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**F93B - CAD MODEL, PERFORMANCE EVALUATION, ROLE
CONVERSION AND COST PREDICTION**

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This thesis is submitted in partial fulfilment of the requirements for the degree of
Master of Science

ACKNOWLEDGEMENT ABSTRACT

European Air forces have a general requirement for a new, multi-purpose large aircraft. Whilst the major need is for a replacement for the Lockheed C-130 Hercules, it is envisaged that the aircraft should be capable of fulfilling other roles. These include anti-submarine maritime patrol and tanker missions.

This thesis discusses the construction of a Computer Aided Design (CAD) model of the F93B and outlines the process required to convert the CAD model into a physical model. It also evaluates in detail the overall performance and costs of the F93B turbo-prop aircraft and basic pressure loading for the wing and fuselage using the SPARV aerodynamic panel program. Additionally, the conversion of the aircraft into an anti-submarine maritime patrol aircraft was assessed.

The F93B proved to be a very capable aircraft with a maximum take-off distance requirement of 1375m which was well within the design specification of 1500m. The rate of climb and service ceiling were a considerable improvement on the C-130 and the 3-engine performance of the F93B was exceptional. The F93B has a ferry range of 10186km compared with 8750km for the C-130 and the payload/range requirement was exceeded by 475km (8.6%).

The role conversion to a Maritime Patrol Aircraft (MPA) involved the addition of a large quantity of equipment both internally and externally including tactical operators positions, surveillance equipment, sonobuoys, radar, Electronic Support Measures (ESM) as well as offensive and defensive weapons for anti-submarine warfare. The maximum endurance of the F93BMPA with an internal fuel tank fitted is 14.6 hours which meets the specification laid down by the RAF for its maritime patrol aircraft of the future. However, the lack of a bomb bay means that the F93B has a limited offensive capability.

The unit cost of the basic F93B transport aircraft is US\$ 51.1 million providing at least 200 aircraft are manufactured whilst the unit cost of the F93BMPA variant rises to US\$ 63 million. The operating costs per flight hour are US\$ 3381 and US\$ 5765 respectively.

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Acknowledgements

To my darling husband whose unstinting support and love made the writing of this script possible.

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C_D	-	Coefficient of drag
C_{D0}	-	Zero lift coefficient of Drag
C_L	-	Coefficient of lift
D	-	Drag
F	-	Force
h_{sc}	-	Screen height
g	-	Acceleration due to gravity
Q	-	Banking force
IAH	-	Inlet Aircraft Regulations
L	-	Lift
m	-	Mass
R	-	Radius of the circular arc described by aircraft path
S	-	Wing area
S_a	-	Airborne distance to screen height
S_{gr}	-	Ground run distance
S_{VELOP}	-	Distance between rotation and lift-off
t	-	Time
u	-	Initial velocity
V_D	-	Drinking speed
V_C	-	Rate of climb
V_D	-	Design diving speed
V_C	-	Equivalent airspeed

NOTATION AND UNITS

Notation

The notation used throughout this thesis is either drawn from the particular reference stated in the text or is defined in the text itself. A summary of frequently used notation is presented below

Units

Some of the references quoted in this thesis use imperial rather than SI units. In the text the unit appropriate to the particular calculation is used with the other unit in parenthesis. Note that sample calculations may have been rounded compared with spreadsheet results.

Computer Aided Design (CAD)

LOM	-	Laminated Object modelling
SLA	-	Stereolithography Interface Apparatus
STL	-	Stereolithography

Performance

a	-	Acceleration
AUM	-	All up mass
C_D	-	Coefficient of drag
C_{D0}	-	Zero lift coefficient of Drag
C_L	-	Coefficient of lift
D	-	Drag
F	-	Force
h_{SC}	-	Screen height
g	-	Acceleration due to gravity
G	-	Braking force
JAR	-	Joint Aircraft Regulations
L	-	Lift
m	-	Mass
R	-	Radius of the circular arc described by an aircraft flight path
S	-	Wing area
S_a	-	Airborne distance to screen height
S_{GR}	-	Ground run distance
$S_{VR-VLOF}$	-	Distance between rotation and lift-off
t	-	Time
u	-	Initial velocity
V_B	-	Braking speed
V_C	-	Rate of climb
V_D	-	Design diving speed
V_e	-	Equivalent airspeed

V_{eff}	-	Engine failure speed
V_F	-	Flare speed
V_{MU}	-	Minimum unstick speed
V_{LOF}	-	Lift-off speed
V_R	-	Rotation speed
V_S	-	Stall speed
V_T	-	Touchdown speed
V_{TR}	-	True airspeed
V_1	-	Decision speed
V_2	-	Minimum speed at screen height with engine failure
V_3	-	Speed at screen height
W	-	Weight
X_G	-	Accelerating force

Symbols

α	-	angle of attack
γ_A	-	approach angle
γ_C	-	climb angle
Δ	-	increment
μ_r	-	coefficient of resistance
μ_B	-	coefficient of braking
ρ	-	air density
σ	-	relative air density

Role Conversion

ASW	-	Anti-Submarine Warfare
ESM	-	Electronic Support Measures
FLIR	-	Forward Looking Infra-red
LLTV	-	Low Light Television
MAD	-	Magnetic Anomaly Detector
MPA	-	Maritime Patrol Aircraft
SAR	-	Search and Rescue

Cost

C_{acq}	-	Acquisition cost
C_{apcm}	-	Airframe program production costs
C_{conmat}	-	Cost of consumable spares
C_{crew}	-	Cost of crew
C_{aedm}	-	Airframe engineering and design cost during manufacture
C_{aedr}	-	Airframe engineering and design cost during research
C_{depot}	-	Cost of depot storage
C_{disