M$	134.355
TOTAL 25.YR O&S COST	M$	1717.508

DISPOSAL		
DISPOSAL LABOUR COST	M$	571.4
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-47.7
TOTAL DISPOSAL COST	M$	523.8
DEFLATED DISPOSAL COST	M$	122.1

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.760
PROD.	M$	52.298
GSE&IS	M$	5.306
O & S	M$	44.039
DISP.	M$	0.194
TOTAL	M$	127.597

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.65	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	37.47	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2345.5
TOTAL FUSELAGE GROUP	1892.5
TOTAL EMPENNAGE GROUP	300.9
TOTAL UNDERCARRIAGE	918.8
ENGINE & INSTALLATION	2630.3
SYSTEMS	
FUEL SYSTEM & TANKS	548.9
POWER & FLYING CONTROLS	369.5
HYDRAULICS & PNEUMATICS	248.8
ELECTRICS & BATTERIES	288.6
AIR SYSTEM (AIRCON/ICE)	238.8
COCKPIT INSTRUMENTS ETC	362.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	263.0
AIRCRAFT EMPTY MASS	12112.8

WING PARAMETERS

WING TYPE	CONV. & LEX	
GROSS AREA	58.067	m^2
TOTAL SPAN	11.913	m
MEAN AERO CHORD	5.459	m
LDG EDGE SWEEP	39.283	Deg
1/4 CHORD SWEEP	29.791	Deg
ASPECT RATIO	2.444	
TAPER RATIO	0.250	
THCK/CHRD RATIO	0.0350	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	90.309	kg/s
MAXIMUM THRUST	219.999	kN
MILITARY THRUST	147.146	kN
INLET DIAMETER	0.828	m
MAXIMUM DIAMETER	1.024	m
BARE ENGINE LENGTH	2.464	m
OVERALL LENGTH	4.426	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.933	m
EFFECTIVE DIA.	2.177	m
MAXIMUM HEIGHT	1.229	m
MAXIMUM WIDTH	3.291	m
ENG INLET AREA	0.806	m^2
BASE DRAG AREA	1.457	m^2
CENTRELINE HT.	2.322	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.129
LANDING CL	1.716
HIGH SPEED CL	1.495
BUFFET MAX CL	0.243
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	AFT-TAIL	
NET TAIL AREA	10.367	m^2
TAIL MEAN AERO CHRD	2.303	m
TAIL ASPECT RATIO	1.955	
TAIL THK/CHRD RATIO	0.0315	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.219	m^2
FIN MEAN AERO CHRD	2.923	m
FIN ASPECT RATIO	0.611	
FIN THCK/CHRD RATIO	0.0315	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							18162.2	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17959.4	0.9888
2	TAKE-OFF	15.0	0.23	0.05	1.20	1.0	1.2	17848.9	0.9938
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	17317.3	0.9702
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	16372.7	0.9455
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	16372.7	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	15963.9	0.9750
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	15733.2	0.9855
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.3	15606.3	0.9919
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	15427.8	0.9886
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	14434.0	0.9356
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	14274.8	0.9890
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	14052.0	0.9844
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	13927.3	0.9911
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	13927.3	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	13057.4	0.9375
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	13057.4	1.0000
17	LANDING	0.0	0.15	0.00	0.00	0.0	84.9	12992.1	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	12789.3	0.9844

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	205.65
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	391.21
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	7.0	7.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	7.50
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	6.16
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.45
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	56.43
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	80.62
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	12112.8 kg	WING LOADING	312.78 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.927
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.226
INTERNAL FUEL MASS	5474.5 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.160
AIRCRAFT ALL-UP MASS	18162.2 kg		
AIRCRAFT GROSS MASS	24187.2 kg		

Aircraft Cost Data
Twin Engine, Aft-Tail Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	15026.6	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.308	
ADV. TECH. FACTOR	1.285	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2583.7
DEVELOPMENT SUPPORT COST	M$	1260.2
FLIGHT TESTING COST	M$	677.8
DEVELOPMENT TOOLING COST	M$	1275.9
MANUFACTURING LABOUR	M$	1397.3
QUALITY CONTROL LABOUR	M$	211.7
MANUFACTURING MATERIAL	M$	117.0
ENGINE DEVELOPMENT COST	M$	3530.2
DEVELOPMENT ENGINE COST	M$	173.4
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	17285.4

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	4259.0
PRODUCTION TOOLING	M$	1464.5
MANUFACTURING LABOUR	M$	13294.7
QUALITY CNTRL LBR COST	M$	2014.7
MANUFACTURING MATERIAL	M$	3424.6
PRODUCTION ENGINE COST	M$	7300.8
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	38709.6
G.S.E. & INITIAL SPARES	M$	3871.0
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.050
FLYING CONTROLS	0.599
FUEL SYSTEM	0.162
HYD. & PNEU.	0.292
UNDERCARRIAGE	0.743
OXYGEN SYSTEM	0.075
MISC. MECH	0.005
TOTAL STRUCTURE	0.660
POWERPLANT SYST.	0.553
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.195
NAV. & COMMS	0.264
ELEC. & INSTR.	0.362
SCHEDULED MAINT.	5.276
TOTAL MAINTENANCE	10.294

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	719.2	
No. SUPP. PERSONNEL	403.5	
OPS. PERSONNEL COST	M$	29.944
SUPP. PERSONNEL COST	M$	16.627
SUPPORT & TRAINING	M$	5.742
UNIT LVL CONSUMPTION	M$	15.303
ENGINE CONTRACT COST	M$	30.761
MECH. CONTRACT COST	M$	19.217
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.201
SUSTAINING SUPPORT	M$	15.041
INSTALLATION SUPPORT	M$	4.478
1st YR BASE O&S COST	M$	155.919
TOTAL 25.YR O&S COST	M$	1993.169

DISPOSAL

DISPOSAL LABOUR COST	M$	664.7
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-52.5
TOTAL DISPOSAL COST	M$	612.3
DEFLATED DISPOSAL COST	M$	142.7

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	27.481
PROD.	M$	61.542
GSE&IS	M$	6.243
O & S	M$	51.107
DISP.	M$	0.227
TOTAL	M$	146.599

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER
NUMBER OF CREW	1
MAX HI-ALT MACH NO:	2.20
MAX LO-ALT MACH NO:	1.20
DESIGN MAX G LOAD	9.00
ULTIMATE LOAD FACTOR	13.50
MASS OF EXT WEAPONS	475.0 kg
MASS OF INT WEAPONS	0.0 kg
VOL. OF INT WEAPONS	0.00 m^3
VOL. OF GUN & AMMO.	0.31 m^3
% ADVANCED MATERIALS	32.74

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1716.3
TOTAL FUSELAGE GROUP	1404.3
TOTAL EMPENNAGE GROUP	113.7
TOTAL UNDERCARRIAGE	805.3
ENGINE & INSTALLATION	2083.1
SYSTEMS	
FUEL SYSTEM & TANKS	473.2
POWER & FLYING CONTROLS	329.8
HYDRAULICS & PNEUMATICS	231.8
ELECTRICS & BATTERIES	256.2
AIR SYSTEM (AIRCON/ICE)	210.7
COCKPIT INSTRUMENTS ETC	341.7
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	208.3
AIRCRAFT EMPTY MASS	9879.5

WING PARAMETERS

WING TYPE	DELTA WING
GROSS AREA	56.766 m^2
TOTAL SPAN	10.710 m
MEAN AERO CHORD	6.483 m
LDG EDGE SWEEP	56.048 Deg
1/4 CHORD SWEEP	47.212 Deg
ASPECT RATIO	2.021
TAPER RATIO	0.100
THCK/CHRD RATIO	0.0462

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.
AIR MASS FLOW RATE	74.736 kg/s
MAXIMUM THRUST	182.063 kN
MILITARY THRUST	121.773 kN
INLET DIAMETER	0.753 m
MAXIMUM DIAMETER	0.943 m
BARE ENGINE LENGTH	2.230 m
OVERALL LENGTH	3.939 m
FAN BYPASS RATIO	0.408
COMP. PRESS. RATIO	26.001

FUSELAGE PARAMETERS

OVERALL LENGTH	14.428 m
EFFECTIVE DIA.	1.980 m
MAXIMUM HEIGHT	1.131 m
MAXIMUM WIDTH	2.965 m
ENG INLET AREA	0.667 m^2
BASE DRAG AREA	1.206 m^2
CENTRELINE HT.	2.112 m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.892
LANDING CL	1.179
HIGH SPEED CL	1.260
BUFFET MAX CL	0.472
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	NONE
NET TAIL AREA	0.000 m^2
TAIL MEAN AERO CHRD	0.000 m
TAIL ASPECT RATIO	1.617
TAIL THK/CHRD RATIO	0.0416
NUMBER OF FINS	1
TOTAL FIN AREA	4.561 m^2
FIN MEAN AERO CHRD	2.602 m
FIN ASPECT RATIO	0.674
FIN THCK/CHRD RATIO	0.0416

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	218.21
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	479.84
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.13
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	7.50
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.24
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	37.84
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	65.45
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							14963.7	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	14795.8	0.9888
2 TAKE-OFF	15.0	0.24	0.05	1.20	1.0	1.2	14702.5	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.4	14283.5	0.9715
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.5	13503.6	0.9454
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.5	13503.6	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.5	13163.1	0.9748
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.1	12973.4	0.9856
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.1	12863.3	0.9915
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.5	12714.4	0.9884
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.5	11891.9	0.9353
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.6	11732.8	0.9866
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.1	11551.1	0.9845
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.1	11442.9	0.9906
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.1	11442.9	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.2	10731.6	0.9378
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.5	10731.6	1.0000
17 LANDING	0.0	0.17	0.00	0.00	0.0	84.7	10677.9	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.7	10510.1	0.9843

AIRCRAFT EMPTY MASS	9879.5 kg	WING LOADING	263.60 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.885
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.215
INTERNAL FUEL MASS	4509.2 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.100
AIRCRAFT ALL-UP MASS	14963.7 kg		
AIRCRAFT GROSS MASS	20988.7 kg		

Aircraft Cost Data
Twin Engine, Delta Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	11759.7	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.261	
ADV. TECH. FACTOR	1.275	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2034.9
DEVELOPMENT SUPPORT COST	M$	1002.5
FLIGHT TESTING COST	M$	506.3
DEVELOPMENT TOOLING COST	M$	1019.4
MANUFACTURING LABOUR	M$	1118.5
QUALITY CONTROL LABOUR	M$	169.5
MANUFACTURING MATERIAL	M$	95.2
ENGINE DEVELOPMENT COST	M$	3519.7
DEVELOPMENT ENGINE COST	M$	158.3
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15682.4

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3354.4
PRODUCTION TOOLING	M$	1170.0
MANUFACTURING LABOUR	M$	10642.1
QUALITY CNTRL LBR COST	M$	1612.8
MANUFACTURING MATERIAL	M$	2787.4
PRODUCTION ENGINE COST	M$	6665.3
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33183.4
G.S.E. & INITIAL SPARES	M$	3318.3
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.045
FLYING CONTROLS	0.563
FUEL SYSTEM	0.141
HYD. & PNEU.	0.246
UNDERCARRIAGE	0.704
OXYGEN SYSTEM	0.074
MISC. MECH	0.005
TOTAL STRUCTURE	0.612
POWERPLANT SYST.	0.483
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.136
NAV. & COMMS	0.226
ELEC. & INSTR.	0.309
SCHEDULED MAINT.	4.767
TOTAL MAINTENANCE	9.369

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	674.4	
No. SUPP. PERSONNEL	395.8	
OPS. PERSONNEL COST	M$	28.165
SUPP. PERSONNEL COST	M$	16.330
SUPPORT & TRAINING	M$	5.407
UNIT LVL CONSUMPTION	M$	14.724
ENGINE CONTRACT COST	M$	26.794
MECH. CONTRACT COST	M$	16.693
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.953
SUSTAINING SUPPORT	M$	12.965
INSTALLATION SUPPORT	M$	4.269
1st YR BASE O&S COST	M$	143.905
TOTAL 25.YR O&S COST	M$	1839.589

DISPOSAL		
DISPOSAL LABOUR COST	M$	532.1
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-44.2
TOTAL DISPOSAL COST	M$	488.0
DEFLATED DISPOSAL COST	M$	113.7

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	24.932
PROD.	M$	52.756
GSE&IS	M$	5.352
O & S	M$	47.169
DISP.	M$	0.181
TOTAL	M$	130.390

Aircraft Configuration Data

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.71	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1644.9
TOTAL FUSELAGE GROUP	1554.5
TOTAL EMPENNAGE GROUP	172.1
TOTAL UNDERCARRIAGE	807.7
ENGINE & INSTALLATION	2067.1
SYSTEMS	
FUEL SYSTEM & TANKS	473.8
POWER & FLYING CONTROLS	330.7
HYDRAULICS & PNEUMATICS	232.2
ELECTRICS & BATTERIES	256.9
AIR SYSTEM (AIRCON/ICE)	211.4
COCKPIT INSTRUMENTS ETC	338.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	206.7
AIRCRAFT EMPTY MASS	10001.0

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	47.841	m^2
TOTAL SPAN	10.998	m
MEAN AERO CHORD	5.142	m
LDG EDGE SWEEP	49.616	Deg
1/4 CHORD SWEEP	41.456	Deg
ASPECT RATIO	2.529	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0511	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	74.264	kg/s
MAXIMUM THRUST	180.913	kN
MILITARY THRUST	121.003	kN
INLET DIAMETER	0.751	m
MAXIMUM DIAMETER	0.940	m
BARE ENGINE LENGTH	2.222	m
OVERALL LENGTH	3.924	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.491	m
EFFECTIVE DIA.	1.974	m
MAXIMUM HEIGHT	1.128	m
MAXIMUM WIDTH	2.955	m
ENG INLET AREA	0.663	m^2
BASE DRAG AREA	1.199	m^2
CENTRELINE HT.	2.224	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.020
LANDING CL	1.349
HIGH SPEED CL	1.566
BUFFET MAX CL	0.540
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.276	m^2
TAIL MEAN AERO CHRD	1.061	m
TAIL ASPECT RATIO	2.023	
TAIL THK/CHRD RATIO	0.0460	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.781	m^2
FIN MEAN AERO CHRD	2.382	m
FIN ASPECT RATIO	0.843	
FIN THCK/CHRD RATIO	0.0460	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15033.3	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	14866.6	0.9889
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	14771.5	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	14346.8	0.9712
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	13586.7	0.9470
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	13586.7	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13252.5	0.9754
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13061.6	0.9856
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	12956.4	0.9919
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	12815.1	0.9891
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	11997.9	0.9362
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	11838.7	0.9867
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	11655.8	0.9845
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	11552.7	0.9912
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	11552.7	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	10850.0	0.9392
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	10850.0	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.9	10795.8	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	10629.0	0.9846

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	230.85
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.10
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.32
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.81
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.44
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	43.66
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	68.41
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10001.0 kg	WING LOADING	314.24 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.876
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.212
INTERNAL FUEL MASS	4457.3 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.147
AIRCRAFT ALL-UP MASS	15033.3 kg		
AIRCRAFT GROSS MASS	21058.3 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12067.7	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.271	
ADV. TECH. FACTOR	1.277	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$ 2096.1	
DEVELOPMENT SUPPORT COST	M$ 1027.8	
FLIGHT TESTING COST	M$ 522.1	
DEVELOPMENT TOOLING COST	M$ 1047.7	
MANUFACTURING LABOUR	M$ 1149.3	
QUALITY CONTROL LABOUR	M$ 174.2	
MANUFACTURING MATERIAL	M$ 97.7	
ENGINE DEVELOPMENT COST	M$ 3519.4	
DEVELOPMENT ENGINE COST	M$ 157.8	
AVION. DEVELOPMENT COST	M$ 3352.2	
AVIONICS PURCHASE COST	M$ 78.5	
SOFTWARE DEVELOPMENT	M$ 2627.4	
TOTAL DEVELOPMENT COST	M$15850.2	

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$ 3455.3	
PRODUCTION TOOLING	M$ 1202.5	
MANUFACTURING LABOUR	M$ 10935.4	
QUALITY CNTRL LBR COST	M$ 1657.2	
MANUFACTURING MATERIAL	M$ 2859.2	
PRODUCTION ENGINE COST	M$ 6645.0	
AVIONICS PURCHASE COST	M$ 6951.3	
TOTAL PRODUCTION COST	M$ 33705.9	
G.S.E. & INITIAL SPARES	M$ 3370.6	
ENGINEERING LABOUR RATE	118.5 $/hr	
TOOLING/QUALITY LABOUR RT	86.8 $/hr	
MANUFACTURING LABOUR RATE	74.5 $/hr	

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.046
FLYING CONTROLS	0.562
FUEL SYSTEM	0.145
HYD. & PNEU.	0.280
UNDERCARRIAGE	0.706
OXYGEN SYSTEM	0.077
MISC. MECH	0.005
TOTAL STRUCTURE	0.609
POWERPLANT SYST.	0.491
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.196
NAV. & COMMS	0.226
ELEC. & INSTR.	0.317
SCHEDULED MAINT.	4.766
TOTAL MAINTENANCE	9.484

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	681.3	
No. SUPP. PERSONNEL	397.0	
OPS. PERSONNEL COST	M$ 28.441	
SUPP. PERSONNEL COST	M$ 16.376	
SUPPORT & TRAINING	M$ 5.457	
UNIT LVL CONSUMPTION	M$ 14.741	
ENGINE CONTRACT COST	M$ 26.674	
MECH. CONTRACT COST	M$ 16.856	
AVIONICS CNTRCT COST	M$ 16.605	
SUPPLY CONTRACT COST	M$ 1.955	
SUSTAINING SUPPORT	M$ 13.148	
INSTALLATION SUPPORT	M$ 4.301	
1st YR BASE O&S COST	M$ 144.553	
TOTAL 25.YR O&S COST	M$1847.877	

DISPOSAL		
DISPOSAL LABOUR COST	M$ 546.8	
ADVANCED MATERIAL DISP.	M$ 0.1	
METALS SCRAP VALUE	M$ -44.7	
TOTAL DISPOSAL COST	M$ 502.1	
DEFLATED DISPOSAL COST	M$ 117.0	

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.199
PROD.	M$	53.587
GSE&IS	M$	5.436
O & S	M$	47.381
DISP.	M$	0.186
TOTAL	M$	131.789

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	34.26	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1689.0
TOTAL FUSELAGE GROUP	1615.4
TOTAL EMPENNAGE GROUP	338.2
TOTAL UNDERCARRIAGE	846.7
ENGINE & INSTALLATION SYSTEMS	2288.2
FUEL SYSTEM & TANKS	501.7
POWER & FLYING CONTROLS	344.4
HYDRAULICS & PNEUMATICS	238.1
ELECTRICS & BATTERIES	268.1
AIR SYSTEM (AIRCON/ICE)	221.0
COCKPIT INSTRUMENTS ETC	347.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	228.8
AIRCRAFT EMPTY MASS	10631.6

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	52.871	m^2
TOTAL SPAN	10.713	m
MEAN AERO CHORD	5.834	m
LDG EDGE SWEEP	53.148	Deg
1/4 CHORD SWEEP	44.819	Deg
ASPECT RATIO	2.171	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0491	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	80.709	kg/s
MAXIMUM THRUST	196.612	kN
MILITARY THRUST	131.504	kN
INLET DIAMETER	0.783	m
MAXIMUM DIAMETER	0.975	m
BARE ENGINE LENGTH	2.322	m
OVERALL LENGTH	4.130	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.211	m
EFFECTIVE DIA.	2.058	m
MAXIMUM HEIGHT	1.170	m
MAXIMUM WIDTH	3.093	m
ENG INLET AREA	0.720	m^2
BASE DRAG AREA	1.303	m^2
CENTRELINE HT.	2.215	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.990
LANDING CL	1.309
HIGH SPEED CL	1.478
BUFFET MAX CL	0.524
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.474	m^2
TAIL MEAN AERO CHRD	1.194	m
TAIL ASPECT RATIO	1.737	
TAIL THK/CHRD RATIO	0.0442	
NUMBER OF FINS	2	
TOTAL FIN AREA	10.227	m^2
FIN MEAN AERO CHRD	3.760	m
FIN ASPECT RATIO	0.724	
FIN THCK/CHRD RATIO	0.0442	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							16127.2	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15946.0	0.9888
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15843.8	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	15387.7	0.9712
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14527.8	0.9441
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14527.8	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	14153.3	0.9742
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13948.0	0.9855
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13829.5	0.9915
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13665.0	0.9881
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12776.9	0.9350
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12617.7	0.9875
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12420.8	0.9844
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.1	12304.3	0.9906
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.1	12304.3	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.2	11520.6	0.9363
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11520.6	1.0000
17 LANDING	0.0	0.17	0.00	0.00	0.0	84.8	11463.0	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	11281.8	0.9842

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	226.71
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.71
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.13
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.21
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.15
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	43.37
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	69.73
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10631.6	kg	WING LOADING 305.03 kg/m^2
CREW INCLUDING EQPT	100.0	kg	THRUST/WEIGHT 0.905
TOTAL WEAPONS MASS	475.0	kg	FUEL FRACTION 0.222
INTERNAL FUEL MASS	4920.6	kg	XS FUEL VOL. 0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.104
AIRCRAFT ALL-UP MASS	16127.2	kg	
AIRCRAFT GROSS MASS	22152.2	kg	

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12802.1	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.276	
ADV. TECH. FACTOR	1.279	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2208.3
DEVELOPMENT SUPPORT COST	M$	1085.0
FLIGHT TESTING COST	M$	560.2
DEVELOPMENT TOOLING COST	M$	1101.0
MANUFACTURING LABOUR	M$	1207.2
QUALITY CONTROL LABOUR	M$	182.9
MANUFACTURING MATERIAL	M$	102.2
ENGINE DEVELOPMENT COST	M$	3523.7
DEVELOPMENT ENGINE COST	M$	164.2
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16192.8

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3640.2
PRODUCTION TOOLING	M$	1263.7
MANUFACTURING LABOUR	M$	11486.1
QUALITY CNTRL LBR COST	M$	1740.7
MANUFACTURING MATERIAL	M$	2991.1
PRODUCTION ENGINE COST	M$	6916.5
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	34989.5
G.S.E. & INITIAL SPARES	M$	3498.9
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.047
FLYING CONTROLS	0.576
FUEL SYSTEM	0.150
HYD. & PNEU.	0.271
UNDERCARRIAGE	0.717
OXYGEN SYSTEM	0.073
MISC. MECH	0.005
TOTAL STRUCTURE	0.630
POWERPLANT SYST.	0.511
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.170
NAV. & COMMS	0.241
ELEC. & INSTR.	0.331
SCHEDULED MAINT.	4.941
TOTAL MAINTENANCE	9.721

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	691.9	
No. SUPP. PERSONNEL	398.8	
OPS. PERSONNEL COST	M$	28.860
SUPP. PERSONNEL COST	M$	16.446
SUPPORT & TRAINING	M$	5.537
UNIT LVL CONSUMPTION	M$	14.949
ENGINE CONTRACT COST	M$	28.316
MECH. CONTRACT COST	M$	17.629
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.047
SUSTAINING SUPPORT	M$	13.652
INSTALLATION SUPPORT	M$	4.350
1st YR BASE O&S COST	M$	148.391
TOTAL 25.YR O&S COST	M$	1896.928

DISPOSAL		
DISPOSAL LABOUR COST	M$	574.3
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-47.0
TOTAL DISPOSAL COST	M$	527.3
DEFLATED DISPOSAL COST	M$	122.9

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.744
PROD.	M$	55.627
GSE&IS	M$	5.643
O & S	M$	48.639
DISP.	M$	0.195
TOTAL	M$	135.849

Aircraft Configuration Data

Twin Engine, Canard Configuration. 2 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	2	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	32.29	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1732.9
TOTAL FUSELAGE GROUP	1740.6
TOTAL EMPENNAGE GROUP	186.6
TOTAL UNDERCARRIAGE	869.6
ENGINE & INSTALLATION SYSTEMS	2403.6
FUEL SYSTEM & TANKS	516.6
POWER & FLYING CONTROLS	352.4
HYDRAULICS & PNEUMATICS	241.6
ELECTRICS & BATTERIES	274.7
AIR SYSTEM (AIRCON/ICE)	283.3
COCKPIT INSTRUMENTS ETC	455.7
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	240.4
AIRCRAFT EMPTY MASS	11002.9

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	56.853	m^2
TOTAL SPAN	10.722	m
MEAN AERO CHORD	6.268	m
LDG EDGE SWEEP	56.028	Deg
1/4 CHORD SWEEP	48.205	Deg
ASPECT RATIO	2.022	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0523	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	83.995	kg/s
MAXIMUM THRUST	204.617	kN
MILITARY THRUST	136.858	kN
INLET DIAMETER	0.799	m
MAXIMUM DIAMETER	0.992	m
BARE ENGINE LENGTH	2.371	m
OVERALL LENGTH	4.233	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.513	m
EFFECTIVE DIA.	2.099	m
MAXIMUM HEIGHT	1.191	m
MAXIMUM WIDTH	3.162	m
ENG INLET AREA	0.749	m^2
BASE DRAG AREA	1.356	m^2
CENTRELINE HT.	2.258	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.958
LANDING CL	1.266
HIGH SPEED CL	1.420
BUFFET MAX CL	0.507
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.604	m^2
TAIL MEAN AERO CHRD	1.269	m
TAIL ASPECT RATIO	1.618	
TAIL THK/CHRD RATIO	0.0470	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.996	m^2
FIN MEAN AERO CHRD	2.722	m
FIN ASPECT RATIO	0.674	
FIN THCK/CHRD RATIO	0.0470	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							16772.8	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	16584.2	0.9888
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	16477.9	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.4	16011.9	0.9717
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.5	15125.4	0.9446
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.5	15125.4	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.5	14737.5	0.9744
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.1	14523.9	0.9855
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.1	14399.3	0.9914
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.5	14226.7	0.9880
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.5	13302.3	0.9350
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.6	13143.2	0.9880
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.1	12938.9	0.9845
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.1	12816.2	0.9905
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.1	12816.2	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.2	12010.1	0.9371
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.5	12010.1	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.7	11950.0	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.7	11761.4	0.9842

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	226.53
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.62
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.06
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.05
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	42.83
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	70.87
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11002.9 kg	WING LOADING	295.02 kg/m^2
CREW INCLUDING EQPT	200.0 kg	THRUST/WEIGHT	0.915
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.223
INTERNAL FUEL MASS	5094.9 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.092
AIRCRAFT ALL-UP MASS	16772.8 kg		
AIRCRAFT GROSS MASS	22797.8 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12977.0	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.257	
ADV. TECH. FACTOR	1.275	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2191.8
DEVELOPMENT SUPPORT COST	M$	1095.0
FLIGHT TESTING COST	M$	569.3
DEVELOPMENT TOOLING COST	M$	1095.7
MANUFACTURING LABOUR	M$	1201.3
QUALITY CONTROL LABOUR	M$	182.0
MANUFACTURING MATERIAL	M$	101.6
ENGINE DEVELOPMENT COST	M$	3526.0
DEVELOPMENT ENGINE COST	M$	167.4
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16188.1

PRODUCTION		
No. PRODUCTION AIRCRAFT		620
AIFRAME ENGINEERING	M$	3612.9
PRODUCTION TOOLING	M$	1257.6
MANUFACTURING LABOUR	M$	11429.7
QUALITY CNTRL LBR COST	M$	1732.1
MANUFACTURING MATERIAL	M$	2973.7
PRODUCTION ENGINE COST	M$	7050.6
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	35008.0
G.S.E. & INITIAL SPARES	M$	3500.8
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.640
FUEL SYSTEM	0.154
HYD. & PNEU.	0.280
UNDERCARRIAGE	0.723
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.666
POWERPLANT SYST.	0.525
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.182
NAV. & COMMS	0.249
ELEC. & INSTR.	0.342
SCHEDULED MAINT.	4.996
TOTAL MAINTENANCE	9.940

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	748.8	
No. SUPP. PERSONNEL	412.0	
OPS. PERSONNEL COST	M$	32.104
SUPP. PERSONNEL COST	M$	17.011
SUPPORT & TRAINING	M$	5.985
UNIT LVL CONSUMPTION	M$	15.073
ENGINE CONTRACT COST	M$	29.153
MECH. CONTRACT COST	M$	18.175
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.100
SUSTAINING SUPPORT	M$	13.693
INSTALLATION SUPPORT	M$	4.630
1st YR BASE O&S COST	M$	154.530
TOTAL 25.YR O&S COST	M$	1975.407

DISPOSAL

DISPOSAL LABOUR COST	M$	571.5
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-49.1
TOTAL DISPOSAL COST	M$	522.4
DEFLATED DISPOSAL COST	M$	121.7

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.736
PROD.	M$	55.657
GSE&IS	M$	5.646
O & S	M$	50.651
DISP.	M$	0.194
TOTAL	M$	137.884

GENERAL PARAMETERS		
AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	2	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	34.52	

TOTAL STRUCTURE & SYSTEMS MASS	(kg)
TOTAL WING GROUP	1636.8
TOTAL FUSELAGE GROUP	1955.2
TOTAL EMPENNAGE GROUP	367.1
TOTAL UNDERCARRIAGE	893.6
ENGINE & INSTALLATION	2493.3
SYSTEMS	
FUEL SYSTEM & TANKS	532.7
POWER & FLYING CONTROLS	360.7
HYDRAULICS & PNEUMATICS	245.1
ELECTRICS & BATTERIES	281.5
AIR SYSTEM (AIRCON/ICE)	290.9
COCKPIT INSTRUMENTS ETC	458.9
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	249.3
AIRCRAFT EMPTY MASS	11470.2

WING PARAMETERS		
WING TYPE	DELTA WING	
GROSS AREA	54.657	m^2
TOTAL SPAN	10.828	m
MEAN AERO CHORD	5.967	m
LDG EDGE SWEEP	49.528	Deg
1/4 CHORD SWEEP	39.606	Deg
ASPECT RATIO	2.145	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0456	

SLS ENGINE PARAMETERS		
NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	86.514	kg/s
MAXIMUM THRUST	210.755	kN
MILITARY THRUST	140.963	kN
INLET DIAMETER	0.811	m
MAXIMUM DIAMETER	1.005	m
BARE ENGINE LENGTH	2.409	m
OVERALL LENGTH	4.311	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS		
OVERALL LENGTH	16.655	m
EFFECTIVE DIA.	2.130	m
MAXIMUM HEIGHT	1.206	m
MAXIMUM WIDTH	3.214	m
ENG INLET AREA	0.772	m^2
BASE DRAG AREA	1.396	m^2
CENTRELINE HT.	2.388	m

AERODYNAMIC PARAMETERS	
TAKE-OFF CL	1.036
LANDING CL	1.370
HIGH SPEED CL	1.528
BUFFET MAX CL	0.548
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS		
TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.476	m^2
TAIL MEAN AERO CHRD	1.201	m
TAIL ASPECT RATIO	1.716	
TAIL THK/CHRD RATIO	0.0411	
NUMBER OF FINS	2	
TOTAL FIN AREA	11.160	m^2
FIN MEAN AERO CHRD	3.951	m
FIN ASPECT RATIO	0.715	
FIN THCK/CHRD RATIO	0.0411	

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	229.60
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.01
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.02
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	51.79
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	75.88
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11470.2	kg	WING LOADING 319.24 kg/m^2
CREW INCLUDING EQPT	200.0	kg	THRUST/WEIGHT 0.916
TOTAL WEAPONS MASS	475.0	kg	FUEL FRACTION 0.226
INTERNAL FUEL MASS	5303.9	kg	XS FUEL VOL. 0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.126
AIRCRAFT ALL-UP MASS	17449.1	kg	
AIRCRAFT GROSS MASS	23474.1	kg	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							17449.1	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17254.8	0.9889
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17144.5	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	16653.3	0.9714
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	15726.8	0.9444
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	15726.8	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	15319.8	0.9741
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	15098.2	0.9855
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	14973.4	0.9917
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	14788.1	0.9876
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	13836.1	0.9356
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	13676.9	0.9885
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	13464.4	0.9845
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	13341.7	0.9909
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	13341.7	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	12495.4	0.9366
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	12495.4	1.0000
17 LANDING	0.0	0.17	0.00	0.00	0.0	84.8	12432.9	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	12238.6	0.9844

Aircraft Cost Data
Twin Engine, Canard Configuration. 2 Crew, 2 Fins. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	13718.0	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.279	
ADV. TECH. FACTOR	1.279	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2338.7
DEVELOPMENT SUPPORT COST	M$	1155.4
FLIGHT TESTING COST	M$	608.2
DEVELOPMENT TOOLING COST	M$	1163.2
MANUFACTURING LABOUR	M$	1274.8
QUALITY CONTROL LABOUR	M$	193.2
MANUFACTURING MATERIAL	M$	107.4
ENGINE DEVELOPMENT COST	M$	3527.7
DEVELOPMENT ENGINE COST	M$	169.8
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16596.3

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3855.0
PRODUCTION TOOLING	M$	1335.2
MANUFACTURING LABOUR	M$	12129.1
QUALITY CNTRL LBR COST	M$	1838.1
MANUFACTURING MATERIAL	M$	3143.8
PRODUCTION ENGINE COST	M$	7151.5
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	36404.0
G.S.E. & INITIAL SPARES	M$	3640.4
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.050
FLYING CONTROLS	0.645
FUEL SYSTEM	0.162
HYD. & PNEU.	0.316
UNDERCARRIAGE	0.732
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.664
POWERPLANT SYST.	0.547
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.242
NAV. & COMMS	0.256
ELEC. & INSTR.	0.359
SCHEDULED MAINT.	5.096
TOTAL MAINTENANCE	10.202

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	762.4	
No. SUPP. PERSONNEL	414.4	
OPS. PERSONNEL COST	M$	32.644
SUPP. PERSONNEL COST	M$	17.101
SUPPORT & TRAINING	M$	6.087
UNIT LVL CONSUMPTION	M$	15.202
ENGINE CONTRACT COST	M$	29.795
MECH. CONTRACT COST	M$	18.783
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.148
SUSTAINING SUPPORT	M$	14.209
INSTALLATION SUPPORT	M$	4.694
1st YR BASE O&S COST	M$	157.267
TOTAL 25.YR O&S COST	M$	2010.405

DISPOSAL

DISPOSAL LABOUR COST	M$	606.5
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-50.2
TOTAL DISPOSAL COST	M$	556.3
DEFLATED DISPOSAL COST	M$	129.6

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	26.385
PROD.	M$	57.876
GSE&IS	M$	5.872
O & S	M$	51.549
DISP.	M$	0.206
TOTAL	M$	141.888

Aircraft Configuration Data
Twin Engine, Canard Configuration. 1 Crew, 1 Fin. Internal Carriage

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	0.0	kg
MASS OF INT WEAPONS	600.0	kg
VOL. OF INT WEAPONS	2.90	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.32	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1605.3
TOTAL FUSELAGE GROUP	1759.5
TOTAL EMPENNAGE GROUP	185.5
TOTAL UNDERCARRIAGE	848.3
ENGINE & INSTALLATION SYSTEMS	2388.9
FUEL SYSTEM & TANKS	502.9
POWER & FLYING CONTROLS	345.0
HYDRAULICS & PNEUMATICS	238.4
ELECTRICS & BATTERIES	268.6
AIR SYSTEM (AIRCON/ICE)	221.4
COCKPIT INSTRUMENTS ETC	348.1
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	238.9
AIRCRAFT EMPTY MASS	10655.8

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	54.288	m^2
TOTAL SPAN	11.139	m
MEAN AERO CHORD	5.761	m
LDG EDGE SWEEP	53.986	Deg
1/4 CHORD SWEEP	46.459	Deg
ASPECT RATIO	2.286	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0645	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	83.580	kg/s
MAXIMUM THRUST	203.607	kN
MILITARY THRUST	136.182	kN
INLET DIAMETER	0.797	m
MAXIMUM DIAMETER	0.990	m
BARE ENGINE LENGTH	2.365	m
OVERALL LENGTH	4.220	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.810	m
EFFECTIVE DIA.	2.094	m
MAXIMUM HEIGHT	1.188	m
MAXIMUM WIDTH	3.153	m
ENG INLET AREA	0.746	m^2
BASE DRAG AREA	1.349	m^2
CENTRELINE HT.	2.289	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.975
LANDING CL	1.289
HIGH SPEED CL	1.476
BUFFET MAX CL	0.516
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.499	m^2
TAIL MEAN AERO CHRD	1.169	m
TAIL ASPECT RATIO	1.829	
TAIL THK/CHRD RATIO	0.0580	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.053	m^2
FIN MEAN AERO CHRD	2.575	m
FIN ASPECT RATIO	0.762	
FIN THCK/CHRD RATIO	0.0580	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							16298.7	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	16111.0	0.9885
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	16007.2	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.3	15560.6	0.9721
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.4	14726.6	0.9464
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.4	14726.6	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.4	14358.1	0.9750
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.0	14146.9	0.9853
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.0	14020.8	0.9911
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.4	13859.1	0.9885
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.4	12939.3	0.9336
11	INTRNL WEAPONS	8382.0	0.90	0.26	0.00	1.0	53.5	12780.3	0.9877
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.0	12576.7	0.9841
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	54.9	12451.7	0.9901
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	54.9	12451.7	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.0	11678.3	0.9379
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.4	11678.3	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.6	11619.9	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.6	11432.2	0.9838

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	219.40
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.83
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.23
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.71
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.32
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	43.34
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	66.66
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10655.8 kg	WING LOADING	300.23 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.935
TOTAL WEAPONS MASS	600.0 kg	FUEL FRACTION	0.223
INTERNAL FUEL MASS	4942.9 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.073
AIRCRAFT ALL-UP MASS	16298.7 kg		
AIRCRAFT GROSS MASS	22198.7 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12604.8	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.267	
ADV. TECH. FACTOR	1.277	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2162.2
DEVELOPMENT SUPPORT COST	M$	1068.3
FLIGHT TESTING COST	M$	549.9
DEVELOPMENT TOOLING COST	M$	1080.0
MANUFACTURING LABOUR	M$	1184.4
QUALITY CONTROL LABOUR	M$	179.5
MANUFACTURING MATERIAL	M$	100.4
ENGINE DEVELOPMENT COST	M$	3525.7
DEVELOPMENT ENGINE COST	M$	167.0
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16075.4

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3564.1
PRODUCTION TOOLING	M$	1239.6
MANUFACTURING LABOUR	M$	11269.0
QUALITY CNTRL LBR COST	M$	1707.8
MANUFACTURING MATERIAL	M$	2937.7
PRODUCTION ENGINE COST	M$	7033.8
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	34703.4
G.S.E. & INITIAL SPARES	M$	3470.3
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.583
FUEL SYSTEM	0.155
HYD. & PNEU.	0.290
UNDERCARRIAGE	0.717
OXYGEN SYSTEM	0.075
MISC. MECH	0.005
TOTAL STRUCTURE	0.641
POWERPLANT SYST.	0.527
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.199
NAV. & COMMS	0.248
ELEC. & INSTR.	0.344
SCHEDULED MAINT.	4.963
TOTAL MAINTENANCE	9.853

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	699.3	
No. SUPP. PERSONNEL	400.1	
OPS. PERSONNEL COST	M$	29.155
SUPP. PERSONNEL COST	M$	16.495
SUPPORT & TRAINING	M$	5.592
UNIT LVL CONSUMPTION	M$	15.022
ENGINE CONTRACT COST	M$	29.047
MECH. CONTRACT COST	M$	17.857
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.083
SUSTAINING SUPPORT	M$	13.577
INSTALLATION SUPPORT	M$	4.385
1st YR BASE O&S COST	M$	149.818
TOTAL 25.YR O&S COST	M$	1915.177

DISPOSAL		
DISPOSAL LABOUR COST	M$	563.5
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-47.0
TOTAL DISPOSAL COST	M$	516.5
DEFLATED DISPOSAL COST	M$	120.4

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.557
PROD.	M$	55.172
GSE&IS	M$	5.597
O & S	M$	49.107
DISP.	M$	0.191
TOTAL	M$	135.625

Aircraft Configuration Data

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. Mass-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	35.36	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1687.7
TOTAL FUSELAGE GROUP	1672.7
TOTAL EMPENNAGE GROUP	174.7
TOTAL UNDERCARRIAGE	802.8
ENGINE & INSTALLATION	1928.9
SYSTEMS	
FUEL SYSTEM & TANKS	469.1
POWER & FLYING CONTROLS	329.0
HYDRAULICS & PNEUMATICS	231.4
ELECTRICS & BATTERIES	255.5
AIR SYSTEM (AIRCON/ICE)	210.1
COCKPIT INSTRUMENTS ETC	337.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	192.9
AIRCRAFT EMPTY MASS	9996.8

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	46.824	m^2
TOTAL SPAN	10.489	m
MEAN AERO CHORD	5.277	m
LDG EDGE SWEEP	49.046	Deg
1/4 CHORD SWEEP	39.951	Deg
ASPECT RATIO	2.350	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0411	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	70.128	kg/s
MAXIMUM THRUST	170.837	kN
MILITARY THRUST	114.264	kN
INLET DIAMETER	0.730	m
MAXIMUM DIAMETER	0.917	m
BARE ENGINE LENGTH	2.156	m
OVERALL LENGTH	3.788	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.764	m
EFFECTIVE DIA.	1.918	m
MAXIMUM HEIGHT	1.100	m
MAXIMUM WIDTH	2.863	m
ENG INLET AREA	0.626	m^2
BASE DRAG AREA	1.132	m^2
CENTRELINE HT.	2.347	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.033
LANDING CL	1.367
HIGH SPEED CL	1.556
BUFFET MAX CL	0.547
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.189	m^2
TAIL MEAN AERO CHRD	1.079	m
TAIL ASPECT RATIO	1.880	
TAIL THK/CHRD RATIO	0.0370	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.952	m^2
FIN MEAN AERO CHRD	2.514	m
FIN ASPECT RATIO	0.783	
FIN THCK/CHRD RATIO	0.0370	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							14895.8	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	14738.3	0.9894
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	14645.2	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.7	14224.0	0.9712
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.8	13469.0	0.9469
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.8	13469.0	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.8	13138.7	0.9755
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	12955.3	0.9860
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	12862.4	0.9928
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	12719.1	0.9889
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	11947.4	0.9393
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	11788.2	0.9867
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.4	11611.5	0.9850
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.4	11520.5	0.9922
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.4	11520.5	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.5	10830.2	0.9401
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.8	10830.2	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	85.1	10776.1	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.1	10618.6	0.9854

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	246.45
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.13
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.11
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.25
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	46.98
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	75.79
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	9996.8 kg	WING LOADING	318.13 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.833
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.207
INTERNAL FUEL MASS	4324.0 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.238
AIRCRAFT ALL-UP MASS	14895.8 kg		
AIRCRAFT GROSS MASS	20920.8 kg		

Aircraft Cost Data
Twin Engine, Canard Configuration. 1 Crew, 1 Fin. Mass-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12410.5	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.287	
ADV. TECH. FACTOR	1.281	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2176.3
DEVELOPMENT SUPPORT COST	M$	1056.8
FLIGHT TESTING COST	M$	539.8
DEVELOPMENT TOOLING COST	M$	1084.1
MANUFACTURING LABOUR	M$	1189.0
QUALITY CONTROL LABOUR	M$	180.2
MANUFACTURING MATERIAL	M$	100.9
ENGINE DEVELOPMENT COST	M$	3516.6
DEVELOPMENT ENGINE COST	M$	153.5
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16055.3

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3587.5
PRODUCTION TOOLING	M$	1244.3
MANUFACTURING LABOUR	M$	11313.0
QUALITY CNTRL LBR COST	M$	1714.4
MANUFACTURING MATERIAL	M$	2952.3
PRODUCTION ENGINE COST	M$	6464.4
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	34227.2
G.S.E. & INITIAL SPARES	M$	3422.7
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.047
FLYING CONTROLS	0.552
FUEL SYSTEM	0.147
HYD. & PNEU.	0.322
UNDERCARRIAGE	0.706
OXYGEN SYSTEM	0.074
MISC. MECH	0.005
TOTAL STRUCTURE	0.593
POWERPLANT SYST.	0.490
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.273
NAV. & COMMS	0.217
ELEC. & INSTR.	0.318
SCHEDULED MAINT.	4.704
TOTAL MAINTENANCE	9.506

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	684.0	
No. SUPP. PERSONNEL	397.4	
OPS. PERSONNEL COST	M$	28.547
SUPP. PERSONNEL COST	M$	16.394
SUPPORT & TRAINING	M$	5.476
UNIT LVL CONSUMPTION	M$	14.678
ENGINE CONTRACT COST	M$	25.620
MECH. CONTRACT COST	M$	16.818
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.914
SUSTAINING SUPPORT	M$	13.302
INSTALLATION SUPPORT	M$	4.313
1st YR BASE O&S COST	M$	143.668
TOTAL 25.YR O&S COST	M$	1836.561

DISPOSAL

DISPOSAL LABOUR COST	M$	565.6
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-44.8
TOTAL DISPOSAL COST	M$	520.9
DEFLATED DISPOSAL COST	M$	121.4

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.525
PROD.	M$	54.415
GSE&IS	M$	5.521
O & S	M$	47.091
DISP.	M$	0.193
TOTAL	M$	132.745

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	0.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2380.4
TOTAL FUSELAGE GROUP	2390.2
TOTAL EMPENNAGE GROUP	273.0
TOTAL UNDERCARRIAGE	917.7
ENGINE & INSTALLATION SYSTEMS	2421.8
FUEL SYSTEM & TANKS	543.7
POWER & FLYING CONTROLS	369.1
HYDRAULICS & PNEUMATICS	248.7
ELECTRICS & BATTERIES	288.3
AIR SYSTEM (AIRCON/ICE)	238.5
COCKPIT INSTRUMENTS ETC	361.3
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	242.2
AIRCRAFT EMPTY MASS	12379.9

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	56.801	m^2
TOTAL SPAN	11.462	m
MEAN AERO CHORD	5.858	m
LDG EDGE SWEEP	48.841	Deg
1/4 CHORD SWEEP	39.501	Deg
ASPECT RATIO	2.313	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0357	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	84.508	kg/s
MAXIMUM THRUST	205.868	kN
MILITARY THRUST	137.695	kN
INLET DIAMETER	0.801	m
MAXIMUM DIAMETER	0.995	m
BARE ENGINE LENGTH	2.379	m
OVERALL LENGTH	4.249	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.336	m
EFFECTIVE DIA.	2.105	m
MAXIMUM HEIGHT	1.194	m
MAXIMUM WIDTH	3.172	m
ENG INLET AREA	0.754	m^2
BASE DRAG AREA	1.364	m^2
CENTRELINE HT.	2.348	m

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.565	m^2
TAIL MEAN AERO CHRD	1.177	m
TAIL ASPECT RATIO	1.850	
TAIL THK/CHRD RATIO	0.0321	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.190	m^2
FIN MEAN AERO CHRD	2.594	m
FIN ASPECT RATIO	0.771	
FIN THCK/CHRD RATIO	0.0321	

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.037
LANDING CL	1.372
HIGH SPEED CL	1.555
BUFFET MAX CL	0.549
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							18131.8	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17942.1	0.9895
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17829.0	0.9937
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.7	17324.8	0.9717
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.8	16433.1	0.9485
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.8	16433.1	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.8	16039.0	0.9760
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.4	15817.0	0.9862
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.4	15706.8	0.9930
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.8	15531.3	0.9888
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.8	14601.3	0.9401
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.9	14442.2	0.9891
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.4	14227.0	0.9851
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.4	14118.8	0.9924
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.4	14118.8	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.5	13298.5	0.9419
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.8	13298.5	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	85.0	13232.0	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	86.0	13042.3	0.9857

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	249.91
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.07
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.15
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	56.30
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	93.77
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	12379.9	kg	WING LOADING 319.22 kg/m^2
CREW INCLUDING EQPT	100.0	kg	THRUST/WEIGHT 0.869
TOTAL WEAPONS MASS	475.0	kg	FUEL FRACTION 0.214
INTERNAL FUEL MASS	5176.9	kg	XS FUEL VOL. 0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.259
AIRCRAFT ALL-UP MASS	18131.8	kg	
AIRCRAFT GROSS MASS	24156.8	kg	

Aircraft Cost Data - 0% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	16132.5	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.000	
ADV. TECH. FACTOR	1.210	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	1968.5
DEVELOPMENT SUPPORT COST	M$	1265.3
FLIGHT TESTING COST	M$	737.6
DEVELOPMENT TOOLING COST	M$	1030.3
MANUFACTURING LABOUR	M$	1127.6
QUALITY CONTROL LABOUR	M$	170.9
MANUFACTURING MATERIAL	M$	94.0
ENGINE DEVELOPMENT COST	M$	3526.3
DEVELOPMENT ENGINE COST	M$	167.9
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	16146.3

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3244.8
PRODUCTION TOOLING	M$	1182.5
MANUFACTURING LABOUR	M$	10728.7
QUALITY CNTRL LBR COST	M$	1625.9
MANUFACTURING MATERIAL	M$	2750.3
PRODUCTION ENGINE COST	M$	7071.3
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33554.9
G.S.E. & INITIAL SPARES	M$	3355.5
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.050
FLYING CONTROLS	0.585
FUEL SYSTEM	0.158
HYD. & PNEU.	0.306
UNDERCARRIAGE	0.748
OXYGEN SYSTEM	0.073
MISC. MECH	0.005
TOTAL STRUCTURE	1.073
POWERPLANT SYST.	0.536
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.227
NAV. & COMMS	0.251
ELEC. & INSTR.	0.351
SCHEDULED MAINT.	4.730
TOTAL MAINTENANCE	10.152

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	722.2	
No. SUPP. PERSONNEL	404.0	
OPS. PERSONNEL COST	M$	30.066
SUPP. PERSONNEL COST	M$	16.647
SUPPORT & TRAINING	M$	5.758
UNIT LVL CONSUMPTION	M$	15.166
ENGINE CONTRACT COST	M$	29.284
MECH. CONTRACT COST	M$	19.042
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.138
SUSTAINING SUPPORT	M$	13.216
INSTALLATION SUPPORT	M$	4.492
1st YR BASE O&S COST	M$	152.413
TOTAL 25.YR O&S COST	M$	1948.352

DISPOSAL

DISPOSAL LABOUR COST	M$	536.4
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-90.2
TOTAL DISPOSAL COST	M$	446.3
DEFLATED DISPOSAL COST	M$	104.0

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.670
PROD.	M$	53.346
GSE&IS	M$	5.412
O & S	M$	49.958
DISP.	M$	0.165
TOTAL	M$	134.551

Aircraft Configuration Data - 5% Advanced Materials
Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	5.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2019.1
TOTAL FUSELAGE GROUP	2299.0
TOTAL EMPENNAGE GROUP	187.1
TOTAL UNDERCARRIAGE	895.2
ENGINE & INSTALLATION	2418.6
SYSTEMS	
FUEL SYSTEM & TANKS	531.1
POWER & FLYING CONTROLS	361.3
HYDRAULICS & PNEUMATICS	245.4
ELECTRICS & BATTERIES	281.9
AIR SYSTEM (AIRCON/ICE)	233.0
COCKPIT INSTRUMENTS ETC	356.3
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	241.9
AIRCRAFT EMPTY MASS	11774.8

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	54.812	m^2
TOTAL SPAN	11.009	m
MEAN AERO CHORD	5.886	m
LDG EDGE SWEEP	49.258	Deg
1/4 CHORD SWEEP	39.578	Deg
ASPECT RATIO	2.211	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0415	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	84.419	kg/s
MAXIMUM THRUST	205.651	kN
MILITARY THRUST	137.549	kN
INLET DIAMETER	0.801	m
MAXIMUM DIAMETER	0.994	m
BARE ENGINE LENGTH	2.378	m
OVERALL LENGTH	4.246	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.927	m
EFFECTIVE DIA.	2.104	m
MAXIMUM HEIGHT	1.193	m
MAXIMUM WIDTH	3.171	m
ENG INLET AREA	0.753	m^2
BASE DRAG AREA	1.362	m^2
CENTRELINE HT.	2.304	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.036
LANDING CL	1.371
HIGH SPEED CL	1.539
BUFFET MAX CL	0.548
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.512	m^2
TAIL MEAN AERO CHRD	1.192	m
TAIL ASPECT RATIO	1.769	
TAIL THK/CHRD RATIO	0.0374	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.097	m^2
FIN MEAN AERO CHRD	2.630	m
FIN ASPECT RATIO	0.737	
FIN THCK/CHRD RATIO	0.0374	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							17493.6	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17304.1	0.9892
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17194.2	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	16706.9	0.9717
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	15819.9	0.9469
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	15819.9	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	15428.8	0.9753
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	15210.1	0.9858
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.3	15093.1	0.9923
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	14914.4	0.9882
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	13985.4	0.9377
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	13826.3	0.9886
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	13615.5	0.9848
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	13500.5	0.9916
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	13500.5	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	12688.6	0.9399
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	12688.6	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	84.9	12625.2	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	12435.6	0.9850

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	238.30
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.03
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.07
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	52.30
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	82.52
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11774.8 kg	WING LOADING	319.16 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.892
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.219
INTERNAL FUEL MASS	5143.9 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.176
AIRCRAFT ALL-UP MASS	17493.6 kg		
AIRCRAFT GROSS MASS	23518.6 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	14869.1	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.022	
ADV. TECH. FACTOR	1.220	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	1901.5
DEVELOPMENT SUPPORT COST	M$	1185.2
FLIGHT TESTING COST	M$	669.4
DEVELOPMENT TOOLING COST	M$	989.3
MANUFACTURING LABOUR	M$	1083.5
QUALITY CONTROL LABOUR	M$	164.2
MANUFACTURING MATERIAL	M$	90.8
ENGINE DEVELOPMENT COST	M$	3526.2
DEVELOPMENT ENGINE COST	M$	167.8
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15836.0

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3134.4
PRODUCTION TOOLING	M$	1135.5
MANUFACTURING LABOUR	M$	10309.0
QUALITY CNTRL LBR COST	M$	1562.3
MANUFACTURING MATERIAL	M$	2657.4
PRODUCTION ENGINE COST	M$	7067.7
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	32817.6
G.S.E. & INITIAL SPARES	M$	3281.8
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.049
FLYING CONTROLS	0.585
FUEL SYSTEM	0.156
HYD. & PNEU.	0.293
UNDERCARRIAGE	0.737
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.982
POWERPLANT SYST.	0.531
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.204
NAV. & COMMS	0.250
ELEC. & INSTR.	0.347
SCHEDULED MAINT.	4.739
TOTAL MAINTENANCE	10.010

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	713.5	
No. SUPP. PERSONNEL	402.5	
OPS. PERSONNEL COST	M$	29.720
SUPP. PERSONNEL COST	M$	16.589
SUPPORT & TRAINING	M$	5.694
UNIT LVL CONSUMPTION	M$	15.115
ENGINE CONTRACT COST	M$	29.261
MECH. CONTRACT COST	M$	18.587
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.120
SUSTAINING SUPPORT	M$	12.949
INSTALLATION SUPPORT	M$	4.451
1st YR BASE O&S COST	M$	151.090
TOTAL 25.YR O&S COST	M$	1931.431

DISPOSAL		
DISPOSAL LABOUR COST	M$	515.4
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-78.9
TOTAL DISPOSAL COST	M$	436.5
DEFLATED DISPOSAL COST	M$	101.7

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.176
PROD.	M$	52.174
GSE&IS	M$	5.293
O & S	M$	49.524
DISP.	M$	0.162
TOTAL	M$	132.329

Aircraft Configuration Data - 10% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	10.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	2053.1
TOTAL FUSELAGE GROUP	2119.7
TOTAL EMPENNAGE GROUP	186.1
TOTAL UNDERCARRIAGE	892.1
ENGINE & INSTALLATION SYSTEMS	2447.6
FUEL SYSTEM & TANKS	530.0
POWER & FLYING CONTROLS	360.2
HYDRAULICS & PNEUMATICS	244.9
ELECTRICS & BATTERIES	281.1
AIR SYSTEM (AIRCON/ICE)	232.2
COCKPIT INSTRUMENTS ETC	357.3
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	244.8
AIRCRAFT EMPTY MASS	11654.1

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	58.281	m^2
TOTAL SPAN	11.034	m
MEAN AERO CHORD	6.244	m
LDG EDGE SWEEP	54.928	Deg
1/4 CHORD SWEEP	46.950	Deg
ASPECT RATIO	2.089	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0457	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	85.234	kg/s
MAXIMUM THRUST	207.637	kN
MILITARY THRUST	138.878	kN
INLET DIAMETER	0.805	m
MAXIMUM DIAMETER	0.999	m
BARE ENGINE LENGTH	2.390	m
OVERALL LENGTH	4.271	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.011	m
EFFECTIVE DIA.	2.114	m
MAXIMUM HEIGHT	1.198	m
MAXIMUM WIDTH	3.187	m
ENG INLET AREA	0.760	m^2
BASE DRAG AREA	1.376	m^2
CENTRELINE HT.	2.208	m

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.680	m^2
TAIL MEAN AERO CHRD	1.266	m
TAIL ASPECT RATIO	1.671	
TAIL THK/CHRD RATIO	0.0411	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.907	m^2
FIN MEAN AERO CHRD	2.655	m
FIN ASPECT RATIO	0.696	
FIN THCK/CHRD RATIO	0.0411	

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.970
LANDING CL	1.283
HIGH SPEED CL	1.443
BUFFET MAX CL	0.513
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							17407.8	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	17216.4	0.9890
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	17106.7	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	16622.9	0.9717
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	15726.0	0.9460
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	15726.0	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	15333.2	0.9750
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	15114.2	0.9857
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	14991.9	0.9919
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	14817.8	0.9884
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	13879.8	0.9367
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	13720.7	0.9885
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	13510.1	0.9847
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	13389.7	0.9911
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	13389.7	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	12570.8	0.9388
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	12570.8	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	84.8	12507.9	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	12316.5	0.9847

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	233.62
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.63
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.06
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.08
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	45.58
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	79.37
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11654.1 kg	WING LOADING	298.69 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.904
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.221
INTERNAL FUEL MASS	5178.8 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.132
AIRCRAFT ALL-UP MASS	17407.8 kg		
AIRCRAFT GROSS MASS	23432.8 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES

AMPR (AIRFRAME) MASS	14537.5	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.055	
ADV. TECH. FACTOR	1.230	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT

No. DEVELOPMENT AIRCRAFT		9
No. STATIC AIRFRAMES		2
No. DEVELOPMENT ENGINES		21.
AIRFRAME ENGINEERING COST	M$	1943.3
DEVELOPMENT SUPPORT COST	M$	1170.8
FLIGHT TESTING COST	M$	651.6
DEVELOPMENT TOOLING COST	M$	1003.5
MANUFACTURING LABOUR	M$	1099.2
QUALITY CONTROL LABOUR	M$	166.6
MANUFACTURING MATERIAL	M$	92.2
ENGINE DEVELOPMENT COST	M$	3526.8
DEVELOPMENT ENGINE COST	M$	168.6
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15880.7

PRODUCTION

No. PRODUCTION AIRCRAFT		620
AIFRAME ENGINEERING	M$	3203.3
PRODUCTION TOOLING	M$	1151.8
MANUFACTURING LABOUR	M$	10458.5
QUALITY CNTRL LBR COST	M$	1584.9
MANUFACTURING MATERIAL	M$	2700.1
PRODUCTION ENGINE COST	M$	7100.4
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33150.4
G.S.E. & INITIAL SPARES	M$	3315.0
ENGINEERING LABOUR RATE		118.5 $/hr
TOOLING/QUALITY LABOUR RT		86.8 $/hr
MANUFACTURING LABOUR RATE		74.5 $/hr

RELIABILITY AND MAINTAINABILITY

	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.587
FUEL SYSTEM	0.153
HYD. & PNEU.	0.264
UNDERCARRIAGE	0.735
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.923
POWERPLANT SYST.	0.525
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.152
NAV. & COMMS	0.252
ELEC. & INSTR.	0.341
SCHEDULED MAINT.	4.805
TOTAL MAINTENANCE	9.919

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE		39.
AVG. YRLY FLIGHT TIME		300.0
No. OPS. PERSONNEL		706.6
No. SUPP. PERSONNEL		401.3
OPS. PERSONNEL COST	M$	29.446
SUPP. PERSONNEL COST	M$	16.544
SUPPORT & TRAINING	M$	5.644
UNIT LVL CONSUMPTION	M$	15.106
ENGINE CONTRACT COST	M$	29.469
MECH. CONTRACT COST	M$	18.444
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.122
SUSTAINING SUPPORT	M$	13.064
INSTALLATION SUPPORT	M$	4.419
1st YR BASE O&S COST	M$	150.862
TOTAL 25.YR O&S COST	M$	1928.521

DISPOSAL

DISPOSAL LABOUR COST	M$	522.9
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-73.1
TOTAL DISPOSAL COST	M$	449.8
DEFLATED DISPOSAL COST	M$	104.8

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.248
PROD.	M$	52.703
GSE&IS	M$	5.347
O & S	M$	49.449
DISP.	M$	0.167
TOTAL	M$	132.914

Aircraft Configuration Data - 15% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	15.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1922.5
TOTAL FUSELAGE GROUP	2091.9
TOTAL EMPENNAGE GROUP	183.8
TOTAL UNDERCARRIAGE	876.8
ENGINE & INSTALLATION	2364.8
SYSTEMS	
FUEL SYSTEM & TANKS	519.7
POWER & FLYING CONTROLS	354.9
HYDRAULICS & PNEUMATICS	242.7
ELECTRICS & BATTERIES	276.7
AIR SYSTEM (AIRCON/ICE)	228.4
COCKPIT INSTRUMENTS ETC	354.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	236.5
AIRCRAFT EMPTY MASS	11357.5

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	56.805	m^2
TOTAL SPAN	10.961	m
MEAN AERO CHORD	6.126	m
LDG EDGE SWEEP	54.810	Deg
1/4 CHORD SWEEP	46.901	Deg
ASPECT RATIO	2.115	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0462	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	82.895	kg/s
MAXIMUM THRUST	201.938	kN
MILITARY THRUST	135.066	kN
INLET DIAMETER	0.793	m
MAXIMUM DIAMETER	0.986	m
BARE ENGINE LENGTH	2.355	m
OVERALL LENGTH	4.199	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.100	m
EFFECTIVE DIA.	2.085	m
MAXIMUM HEIGHT	1.184	m
MAXIMUM WIDTH	3.139	m
ENG INLET AREA	0.740	m^2
BASE DRAG AREA	1.338	m^2
CENTRELINE HT.	2.210	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.970
LANDING CL	1.283
HIGH SPEED CL	1.447
BUFFET MAX CL	0.513
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.622	m^2
TAIL MEAN AERO CHRD	1.245	m
TAIL ASPECT RATIO	1.692	
TAIL THK/CHRD RATIO	0.0416	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.880	m^2
FIN MEAN AERO CHRD	2.631	m
FIN ASPECT RATIO	0.705	
FIN THCK/CHRD RATIO	0.0416	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							16975.0	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	16788.8	0.9890
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	16681.9	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	16209.3	0.9717
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	15334.6	0.9460
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	15334.6	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	14952.0	0.9750
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	14738.8	0.9857
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	14620.5	0.9920
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	14451.9	0.9885
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	13539.6	0.9369
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	13380.5	0.9882
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	13175.5	0.9847
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	13059.1	0.9912
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	13059.1	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	12261.0	0.9389
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	12261.0	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.8	12199.7	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	12013.5	0.9847

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	234.47
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.65
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.08
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.03
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.11
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	44.97
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	78.06
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	11357.5 kg	WING LOADING	298.83 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.895
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.219
INTERNAL FUEL MASS	5042.4 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.139
AIRCRAFT ALL-UP MASS	16975.0 kg		
AIRCRAFT GROSS MASS	23000.0 kg		

Aircraft Cost Data - 15% Advanced Materials
Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	14130.2	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.095	
ADV. TECH. FACTOR	1.240	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	1987.7
DEVELOPMENT SUPPORT COST	M$	1150.4
FLIGHT TESTING COST	M$	630.0
DEVELOPMENT TOOLING COST	M$	1019.0
MANUFACTURING LABOUR	M$	1116.4
QUALITY CONTROL LABOUR	M$	169.2
MANUFACTURING MATERIAL	M$	93.9
ENGINE DEVELOPMENT COST	M$	3525.2
DEVELOPMENT ENGINE COST	M$	166.4
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15916.2

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3276.6
PRODUCTION TOOLING	M$	1169.6
MANUFACTURING LABOUR	M$	10622.2
QUALITY CNTRL LBR COST	M$	1609.7
MANUFACTURING MATERIAL	M$	2747.7
PRODUCTION ENGINE COST	M$	7006.0
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33383.1
G.S.E. & INITIAL SPARES	M$	3338.3
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.582
FUEL SYSTEM	0.151
HYD. & PNEU.	0.267
UNDERCARRIAGE	0.730
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.855
POWERPLANT SYST.	0.517
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.161
NAV. & COMMS	0.246
ELEC. & INSTR.	0.336
SCHEDULED MAINT.	4.793
TOTAL MAINTENANCE	9.820

OPERATION & SUPPORT PER BASE

No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	701.0	
No. SUPP. PERSONNEL	400.3	
OPS. PERSONNEL COST	M$	29.221
SUPP. PERSONNEL COST	M$	16.506
SUPPORT & TRAINING	M$	5.602
UNIT LVL CONSUMPTION	M$	15.030
ENGINE CONTRACT COST	M$	28.873
MECH. CONTRACT COST	M$	18.132
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.087
SUSTAINING SUPPORT	M$	13.122
INSTALLATION SUPPORT	M$	4.393
1st YR BASE O&S COST	M$	149.572
TOTAL 25.YR O&S COST	M$	1912.027

DISPOSAL

DISPOSAL LABOUR COST	M$	531.1
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-67.1
TOTAL DISPOSAL COST	M$	464.0
DEFLATED DISPOSAL COST	M$	108.1

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.304
PROD.	M$	53.073
GSE&IS	M$	5.384
O & S	M$	49.026
DISP.	M$	0.172
TOTAL	M$	132.960

Aircraft Configuration Data - 20% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	20.00	

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	50.931	m^2
TOTAL SPAN	10.590	m
MEAN AERO CHORD	5.685	m
LDG EDGE SWEEP	49.375	Deg
1/4 CHORD SWEEP	39.694	Deg
ASPECT RATIO	2.202	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0438	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.107	m
EFFECTIVE DIA.	2.032	m
MAXIMUM HEIGHT	1.157	m
MAXIMUM WIDTH	3.051	m
ENG INLET AREA	0.702	m^2
BASE DRAG AREA	1.270	m^2
CENTRELINE HT.	2.305	m

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.365	m^2
TAIL MEAN AERO CHRD	1.159	m
TAIL ASPECT RATIO	1.762	
TAIL THK/CHRD RATIO	0.0394	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.014	m^2
FIN MEAN AERO CHRD	2.613	m
FIN ASPECT RATIO	0.734	
FIN THCK/CHRD RATIO	0.0394	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1639.8
TOTAL FUSELAGE GROUP	2113.7
TOTAL EMPENNAGE GROUP	180.9
TOTAL UNDERCARRIAGE	850.8
ENGINE & INSTALLATION	2219.2
SYSTEMS	
FUEL SYSTEM & TANKS	502.2
POWER & FLYING CONTROLS	345.8
HYDRAULICS & PNEUMATICS	238.8
ELECTRICS & BATTERIES	269.3
AIR SYSTEM (AIRCON/ICE)	222.0
COCKPIT INSTRUMENTS ETC	346.9
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	221.9
AIRCRAFT EMPTY MASS	10856.2

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	78.718	kg/s
MAXIMUM THRUST	191.764	kN
MILITARY THRUST	128.261	kN
INLET DIAMETER	0.773	m
MAXIMUM DIAMETER	0.964	m
BARE ENGINE LENGTH	2.292	m
OVERALL LENGTH	4.067	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.036
LANDING CL	1.369
HIGH SPEED CL	1.536
BUFFET MAX CL	0.548
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							16241.3	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	16064.5	0.9891
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15962.4	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	15508.8	0.9716
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	14674.3	0.9462
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	14674.3	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	14307.5	0.9750
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	14104.0	0.9858
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13994.1	0.9922
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	13827.5	0.9881
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	12961.2	0.9374
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	12802.1	0.9877
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	12606.5	0.9847
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	12498.5	0.9914
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	12498.5	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11736.6	0.9390
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	11736.6	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.9	11677.9	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	11501.1	0.9849

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	236.87
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.03
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.08
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	48.97
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	75.72
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10856.2 kg	WING LOADING	318.89 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.878
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.216
INTERNAL FUEL MASS	4810.1 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.166
AIRCRAFT ALL-UP MASS	16241.3 kg		
AIRCRAFT GROSS MASS	22266.3 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			
AMPR (AIRFRAME) MASS	13457.8	lb.	
MAX. TRUE AIRSPEED	1258.0	kt.	
UNINST. AVION. MASS	1597.2	lb.	
NUMBER OF PARTNERS	2.672		
COLLABORATION FACTOR	1.383		
ADV. MATERIAL FACTOR	1.139		
ADV. TECH. FACTOR	1.250		
ADV. DESIGN TOOLS	1.300		
ADV. TECH TESTING	1.200		
LOW OBSERVABLES	1.200		
SECURITY FACTOR	1.200		
FIRST FLIGHT YEAR	1996.		
FINANCIAL YEAR	2000.		
CONSUMER PRICE INDEX	1.777		
DISCOUNTING RATE	6.00%		

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2005.7
DEVELOPMENT SUPPORT COST	M$	1109.8
FLIGHT TESTING COST	M$	594.5
DEVELOPMENT TOOLING COST	M$	1021.3
MANUFACTURING LABOUR	M$	1119.4
QUALITY CONTROL LABOUR	M$	169.6
MANUFACTURING MATERIAL	M$	94.4
ENGINE DEVELOPMENT COST	M$	3522.4
DEVELOPMENT ENGINE COST	M$	162.3
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15857.5

PRODUCTION			
No. PRODUCTION AIRCRAFT		620	
AIFRAME ENGINEERING	M$	3306.2	
PRODUCTION TOOLING	M$	1172.3	
MANUFACTURING LABOUR	M$	10651.1	
QUALITY CNTRL LBR COST	M$	1614.1	
MANUFACTURING MATERIAL	M$	2764.3	
PRODUCTION ENGINE COST	M$	6833.9	
AVIONICS PURCHASE COST	M$	6951.3	
TOTAL PRODUCTION COST	M$	33293.3	
G.S.E. & INITIAL SPARES	M$	3329.3	
ENGINEERING LABOUR RATE		118.5	$/hr
TOOLING/QUALITY LABOUR RT		86.8	$/hr
MANUFACTURING LABOUR RATE		74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.572
FUEL SYSTEM	0.152
HYD. & PNEU.	0.300
UNDERCARRIAGE	0.721
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.779
POWERPLANT SYST.	0.513
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.223
NAV. & COMMS	0.237
ELEC. & INSTR.	0.334
SCHEDULED MAINT.	4.725
TOTAL MAINTENANCE	9.739

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	697.5	
No. SUPP. PERSONNEL	399.8	
OPS. PERSONNEL COST	M$	29.085
SUPP. PERSONNEL COST	M$	16.483
SUPPORT & TRAINING	M$	5.576
UNIT LVL CONSUMPTION	M$	14.919
ENGINE CONTRACT COST	M$	27.809
MECH. CONTRACT COST	M$	17.702
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.031
SUSTAINING SUPPORT	M$	13.055
INSTALLATION SUPPORT	M$	4.377
1st YR BASE O&S COST	M$	147.641
TOTAL 25.YR O&S COST	M$	1887.346

DISPOSAL		
DISPOSAL LABOUR COST	M$	532.6
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-60.2
TOTAL DISPOSAL COST	M$	472.4
DEFLATED DISPOSAL COST	M$	110.1

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.211
PROD.	M$	52.930
GSE&IS	M$	5.370
O & S	M$	48.393
DISP.	M$	0.175
TOTAL	M$	132.080

Aircraft Configuration Data - 25% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	25.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1588.4
TOTAL FUSELAGE GROUP	1901.4
TOTAL EMPENNAGE GROUP	178.0
TOTAL UNDERCARRIAGE	835.3
ENGINE & INSTALLATION	2186.6
SYSTEMS	
FUEL SYSTEM & TANKS	492.5
POWER & FLYING CONTROLS	340.4
HYDRAULICS & PNEUMATICS	236.4
ELECTRICS & BATTERIES	264.8
AIR SYSTEM (AIRCON/ICE)	218.2
COCKPIT INSTRUMENTS ETC	343.6
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	218.7
AIRCRAFT EMPTY MASS	10509.3

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	49.688	m^2
TOTAL SPAN	10.555	m
MEAN AERO CHORD	5.565	m
LDG EDGE SWEEP	49.457	Deg
1/4 CHORD SWEEP	40.009	Deg
ASPECT RATIO	2.242	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0467	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	77.773	kg/s
MAXIMUM THRUST	189.459	kN
MILITARY THRUST	126.720	kN
INLET DIAMETER	0.769	m
MAXIMUM DIAMETER	0.959	m
BARE ENGINE LENGTH	2.277	m
OVERALL LENGTH	4.037	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.875	m
EFFECTIVE DIA.	2.020	m
MAXIMUM HEIGHT	1.151	m
MAXIMUM WIDTH	3.031	m
ENG INLET AREA	0.694	m^2
BASE DRAG AREA	1.255	m^2
CENTRELINE HT.	2.277	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.033
LANDING CL	1.366
HIGH SPEED CL	1.539
BUFFET MAX CL	0.546
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.329	m^2
TAIL MEAN AERO CHRD	1.139	m
TAIL ASPECT RATIO	1.794	
TAIL THK/CHRD RATIO	0.0420	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.942	m^2
FIN MEAN AERO CHRD	2.572	m
FIN ASPECT RATIO	0.747	
FIN THCK/CHRD RATIO	0.0420	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							15806.4	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15631.7	0.9890
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15531.9	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	15090.0	0.9715
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14272.4	0.9458
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14272.4	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13914.4	0.9749
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13714.8	0.9857
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13604.1	0.9919
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13441.8	0.9881
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12585.9	0.9363
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12426.8	0.9874
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12235.2	0.9846
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	12126.5	0.9911
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	12126.5	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11381.6	0.9386
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11381.6	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	84.8	11324.7	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	11150.0	0.9846

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	232.06
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.09
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.17
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.14
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	47.17
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	71.32
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10509.3 kg	WING LOADING	318.11 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.885
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.216
INTERNAL FUEL MASS	4722.1 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.143
AIRCRAFT ALL-UP MASS	15806.4 kg		
AIRCRAFT GROSS MASS	21831.4 kg		

Aircraft Cost Data - 25% Advanced Materials
Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	12816.6	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.186	
ADV. TECH. FACTOR	1.260	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2024.9
DEVELOPMENT SUPPORT COST	M$	1070.4
FLIGHT TESTING COST	M$	560.9
DEVELOPMENT TOOLING COST	M$	1024.3
MANUFACTURING LABOUR	M$	1123.1
QUALITY CONTROL LABOUR	M$	170.2
MANUFACTURING MATERIAL	M$	95.1
ENGINE DEVELOPMENT COST	M$	3521.8
DEVELOPMENT ENGINE COST	M$	161.3
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15810.1

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3337.8
PRODUCTION TOOLING	M$	1175.7
MANUFACTURING LABOUR	M$	10686.5
QUALITY CNTRL LBR COST	M$	1619.5
MANUFACTURING MATERIAL	M$	2782.7
PRODUCTION ENGINE COST	M$	6794.3
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33347.9
G.S.E. & INITIAL SPARES	M$	3334.8
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.570
FUEL SYSTEM	0.150
HYD. & PNEU.	0.293
UNDERCARRIAGE	0.715
OXYGEN SYSTEM	0.073
MISC. MECH	0.005
TOTAL STRUCTURE	0.713
POWERPLANT SYST.	0.507
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.212
NAV. & COMMS	0.235
ELEC. & INSTR.	0.329
SCHEDULED MAINT.	4.745
TOTAL MAINTENANCE	9.652

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	691.9	
No. SUPP. PERSONNEL	398.8	
OPS. PERSONNEL COST	M$	28.860
SUPP. PERSONNEL COST	M$	16.446
SUPPORT & TRAINING	M$	5.535
UNIT LVL CONSUMPTION	M$	14.868
ENGINE CONTRACT COST	M$	27.568
MECH. CONTRACT COST	M$	17.403
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	2.010
SUSTAINING SUPPORT	M$	13.060
INSTALLATION SUPPORT	M$	4.350
1st YR BASE O&S COST	M$	146.705
TOTAL 25.YR O&S COST	M$	1875.379

DISPOSAL		
DISPOSAL LABOUR COST	M$	534.3
ADVANCED MATERIAL DISP.	M$	0.0
METALS SCRAP VALUE	M$	-53.7
TOTAL DISPOSAL COST	M$	480.6
DEFLATED DISPOSAL COST	M$	112.0

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT

RDT&E	M$	25.135
PROD.	M$	53.017
GSE&IS	M$	5.379
O & S	M$	48.087
DISP.	M$	0.178
TOTAL	M$	131.796

Aircraft Configuration Data - 30% Advanced Materials

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	30.00	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1688.9
TOTAL FUSELAGE GROUP	1679.4
TOTAL EMPENNAGE GROUP	175.4
TOTAL UNDERCARRIAGE	820.4
ENGINE & INSTALLATION	2092.3
SYSTEMS	
FUEL SYSTEM & TANKS	482.3
POWER & FLYING CONTROLS	335.1
HYDRAULICS & PNEUMATICS	234.1
ELECTRICS & BATTERIES	260.6
AIR SYSTEM (AIRCON/ICE)	214.5
COCKPIT INSTRUMENTS ETC	341.5
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	209.2
AIRCRAFT EMPTY MASS	10238.6

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	50.545	m^2
TOTAL SPAN	10.545	m
MEAN AERO CHORD	5.666	m
LDG EDGE SWEEP	53.175	Deg
1/4 CHORD SWEEP	44.987	Deg
ASPECT RATIO	2.200	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0480	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	75.009	kg/s
MAXIMUM THRUST	182.726	kN
MILITARY THRUST	122.216	kN
INLET DIAMETER	0.755	m
MAXIMUM DIAMETER	0.944	m
BARE ENGINE LENGTH	2.234	m
OVERALL LENGTH	3.948	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.548	m
EFFECTIVE DIA.	1.984	m
MAXIMUM HEIGHT	1.133	m
MAXIMUM WIDTH	2.971	m
ENG INLET AREA	0.669	m^2
BASE DRAG AREA	1.211	m^2
CENTRELINE HT.	2.233	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	0.988
LANDING CL	1.307
HIGH SPEED CL	1.481
BUFFET MAX CL	0.523
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.374	m^2
TAIL MEAN AERO CHRD	1.161	m
TAIL ASPECT RATIO	1.760	
TAIL THK/CHRD RATIO	0.0432	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.810	m^2
FIN MEAN AERO CHRD	2.561	m
FIN ASPECT RATIO	0.733	
FIN THCK/CHRD RATIO	0.0432	

MISSION PERFORMANCE PARAMETERS

	ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
PRE-MISSION							15387.0	1.0000
1 START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15218.6	0.9891
2 TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15121.6	0.9936
3 CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	14690.4	0.9715
4 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	13894.6	0.9458
5 DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	13894.6	1.0000
6 LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	13546.5	0.9749
7 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	13353.5	0.9858
8 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13247.5	0.9921
9 COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	13095.4	0.9885
10 COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	12270.0	0.9370
11 EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	12110.8	0.9870
12 CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	11925.4	0.9847
13 SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	11821.4	0.9913
14 DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	11821.4	1.0000
15 SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11096.2	0.9387
16 DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11096.2	1.0000
17 LANDING	0.0	0.18	0.00	0.00	0.0	84.9	11040.7	0.9950
18 GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	10872.3	0.9847

PERFORMANCE CONSTRAINT ANALYSIS

CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1 TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	235.02
2 LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3 ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.70
4 ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.12
5 SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.13
6 SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.19
7 MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	43.36
8 MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	71.43
9 CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10238.6	kg	WING LOADING 304.42 kg/m^2
CREW INCLUDING EQPT	100.0	kg	THRUST/WEIGHT 0.870
TOTAL WEAPONS MASS	475.0	kg	FUEL FRACTION 0.214
INTERNAL FUEL MASS	4573.4	kg	XS FUEL VOL. 0.000 m^3
EXTERNAL FUEL MASS	0.0	kg	THRUST RATIO 1.152
AIRCRAFT ALL-UP MASS	15387.0	kg	
AIRCRAFT GROSS MASS	21412.0	kg	

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	12491.7	lb.	No. DEVELOPMENT AIRCRAFT	9		No. PRODUCTION AIRCRAFT	620		AIR SERVICES	0.047
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES	2		AIFRAME ENGINEERING	M$ 3430.3		FLYING CONTROLS	0.563
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES	21.		PRODUCTION TOOLING	M$ 1199.6		FUEL SYSTEM	0.146
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$ 2081.0		MANUFACTURING LABOUR	M$ 10906.2		HYD. & PNEU.	0.282
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$ 1054.2		QUALITY CNTRL LBR COST	M$ 1652.8		UNDERCARRIAGE	0.710
ADV. MATERIAL FACTOR	1.235		FLIGHT TESTING COST	M$ 544.0		MANUFACTURING MATERIAL	M$ 2844.9		OXYGEN SYSTEM	0.074
ADV. TECH. FACTOR	1.270		DEVELOPMENT TOOLING COST	M$ 1045.2		PRODUCTION ENGINE COST	M$ 6677.0		MISC. MECH	0.005
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$ 1146.2		AVIONICS PURCHASE COST	M$ 6951.3		TOTAL STRUCTURE	0.653
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$ 173.7					POWERPLANT SYST.	0.494
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$ 97.2		TOTAL PRODUCTION COST	M$ 33662.1		ARMAMENT SYSTEM	1.058
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$ 3519.9					AVIONICS SYSTEM	0.198
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$ 158.5		G.S.E. & INITIAL SPARES	M$ 3366.2		NAV. & COMMS	0.228
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$ 3352.2					ELEC. & INSTR.	0.319
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$ 78.5		ENGINEERING LABOUR RATE	118.5 $/hr		SCHEDULED MAINT.	4.745
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$ 2627.4		TOOLING/QUALITY LABOUR RT	86.8 $/hr		TOTAL MAINTENANCE	9.524
			TOTAL DEVELOPMENT COST	M$15878.0		MANUFACTURING LABOUR RATE	74.5 $/hr			

OPERATION & SUPPORT PER BASE			DISPOSAL			AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
No. AIRCRAFT ON BASE	39.		DISPOSAL LABOUR COST	M$	545.3	RDT&E	M$	25.243
AVG. YRLY FLIGHT TIME	300.0		ADVANCED MATERIAL DISP.	M$	0.1	PROD.	M$	53.517
No. OPS. PERSONNEL	684.2		METALS SCRAP VALUE	M$	-48.9	GSE&IS	M$	5.429
No. SUPP. PERSONNEL	397.5					O & S	M$	47.582
OPS. PERSONNEL COST	M$	28.553	TOTAL DISPOSAL COST	M$	496.5	DISP.	M$	0.184
SUPP. PERSONNEL COST	M$	16.395						
SUPPORT & TRAINING	M$	5.478	DEFLATED DISPOSAL COST	M$	115.7	TOTAL	M$	131.955
UNIT LVL CONSUMPTION	M$	14.779						
ENGINE CONTRACT COST	M$	26.864						
MECH. CONTRACT COST	M$	17.064						
AVIONICS CNTRCT COST	M$	16.605						
SUPPLY CONTRACT COST	M$	1.970						
SUSTAINING SUPPORT	M$	13.142						
INSTALLATION SUPPORT	M$	4.314						
1st YR BASE O&S COST	M$	145.164						
TOTAL 25.YR O&S COST	M$	1855.688						

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.32	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1644.3
TOTAL FUSELAGE GROUP	1665.6
TOTAL EMPENNAGE GROUP	177.5
TOTAL UNDERCARRIAGE	830.5
ENGINE & INSTALLATION	2157.7
SYSTEMS	
FUEL SYSTEM & TANKS	488.6
POWER & FLYING CONTROLS	508.0
HYDRAULICS & PNEUMATICS	117.8
ELECTRICS & BATTERIES	395.2
AIR SYSTEM (AIRCON/ICE)	217.0
COCKPIT INSTRUMENTS ETC	342.7
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	215.8
AIRCRAFT EMPTY MASS	10465.7

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	49.472	m^2
TOTAL SPAN	10.834	m
MEAN AERO CHORD	5.398	m
LDG EDGE SWEEP	49.370	Deg
1/4 CHORD SWEEP	40.495	Deg
ASPECT RATIO	2.373	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0476	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	76.929	kg/s
MAXIMUM THRUST	187.404	kN
MILITARY THRUST	125.345	kN
INLET DIAMETER	0.764	m
MAXIMUM DIAMETER	0.955	m
BARE ENGINE LENGTH	2.264	m
OVERALL LENGTH	4.010	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.920	m
EFFECTIVE DIA.	2.009	m
MAXIMUM HEIGHT	1.146	m
MAXIMUM WIDTH	3.013	m
ENG INLET AREA	0.686	m^2
BASE DRAG AREA	1.242	m^2
CENTRELINE HT.	2.279	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.029
LANDING CL	1.360
HIGH SPEED CL	1.554
BUFFET MAX CL	0.544
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.319	m^2
TAIL MEAN AERO CHRD	1.105	m
TAIL ASPECT RATIO	1.898	
TAIL THK/CHRD RATIO	0.0428	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.933	m^2
FIN MEAN AERO CHRD	2.498	m
FIN ASPECT RATIO	0.791	
FIN THCK/CHRD RATIO	0.0428	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15671.8	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15499.1	0.9890
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15400.2	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	14960.4	0.9714
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14167.3	0.9470
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14167.3	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13820.3	0.9755
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13622.6	0.9857
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13514.0	0.9920
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13359.9	0.9886
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12513.3	0.9366
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12354.2	0.9873
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12163.6	0.9846
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	12057.1	0.9913
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	12057.1	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11331.5	0.9398
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11331.5	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.9	11274.8	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	11102.1	0.9847

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	232.86
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.18
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.42
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.29
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	46.52
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	72.38
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10465.7 kg	WING LOADING	316.78 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.881
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.213
INTERNAL FUEL MASS	4631.2 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.154
AIRCRAFT ALL-UP MASS	15671.8 kg		
AIRCRAFT GROSS MASS	21696.8 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	12515.4	lb.	No. DEVELOPMENT AIRCRAFT		9	No. PRODUCTION AIRCRAFT		620		
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES		2	AIFRAME ENGINEERING	M$	3615.0	AIR SERVICES	0.047
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES		21.	PRODUCTION TOOLING	M$	1232.9	FLYING CONTROLS	0.284
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$	2193.0	MANUFACTURING LABOUR	M$	11208.5	FUEL SYSTEM	0.150
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$	1082.7	QUALITY CNTRL LBR COST	M$	1698.6	HYD. & PNEU.	0.294
ADV. MATERIAL FACTOR	1.267		FLIGHT TESTING COST	M$	545.3	MANUFACTURING MATERIAL	M$	2923.3	UNDERCARRIAGE	0.714
ADV. TECH. FACTOR	1.302		DEVELOPMENT TOOLING COST	M$	1074.2	PRODUCTION ENGINE COST	M$	6758.7	OXYGEN SYSTEM	0.076
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$	1178.0	AVIONICS PURCHASE COST	M$	6951.3	MISC. MECH	0.005
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$	178.5				TOTAL STRUCTURE	0.632
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$	99.9	TOTAL PRODUCTION COST	M$	34388.4	POWERPLANT SYST.	0.505
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$	3521.2				ARMAMENT SYSTEM	1.058
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$	160.5	G.S.E. & INITIAL SPARES	M$	3438.8	AVIONICS SYSTEM	0.216
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$	3352.2				NAV. & COMMS	0.233
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$	78.5	ENGINEERING LABOUR RATE		118.5 $/hr	ELEC. & INSTR.	0.360
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$	2627.4	TOOLING/QUALITY LABOUR RT		86.8 $/hr	SCHEDULED MAINT.	4.638
			TOTAL DEVELOPMENT COST	M$	16091.3	MANUFACTURING LABOUR RATE		74.5 $/hr	TOTAL MAINTENANCE	9.211

OPERATION & SUPPORT PER BASE			DISPOSAL			AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
No. AIRCRAFT ON BASE	39.		DISPOSAL LABOUR COST	M$	560.4	RDT&E	M$	25.582
AVG. YRLY FLIGHT TIME	300.0		ADVANCED MATERIAL DISP.	M$	0.1	PROD.	M$	54.672
No. OPS. PERSONNEL	667.7		METALS SCRAP VALUE	M$	-46.6	GSE&IS	M$	5.547
No. SUPP. PERSONNEL	394.7					O & S	M$	47.427
OPS. PERSONNEL COST	M$	27.899	TOTAL DISPOSAL COST	M$	513.8	DISP.	M$	0.190
SUPP. PERSONNEL COST	M$	16.286						
SUPPORT & TRAINING	M$	5.355	DEFLATED DISPOSAL COST	M$	119.7	TOTAL	M$	133.418
UNIT LVL CONSUMPTION	M$	14.741						
ENGINE CONTRACT COST	M$	27.353						
MECH. CONTRACT COST	M$	16.842						
AVIONICS CNTRCT COST	M$	16.605						
SUPPLY CONTRACT COST	M$	1.979						
SUSTAINING SUPPORT	M$	13.397						
INSTALLATION SUPPORT	M$	4.237						
1st YR BASE O&S COST	M$	144.693						
TOTAL 25.YR O&S COST	M$	1849.667						

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.50	1.00	1.00	0.50	1.50	1.00	1.00	1.02	1.00
SCHED. MAINT	**AIRSYS R&M**	**FCS R&M**	**HYD/PNU R&M**	**ENGINE R&M**	**ELECTRIC R&M**	**GSE/O&S FACT**	**ENG. BLEED**	**ENG. POWER**
0.90	1.00	0.50	1.00	1.00	1.10	1.00	1.00	0.90

More Electric Aircraft Configuration Data – EHA Level 2

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.49	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1593.7
TOTAL FUSELAGE GROUP	1720.4
TOTAL EMPENNAGE GROUP	178.7
TOTAL UNDERCARRIAGE	828.7
ENGINE & INSTALLATION	2113.7
SYSTEMS	
FUEL SYSTEM & TANKS	487.4
POWER & FLYING CONTROLS	507.1
HYDRAULICS & PNEUMATICS	0.0
ELECTRICS & BATTERIES	525.9
AIR SYSTEM (AIRCON/ICE)	216.6
COCKPIT INSTRUMENTS ETC	342.4
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	211.4
AIRCRAFT EMPTY MASS	10431.0

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	49.030	m^2
TOTAL SPAN	10.409	m
MEAN AERO CHORD	5.568	m
LDG EDGE SWEEP	49.415	Deg
1/4 CHORD SWEEP	39.789	Deg
ASPECT RATIO	2.210	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0446	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	75.638	kg/s
MAXIMUM THRUST	184.261	kN
MILITARY THRUST	123.243	kN
INLET DIAMETER	0.758	m
MAXIMUM DIAMETER	0.948	m
BARE ENGINE LENGTH	2.244	m
OVERALL LENGTH	3.969	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.413	m
EFFECTIVE DIA.	1.992	m
MAXIMUM HEIGHT	1.137	m
MAXIMUM WIDTH	2.985	m
ENG INLET AREA	0.675	m^2
BASE DRAG AREA	1.221	m^2
CENTRELINE HT.	2.328	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.035
LANDING CL	1.368
HIGH SPEED CL	1.537
BUFFET MAX CL	0.547
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.283	m^2
TAIL MEAN AERO CHRD	1.136	m
TAIL ASPECT RATIO	1.768	
TAIL THK/CHRD RATIO	0.0401	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.010	m^2
FIN MEAN AERO CHRD	2.608	m
FIN ASPECT RATIO	0.737	
FIN THCK/CHRD RATIO	0.0401	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15622.5	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15452.7	0.9891
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15354.4	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.6	14916.7	0.9715
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.7	14116.5	0.9464
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.7	14116.5	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.7	13765.5	0.9751
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.3	13569.9	0.9858
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.3	13464.8	0.9923
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.7	13305.1	0.9881
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.7	12472.7	0.9374
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.8	12313.5	0.9872
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.3	12125.8	0.9847
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.3	12022.5	0.9915
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.3	12022.5	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.4	11293.0	0.9393
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.7	11293.0	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.9	11236.6	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.9	11066.7	0.9849

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	237.22
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.04
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.10
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	47.69
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	73.38
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10431.0 kg	WING LOADING	318.63 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.868
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.213
INTERNAL FUEL MASS	4616.5 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.172
AIRCRAFT ALL-UP MASS	15622.5 kg		
AIRCRAFT GROSS MASS	21647.5 kg		

More Electric Aircraft Cost Data – EHA Level 2

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	12261.6	lb.	No. DEVELOPMENT AIRCRAFT		9	No. PRODUCTION AIRCRAFT		620	AIR SERVICES	0.048
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES		2	AIFRAME ENGINEERING	M$	3596.8	FLYING CONTROLS	0.282
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES		21.	PRODUCTION TOOLING	M$	1215.2	FUEL SYSTEM	0.151
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$	2182.0	MANUFACTURING LABOUR	M$	11049.3	HYD. & PNEU.	0.000
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$	1073.6	QUALITY CNTRL LBR COST	M$	1674.5	UNDERCARRIAGE	0.713
ADV. MATERIAL FACTOR	1.269		FLIGHT TESTING COST	M$	532.1	MANUFACTURING MATERIAL	M$	2885.8	OXYGEN SYSTEM	0.073
ADV. TECH. FACTOR	1.315		DEVELOPMENT TOOLING COST	M$	1058.7	PRODUCTION ENGINE COST	M$	6703.9	MISC. MECH	0.005
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$	1161.3	AVIONICS PURCHASE COST	M$	6951.3	TOTAL STRUCTURE	0.629
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$	176.0				POWERPLANT SYST.	0.505
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$	98.6	TOTAL PRODUCTION COST	M$	34076.9	ARMAMENT SYSTEM	1.058
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$	3520.3				AVIONICS SYSTEM	0.245
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$	159.2	G.S.E. & INITIAL SPARES	M$	3066.9	NAV. & COMMS	0.230
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$	3352.2				ELEC. & INSTR.	0.394
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$	78.5	ENGINEERING LABOUR RATE		118.5 $/hr	SCHEDULED MAINT.	4.410
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$	2627.4	TOOLING/QUALITY LABOUR RT		86.8 $/hr	TOTAL MAINTENANCE	8.5
			TOTAL DEVELOPMENT COST	M$	16019.9	MANUFACTURING LABOUR RATE		74.5 $/hr		

OPERATION & SUPPORT PER BASE			DISPOSAL			AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
No. AIRCRAFT ON BASE	39.		DISPOSAL LABOUR COST	M$	552.5	RDT&E	M$	25.469
AVG. YRLY FLIGHT TIME	300.0		ADVANCED MATERIAL DISP.	M$	0.1	PROD.	M$	54.176
No. OPS. PERSONNEL	644.5		METALS SCRAP VALUE	M$	-45.6	GSE&IS	M$	4.947
No. SUPP. PERSONNEL	390.7					O & S	M$	46.422
OPS. PERSONNEL COST	M$	26.978	TOTAL DISPOSAL COST	M$	506.9	DISP.	M$	0.188
SUPP. PERSONNEL COST	M$	16.132						
SUPPORT & TRAINING	M$	5.182	DEFLATED DISPOSAL COST	M$	118.1	TOTAL	M$	131.201
UNIT LVL CONSUMPTION	M$	14.613						
ENGINE CONTRACT COST	M$	27.024						
MECH. CONTRACT COST	M$	16.291						
AVIONICS CNTRCT COST	M$	16.605						
SUPPLY CONTRACT COST	M$	1.944						
SUSTAINING SUPPORT	M$	12.728						
INSTALLATION SUPPORT	M$	4.129						
1st YR BASE O&S COST	M$	141.625						
TOTAL 25.YR O&S COST	M$	1810.442						

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.50	1.00	1.00	0.00	2.00	1.00	1.00	1.03	1.00
SCHED. MAINT	**AIRSYS R&M**	**FCS R&M**	**HYD/PNU R&M**	**ENGINE R&M**	**ELECTRIC R&M**	**GSE/O&S FACT**	**ENG. BLEED**	**ENG. POWER**
0.80	1.00	0.50	0.00	1.00	1.20	0.90	1.00	0.90

More Electric Aircraft Configuration Data – EHA Level 3

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.08	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1500.8
TOTAL FUSELAGE GROUP	1745.8
TOTAL EMPENNAGE GROUP	179.7
TOTAL UNDERCARRIAGE	829.6
ENGINE & INSTALLATION SYSTEMS	1964.1
FUEL SYSTEM & TANKS	489.4
POWER & FLYING CONTROLS	507.6
HYDRAULICS & PNEUMATICS	0.0
ELECTRICS & BATTERIES	658.0
AIR SYSTEM (AIRCON/ICE)	216.8
COCKPIT INSTRUMENTS ETC	342.5
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	218.2
AIRCRAFT EMPTY MASS	10357.5

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	49.084	m^2
TOTAL SPAN	10.248	m
MEAN AERO CHORD	5.662	m
LDG EDGE SWEEP	49.668	Deg
1/4 CHORD SWEEP	39.774	Deg
ASPECT RATIO	2.140	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0478	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	77.648	kg/s
MAXIMUM THRUST	189.157	kN
MILITARY THRUST	126.518	kN
INLET DIAMETER	0.768	m
MAXIMUM DIAMETER	0.959	m
BARE ENGINE LENGTH	2.275	m
OVERALL LENGTH	4.033	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.347	m
EFFECTIVE DIA.	2.018	m
MAXIMUM HEIGHT	1.150	m
MAXIMUM WIDTH	3.029	m
ENG INLET AREA	0.693	m^2
BASE DRAG AREA	1.253	m^2
CENTRELINE HT.	2.327	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.035
LANDING CL	1.369
HIGH SPEED CL	1.526
BUFFET MAX CL	0.547
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.288	m^2
TAIL MEAN AERO CHRD	1.156	m
TAIL ASPECT RATIO	1.712	
TAIL THK/CHRD RATIO	0.0430	
NUMBER OF FINS	1	
TOTAL FIN AREA	5.041	m^2
FIN MEAN AERO CHRD	2.659	m
FIN ASPECT RATIO	0.713	
FIN THCK/CHRD RATIO	0.0430	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15646.3	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15472.0	0.9889
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15373.0	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	14936.3	0.9716
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	14120.0	0.9453
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	14120.0	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13761.6	0.9746
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13562.8	0.9856
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	13451.0	0.9918
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	13284.6	0.9876
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12430.1	0.9357
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	12270.9	0.9872
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	12080.9	0.9845
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	11971.1	0.9909
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	11971.1	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	11228.3	0.9380
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	11228.3	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.8	11172.2	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	10997.8	0.9844

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	229.35
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.01
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.04
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	47.23
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	67.96
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10357.5 kg	WING LOADING	318.76 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.890
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.218
INTERNAL FUEL MASS	4713.8 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.128
AIRCRAFT ALL-UP MASS	15646.3 kg		
AIRCRAFT GROSS MASS	21671.3 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES		
AMPR (AIRFRAME) MASS	11637.6	lb.
MAX. TRUE AIRSPEED	1258.0	kt.
UNINST. AVION. MASS	1597.2	lb.
NUMBER OF PARTNERS	2.672	
COLLABORATION FACTOR	1.383	
ADV. MATERIAL FACTOR	1.265	
ADV. TECH. FACTOR	1.365	
ADV. DESIGN TOOLS	1.300	
ADV. TECH TESTING	1.200	
LOW OBSERVABLES	1.200	
SECURITY FACTOR	1.200	
FIRST FLIGHT YEAR	1996.	
FINANCIAL YEAR	2000.	
CONSUMER PRICE INDEX	1.777	
DISCOUNTING RATE	6.00%	

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT		9
No. STATIC AIRFRAMES		2
No. DEVELOPMENT ENGINES		21.
AIRFRAME ENGINEERING COST	M$	2166.3
DEVELOPMENT SUPPORT COST	M$	1063.2
FLIGHT TESTING COST	M$	500.1
DEVELOPMENT TOOLING COST	M$	1013.9
MANUFACTURING LABOUR	M$	1112.6
QUALITY CONTROL LABOUR	M$	168.6
MANUFACTURING MATERIAL	M$	94.8
ENGINE DEVELOPMENT COST	M$	3521.7
DEVELOPMENT ENGINE COST	M$	161.2
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15860.5

PRODUCTION		
No. PRODUCTION AIRCRAFT		620
AIFRAME ENGINEERING	M$	3570.9
PRODUCTION TOOLING	M$	1163.8
MANUFACTURING LABOUR	M$	10586.2
QUALITY CNTRL LBR COST	M$	1604.3
MANUFACTURING MATERIAL	M$	2774.7
PRODUCTION ENGINE COST	M$	6789.1
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	33440.3
G.S.E. & INITIAL SPARES	M$	3009.6
ENGINEERING LABOUR RATE		118.5 $/hr
TOOLING/QUALITY LABOUR RT		86.8 $/hr
MANUFACTURING LABOUR RATE		74.5 $/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.048
FLYING CONTROLS	0.285
FUEL SYSTEM	0.152
HYD. & PNEU.	0.000
UNDERCARRIAGE	0.712
OXYGEN SYSTEM	0.073
MISC. MECH	0.005
TOTAL STRUCTURE	0.630
POWERPLANT SYST.	0.511
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.239
NAV. & COMMS	0.235
ELEC. & INSTR.	0.400
SCHEDULED MAINT	236
TOTAL MAINTENANCE	8.582

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE		39.
AVG. YRLY FLIGHT TIME		300.0
No. OPS. PERSONNEL		638.4
No. SUPP. PERSONNEL		389.7
OPS. PERSONNEL COST	M$	26.737
SUPP. PERSONNEL COST	M$	16.092
SUPPORT & TRAINING	M$	5.135
UNIT LVL CONSUMPTION	M$	14.610
ENGINE CONTRACT COST	M$	27.536
MECH. CONTRACT COST	M$	16.118
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.956
SUSTAINING SUPPORT	M$	12.524
INSTALLATION SUPPORT	M$	4.101
1st YR BASE O&S COST	M$	141.414
TOTAL 25.YR O&S COST	M$	1807.745

DISPOSAL		
DISPOSAL LABOUR COST	M$	529.3
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-43.5
TOTAL DISPOSAL COST	M$	485.8
DEFLATED DISPOSAL COST	M$	113.2

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	25.215
PROD.	M$	53.164
GSE&IS	M$	4.854
O & S	M$	46.352
DISP.	M$	0.180
TOTAL	M$	129.766

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.50	1.00	0.90	0.00	2.50	1.00	1.00	1.07	1.00

SCHED. MAINT	AIRSYS R&M	FCS R&M	HYD/PNU R&M	ENGINE R&M	ELECTRIC R&M	GSE/O&S FACT	ENG. BLEED	ENG. POWER
0.70	1.00	0.50	0.00	1.00	1.20	0.90	1.00	0.90

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	31.74	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1407.3
TOTAL FUSELAGE GROUP	1641.8
TOTAL EMPENNAGE GROUP	176.4
TOTAL UNDERCARRIAGE	826.4
ENGINE & INSTALLATION	2282.6
SYSTEMS	
FUEL SYSTEM & TANKS	489.2
POWER & FLYING CONTROLS	337.2
HYDRAULICS & PNEUMATICS	117.5
ELECTRICS & BATTERIES	393.4
AIR SYSTEM (AIRCON/ICE)	216.0
COCKPIT INSTRUMENTS ETC	341.6
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	228.3
AIRCRAFT EMPTY MASS	10162.7

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	48.826	m^2
TOTAL SPAN	10.141	m
MEAN AERO CHORD	5.692	m
LDG EDGE SWEEP	49.885	Deg
1/4 CHORD SWEEP	39.895	Deg
ASPECT RATIO	2.106	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0520	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	80.548	kg/s
MAXIMUM THRUST	196.221	kN
MILITARY THRUST	131.242	kN
INLET DIAMETER	0.782	m
MAXIMUM DIAMETER	0.974	m
BARE ENGINE LENGTH	2.320	m
OVERALL LENGTH	4.125	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.391	m
EFFECTIVE DIA.	2.056	m
MAXIMUM HEIGHT	1.169	m
MAXIMUM WIDTH	3.090	m
ENG INLET AREA	0.719	m^2
BASE DRAG AREA	1.300	m^2
CENTRELINE HT.	2.234	m

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.317	m^2
TAIL MEAN AERO CHRD	1.173	m
TAIL ASPECT RATIO	1.685	
TAIL THK/CHRD RATIO	0.0468	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.897	m^2
FIN MEAN AERO CHRD	2.641	m
FIN ASPECT RATIO	0.702	
FIN THCK/CHRD RATIO	0.0468	

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.034
LANDING CL	1.367
HIGH SPEED CL	1.519
BUFFET MAX CL	0.547
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15555.4	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	15374.5	0.9884
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	15275.2	0.9935
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.3	14843.4	0.9717
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.4	14015.2	0.9442
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.4	14015.2	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.4	13652.7	0.9741
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.0	13450.3	0.9852
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.0	13328.0	0.9909
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.4	13154.5	0.9870
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.4	12268.0	0.9326
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.5	12108.9	0.9870
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.0	11916.3	0.9841
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.0	11795.9	0.9899
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.0	11795.9	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.1	11044.1	0.9363
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.4	11044.1	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.6	10988.8	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.6	10808.0	0.9835

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	216.44
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.02
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.18
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.03
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	45.20
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	58.57
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10162.7 kg	WING LOADING	318.59 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.927
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.223
INTERNAL FUEL MASS	4817.7 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.065
AIRCRAFT ALL-UP MASS	15555.4 kg		
AIRCRAFT GROSS MASS	21580.4 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	11553.0	lb.	No. DEVELOPMENT AIRCRAFT		9	No. PRODUCTION AIRCRAFT		620		
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES		2	AIFRAME ENGINEERING	M$	3375.0	AIR SERVICES	0.047
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES		21.	PRODUCTION TOOLING	M$	1145.3	FLYING CONTROLS	0.288
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$	2047.4	MANUFACTURING LABOUR	M$	10418.5	FUEL SYSTEM	0.151
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$	1014.6	QUALITY CNTRL LBR COST	M$	1578.9	HYD. & PNEU.	0.277
ADV. MATERIAL FACTOR	1.252		FLIGHT TESTING COST	M$	495.7	MANUFACTURING MATERIAL	M$	2732.1	UNDERCARRIAGE	0.709
ADV. TECH. FACTOR	1.312		DEVELOPMENT TOOLING COST	M$	997.8	PRODUCTION ENGINE COST	M$	6909.9	OXYGEN SYSTEM	0.072
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$	1095.0	AVIONICS PURCHASE COST	M$	6951.3	MISC. MECH	0.005
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$	165.9				TOTAL STRUCTURE	0.634
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$	93.3	TOTAL PRODUCTION COST	M$	33110.9	POWERPLANT SYST.	0.512
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$	3523.6				ARMAMENT SYSTEM	1.058
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$	164.1	G.S.E. & INITIAL SPARES	M$	3311.1	AVIONICS SYSTEM	0.181
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$	3352.2				NAV. & COMMS	0.241
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$	78.5	ENGINEERING LABOUR RATE		118.5 $/hr	ELEC. & INSTR.	0.366
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$	2627.4	TOOLING/QUALITY LABOUR RT		86.8 $/hr	SCHEDULED MAINT.	4.668
			TOTAL DEVELOPMENT COST	M$	15655.6	MANUFACTURING LABOUR RATE		74.5 $/hr	TOTAL MAINTENANCE	9.206

OPERATION & SUPPORT PER BASE			DISPOSAL			AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
No. AIRCRAFT ON BASE	39.		DISPOSAL LABOUR COST	M$	520.9	RDT&E	M$	24.890
AVG. YRLY FLIGHT TIME	300.0		ADVANCED MATERIAL DISP.	M$	0.1	PROD.	M$	52.641
No. OPS. PERSONNEL	666.7		METALS SCRAP VALUE	M$	-44.1	GSE&IS	M$	5.340
No. SUPP. PERSONNEL	394.5					O & S	M$	47.585
OPS. PERSONNEL COST	M$	27.861	TOTAL DISPOSAL COST	M$	476.9	DISP.	M$	0.177
SUPP. PERSONNEL COST	M$	16.280						
SUPPORT & TRAINING	M$	5.349	DEFLATED DISPOSAL COST	M$	111.1	TOTAL	M$	130.632
UNIT LVL CONSUMPTION	M$	14.803						
ENGINE CONTRACT COST	M$	28.275						
MECH. CONTRACT COST	M$	16.782						
AVIONICS CNTRCT COST	M$	16.605						
SUPPLY CONTRACT COST	M$	2.011						
SUSTAINING SUPPORT	M$	12.975						
INSTALLATION SUPPORT	M$	4.233						
1st YR BASE O&S COST	M$	145.174						
TOTAL 25.YR O&S COST	M$	1855.806						

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.00	1.00	1.00	0.50	1.50	1.00	1.00	1.03	1.00
SCHED. MAINT	**AIRSYS R&M**	**FCS R&M**	**HYD/PNU R&M**	**ENGINE R&M**	**ELECTRIC R&M**	**GSE/O&S FACT**	**ENG. BLEED**	**ENG. POWER**
0.90	1.00	0.50	1.00	1.00	1.10	1.00	1.00	0.90

More Electric Aircraft Configuration Data – EMA Level 2

Twin Engine, Canard Configuration. 1 Crew, 1 Fin. LCC-Optimised

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.41	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1571.3
TOTAL FUSELAGE GROUP	1596.9
TOTAL EMPENNAGE GROUP	174.1
TOTAL UNDERCARRIAGE	809.6
ENGINE & INSTALLATION SYSTEMS	2067.8
FUEL SYSTEM & TANKS	475.6
POWER & FLYING CONTROLS	331.4
HYDRAULICS & PNEUMATICS	0.0
ELECTRICS & BATTERIES	515.0
AIR SYSTEM (AIRCON/ICE)	211.8
COCKPIT INSTRUMENTS ETC	338.7
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	206.8
AIRCRAFT EMPTY MASS	10004.0

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	48.499	m^2
TOTAL SPAN	10.317	m
MEAN AERO CHORD	5.557	m
LDG EDGE SWEEP	51.505	Deg
1/4 CHORD SWEEP	42.633	Deg
ASPECT RATIO	2.195	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0485	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	74.285	kg/s
MAXIMUM THRUST	180.964	kN
MILITARY THRUST	121.038	kN
INLET DIAMETER	0.751	m
MAXIMUM DIAMETER	0.940	m
BARE ENGINE LENGTH	2.223	m
OVERALL LENGTH	3.925	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	15.767	m
EFFECTIVE DIA.	1.974	m
MAXIMUM HEIGHT	1.128	m
MAXIMUM WIDTH	2.956	m
ENG INLET AREA	0.663	m^2
BASE DRAG AREA	1.199	m^2
CENTRELINE HT.	2.254	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.010
LANDING CL	1.336
HIGH SPEED CL	1.505
BUFFET MAX CL	0.534
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.289	m^2
TAIL MEAN AERO CHRD	1.142	m
TAIL ASPECT RATIO	1.756	
TAIL THK/CHRD RATIO	0.0437	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.841	m^2
FIN MEAN AERO CHRD	2.572	m
FIN ASPECT RATIO	0.732	
FIN THCK/CHRD RATIO	0.0437	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15086.5	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	14919.7	0.9889
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	14824.5	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	14401.7	0.9715
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	13620.9	0.9458
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	13620.9	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13279.7	0.9750
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.2	13089.3	0.9857
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.2	12983.2	0.9919
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	12829.5	0.9882
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	12012.0	0.9363
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	11852.8	0.9868
12	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	54.2	11670.1	0.9846
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.2	11565.8	0.9911
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.2	11565.8	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.3	10855.6	0.9386
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.6	10855.6	1.0000
17	LANDING	0.0	0.18	0.00	0.00	0.0	84.8	10801.3	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	10634.5	0.9846

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	231.83
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.66
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.09
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.12
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.15
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	43.82
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	68.08
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	10004.0 kg	WING LOADING	311.07 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.874
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.214
INTERNAL FUEL MASS	4507.6 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.138
AIRCRAFT ALL-UP MASS	15086.5 kg		
AIRCRAFT GROSS MASS	21111.5 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			DEVELOPMENT			PRODUCTION			RELIABILITY AND MAINTAINABILITY	
										EFFORT
AMPR (AIRFRAME) MASS	11497.9	lb.	No. DEVELOPMENT AIRCRAFT	9		No. PRODUCTION AIRCRAFT	620		AIR SERVICES	0.047
MAX. TRUE AIRSPEED	1258.0	kt.	No. STATIC AIRFRAMES	2		AIFRAME ENGINEERING	M$	3448.1	FLYING CONTROLS	0.281
UNINST. AVION. MASS	1597.2	lb.	No. DEVELOPMENT ENGINES	21.		PRODUCTION TOOLING	M$	1156.0	FUEL SYSTEM	0.147
NUMBER OF PARTNERS	2.672		AIRFRAME ENGINEERING COST	M$	2091.8	MANUFACTURING LABOUR	M$	10516.2	HYD. & PNEU.	0.000
COLLABORATION FACTOR	1.383		DEVELOPMENT SUPPORT COST	M$	1022.7	QUALITY CNTRL LBR COST	M$	1593.7	UNDERCARRIAGE	0.706
ADV. MATERIAL FACTOR	1.268		FLIGHT TESTING COST	M$	492.9	MANUFACTURING MATERIAL	M$	2758.6	OXYGEN SYSTEM	0.074
ADV. TECH. FACTOR	1.328		DEVELOPMENT TOOLING COST	M$	1007.1	PRODUCTION ENGINE COST	M$	6646.0	MISC. MECH	0.005
ADV. DESIGN TOOLS	1.300		MANUFACTURING LABOUR	M$	1105.2	AVIONICS PURCHASE COST	M$	6951.3	TOTAL STRUCTURE	0.612
ADV. TECH TESTING	1.200		QUALITY CONTROL LABOUR	M$	167.5				POWERPLANT SYST.	0.494
LOW OBSERVABLES	1.200		MANUFACTURING MATERIAL	M$	94.2	TOTAL PRODUCTION COST	M$	33069.8	ARMAMENT SYSTEM	1.058
SECURITY FACTOR	1.200		ENGINE DEVELOPMENT COST	M$	3519.4				AVIONICS SYSTEM	0.211
FIRST FLIGHT YEAR	1996.		DEVELOPMENT ENGINE COST	M$	157.8	G.S.E. & INITIAL SPARES	M$	2976.3	NAV. & COMMS	0.226
FINANCIAL YEAR	2000.		AVION. DEVELOPMENT COST	M$	3352.2				ELEC. & INSTR	0.383
CONSUMER PRICE INDEX	1.777		AVIONICS PURCHASE COST	M$	78.5	ENGINEERING LABOUR RATE	118.5	$/hr	SCHEDULED MAINT.	4.366
DISCOUNTING RATE	6.00%		SOFTWARE DEVELOPMENT	M$	2627.4	TOOLING/QUALITY LABOUR RT	86.8	$/hr	TOTAL MAINTENANCE	8.609
			TOTAL DEVELOPMENT COST	M$	15716.9	MANUFACTURING LABOUR RATE	74.5	$/hr		

OPERATION & SUPPORT PER BASE			DISPOSAL			AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
No. AIRCRAFT ON BASE	39.		DISPOSAL LABOUR COST	M$	525.8	RDT&E	M$	24.987
AVG. YRLY FLIGHT TIME	300.0		ADVANCED MATERIAL DISP.	M$	0.1	PROD.	M$	52.575
No. OPS. PERSONNEL	637.3		METALS SCRAP VALUE	M$	-42.8	GSE&IS	M$	4.800
No. SUPP. PERSONNEL	389.5					O & S	M$	45.891
OPS. PERSONNEL COST	M$	26.691	TOTAL DISPOSAL COST	M$	483.1	DISP.	M$	0.179
SUPP. PERSONNEL COST	M$	16.085						
SUPPORT & TRAINING	M$	5.128	DEFLATED DISPOSAL COST	M$	112.6	TOTAL	M$	128.432
UNIT LVL CONSUMPTION	M$	14.542						
ENGINE CONTRACT COST	M$	26.679						
MECH. CONTRACT COST	M$	15.890						
AVIONICS CNTRCT COST	M$	16.605						
SUPPLY CONTRACT COST	M$	1.915						
SUSTAINING SUPPORT	M$	12.376						
INSTALLATION SUPPORT	M$	4.095						
1st YR BASE O&S COST	M$	140.005						
TOTAL 25.YR O&S COST	M$	1789.739						

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.00	1.00	1.00	0.00	2.00	1.00	1.00	1.04	1.00
SCHED. MAINT	**AIRSYS R&M**	**FCS R&M**	**HYD/PNU R&M**	**ENGINE R&M**	**ELECTRIC R&M**	**GSE/O&S FACT**	**ENG. BLEED**	**ENG. POWER**
0.80	1.00	0.50	0.00	1.00	1.20	0.90	1.00	0.90

GENERAL PARAMETERS

AIRCRAFT ROLE	FIGHTER	
NUMBER OF CREW	1	
MAX HI-ALT MACH NO:	2.20	
MAX LO-ALT MACH NO:	1.20	
DESIGN MAX G LOAD	9.00	
ULTIMATE LOAD FACTOR	13.50	
MASS OF EXT WEAPONS	475.0	kg
MASS OF INT WEAPONS	0.0	kg
VOL. OF INT WEAPONS	0.00	m^3
VOL. OF GUN & AMMO.	0.31	m^3
% ADVANCED MATERIALS	33.23	

TOTAL STRUCTURE & SYSTEMS MASS

	(kg)
TOTAL WING GROUP	1434.3
TOTAL FUSELAGE GROUP	1683.7
TOTAL EMPENNAGE GROUP	175.8
TOTAL UNDERCARRIAGE	808.7
ENGINE & INSTALLATION	1893.7
SYSTEMS	
FUEL SYSTEM & TANKS	476.1
POWER & FLYING CONTROLS	331.0
HYDRAULICS & PNEUMATICS	0.0
ELECTRICS & BATTERIES	643.0
AIR SYSTEM (AIRCON/ICE)	211.6
COCKPIT INSTRUMENTS ETC	338.0
TOTAL AVIONICS SYSTEM	805.0
FIXED ARMAMENT ITEMS	900.0
AUXILIARY POWER UNIT	210.4
AIRCRAFT EMPTY MASS	9911.4

WING PARAMETERS

WING TYPE	DELTA WING	
GROSS AREA	47.234	m^2
TOTAL SPAN	10.004	m
MEAN AERO CHORD	5.581	m
LDG EDGE SWEEP	49.738	Deg
1/4 CHORD SWEEP	39.757	Deg
ASPECT RATIO	2.119	
TAPER RATIO	0.150	
THCK/CHRD RATIO	0.0489	

SLS ENGINE PARAMETERS

NUMBER OF ENGINES	2.	
AIR MASS FLOW RATE	75.357	kg/s
MAXIMUM THRUST	183.576	kN
MILITARY THRUST	122.785	kN
INLET DIAMETER	0.757	m
MAXIMUM DIAMETER	0.946	m
BARE ENGINE LENGTH	2.239	m
OVERALL LENGTH	3.959	m
FAN BYPASS RATIO	0.408	
COMP. PRESS. RATIO	26.001	

FUSELAGE PARAMETERS

OVERALL LENGTH	16.211	m
EFFECTIVE DIA.	1.988	m
MAXIMUM HEIGHT	1.135	m
MAXIMUM WIDTH	2.979	m
ENG INLET AREA	0.672	m^2
BASE DRAG AREA	1.216	m^2
CENTRELINE HT.	2.305	m

AERODYNAMIC PARAMETERS

TAKE-OFF CL	1.035
LANDING CL	1.369
HIGH SPEED CL	1.523
BUFFET MAX CL	0.548
FLAP TYPE	SINGLE SLOT
SLAT / LE FLAP	YES
LDG FLAP ANGLE	30.0 Deg

EMPENNAGE PARAMETERS

TAIL PLACEMENT	CANARD	
NET TAIL AREA	2.225	m^2
TAIL MEAN AERO CHRD	1.146	m
TAIL ASPECT RATIO	1.695	
TAIL THK/CHRD RATIO	0.0440	
NUMBER OF FINS	1	
TOTAL FIN AREA	4.960	m^2
FIN MEAN AERO CHRD	2.650	m
FIN ASPECT RATIO	0.706	
FIN THCK/CHRD RATIO	0.0440	

MISSION PERFORMANCE PARAMETERS

		ALT.	MACH	DATA1	DATA2	A/B	TIME	MASS	FRAC
	PRE-MISSION							15060.2	1.0000
1	START, WARM-UP	0.0	0.00	1.00	0.00	0.0	1.0	14891.0	0.9888
2	TAKE-OFF	15.0	0.25	0.05	1.20	1.0	1.2	14795.6	0.9936
3	CLIMB / ACCEL.	9144.0	0.85	0.00	0.00	0.0	4.5	14374.9	0.9716
4	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	33.6	13581.4	0.9448
5	DESCEND/DECEL.	9144.0	0.75	0.00	0.00	0.0	33.6	13581.4	1.0000
6	LOITER/C.A.P.	9144.0	0.75	15.00	0.00	0.0	48.6	13233.1	0.9744
7	CLIMB / ACCEL.	9144.0	1.40	0.00	0.00	1.0	49.1	13040.9	0.9855
8	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	50.1	12931.2	0.9916
9	COMBAT TURNS	9144.0	1.20	180.00	0.00	1.0	50.6	12769.0	0.9875
10	COMBAT TURNS	9144.0	0.90	3.00	0.00	1.0	53.6	11939.7	0.9351
11	EXTRNL WEAPONS	8382.0	0.90	0.34	1.00	1.0	53.7	11780.5	0.9867
12	CLIMB / ACCEL.	9144 0	1.40	0.00	0.00	1.0	54.1	11597.1	0.9844
13	SUPERCRUISE	9144.0	1.40	25.00	0.00	0.0	55.1	11489.3	0.9907
14	DESCEND/DECEL.	9144.0	0.85	0.00	0.00	0.0	55.1	11489.3	1.0000
15	SUBSON. CRUISE	9144.0	0.85	450.00	0.00	0.0	84.2	10768.2	0.9372
16	DESCEND/DECEL.	0.0	0.30	0.00	0.00	0.0	84.5	10768.2	1.0000
17	LANDING	0.0	0.17	0.00	0.00	0.0	84.8	10714.3	0.9950
18	GROUND MOVEMNT	0.0	0.01	1.00	0.00	0.0	85.8	10545.1	0.9842

PERFORMANCE CONSTRAINT ANALYSIS

	CONSTRAINT	ALT.	MACH	DATA1	DATA2	DATA3	DATA4	REQ.D	DEL.D
1	TAKE-OFF DIST.	0.0	0.00	0.0	0.95	1.00	1.0	250.0	226.78
2	LANDING DIST.	0.0	0.00	0.4	0.95	1.00	0.3	500.0	500.00
3	ATTAINED TURN	12750.0	1.40	1.0	0.50	1.00	0.0	9.0	9.00
4	ATTAINED TURN	10805.0	0.90	1.0	0.50	1.00	0.0	7.5	8.00
5	SUSTAINED TURN	6050.0	0.80	1.0	0.50	1.00	1.0	5.0	5.00
6	SUSTAINED TURN	12825.0	1.60	1.0	0.50	1.00	1.0	3.0	3.03
7	MACH XS THRUST	11500.0	2.00	0.0	0.50	1.00	1.0	0.0	45.38
8	MACH XS THRUST	0.0	1.20	0.0	0.80	1.00	1.0	0.0	63.60
9	CLIMB/ACC TIME	0.0	0.25	9144.0	1.50	1.00	1.0	1.5	1.50

AIRCRAFT EMPTY MASS	9911.4 kg	WING LOADING	318.85 kg/m^2
CREW INCLUDING EQPT	100.0 kg	THRUST/WEIGHT	0.888
TOTAL WEAPONS MASS	475.0 kg	FUEL FRACTION	0.217
INTERNAL FUEL MASS	4573.8 kg	XS FUEL VOL.	0.000 m^3
EXTERNAL FUEL MASS	0.0 kg	THRUST RATIO	1.114
AIRCRAFT ALL-UP MASS	15060.2 kg		
AIRCRAFT GROSS MASS	21085.2 kg		

LIFE CYCLE COST OUTPUT

GENERAL VALUES			
AMPR (AIRFRAME) MASS	10924.6	lb.	
MAX. TRUE AIRSPEED	1258.0	kt.	
UNINST. AVION. MASS	1597.2	lb.	
NUMBER OF PARTNERS	2.672		
COLLABORATION FACTOR	1.383		
ADV. MATERIAL FACTOR	1.266		
ADV. TECH. FACTOR	1.379		
ADV. DESIGN TOOLS	1.300		
ADV. TECH TESTING	1.200		
LOW OBSERVABLES	1.200		
SECURITY FACTOR	1.200		
FIRST FLIGHT YEAR	1996.		
FINANCIAL YEAR	2000.		
CONSUMER PRICE INDEX	1.777		
DISCOUNTING RATE	6.00%		

DEVELOPMENT		
No. DEVELOPMENT AIRCRAFT	9	
No. STATIC AIRFRAMES	2	
No. DEVELOPMENT ENGINES	21.	
AIRFRAME ENGINEERING COST	M$	2082.2
DEVELOPMENT SUPPORT COST	M$	1013.9
FLIGHT TESTING COST	M$	463.8
DEVELOPMENT TOOLING COST	M$	967.0
MANUFACTURING LABOUR	M$	1061.6
QUALITY CONTROL LABOUR	M$	160.9
MANUFACTURING MATERIAL	M$	90.8
ENGINE DEVELOPMENT COST	M$	3520.1
DEVELOPMENT ENGINE COST	M$	158.9
AVION. DEVELOPMENT COST	M$	3352.2
AVIONICS PURCHASE COST	M$	78.5
SOFTWARE DEVELOPMENT	M$	2627.4
TOTAL DEVELOPMENT COST	M$	15577.3

PRODUCTION		
No. PRODUCTION AIRCRAFT	620	
AIFRAME ENGINEERING	M$	3432.2
PRODUCTION TOOLING	M$	1109.9
MANUFACTURING LABOUR	M$	10101.4
QUALITY CNTRL LBR COST	M$	1530.8
MANUFACTURING MATERIAL	M$	2659.1
PRODUCTION ENGINE COST	M$	6691.9
AVIONICS PURCHASE COST	M$	6951.3
TOTAL PRODUCTION COST	M$	32476.7
G.S.E. & INITIAL SPARES	M$	2922.9
ENGINEERING LABOUR RATE	118.5	$/hr
TOOLING/QUALITY LABOUR RT	86.8	$/hr
MANUFACTURING LABOUR RATE	74.5	$/hr

RELIABILITY AND MAINTAINABILITY	
	EFFORT
AIR SERVICES	0.047
FLYING CONTROLS	0.282
FUEL SYSTEM	0.150
HYD. & PNEU.	0.000
UNDERCARRIAGE	0.704
OXYGEN SYSTEM	0.072
MISC. MECH	0.005
TOTAL STRUCTURE	0.609
POWERPLANT SYST.	0.502
ARMAMENT SYSTEM	1.058
AVIONICS SYSTEM	0.234
NAV. & COMMS	0.229
ELEC. & INSTR.	0.391
SCHEDULED MAINT.	4.172
TOTAL MAINTENANCE	8.457

OPERATION & SUPPORT PER BASE		
No. AIRCRAFT ON BASE	39.	
AVG. YRLY FLIGHT TIME	300.0	
No. OPS. PERSONNEL	632.2	
No. SUPP. PERSONNEL	388.6	
OPS. PERSONNEL COST	M$	26.491
SUPP. PERSONNEL COST	M$	16.051
SUPPORT & TRAINING	M$	5.089
UNIT LVL CONSUMPTION	M$	14.526
ENGINE CONTRACT COST	M$	26.952
MECH. CONTRACT COST	M$	15.705
AVIONICS CNTRCT COST	M$	16.605
SUPPLY CONTRACT COST	M$	1.918
SUSTAINING SUPPORT	M$	12.181
INSTALLATION SUPPORT	M$	4.072
1st YR BASE O&S COST	M$	139.589
TOTAL 25.YR O&S COST	M$	1784.414

DISPOSAL		
DISPOSAL LABOUR COST	M$	505.1
ADVANCED MATERIAL DISP.	M$	0.1
METALS SCRAP VALUE	M$	-40.8
TOTAL DISPOSAL COST	M$	464.4
DEFLATED DISPOSAL COST	M$	108.2

AVERAGE DISCOUNTED LIFE CYCLE COST/AIRCRAFT		
RDT&E	M$	24.765
PROD.	M$	51.632
GSE&IS	M$	4.714
O & S	M$	45.754
DISP.	M$	0.172
TOTAL	M$	127.038

MORE ELECTRIC AIRCRAFT SYSTEM FACTORS

FCS MASS	U/C MASS	ENGINST MASS	HYD/PNU.MASS	ELEC. MASS	AIRSYS MASS	APU MASS	ADV.TECH.F	EXCRES. DRAG
1.00	1.00	0.90	0.00	2.50	1.00	1.00	1.08	1.00

SCHED. MAINT	AIRSYS R&M	FCS R&M	HYD/PNU R&M	ENGINE R&M	ELECTRIC R&M	GSE/O&S FACT	ENG. BLEED	ENG. POWER
0.70	1.00	0.50	0.00	1.00	1.20	0.90	1.00	0.90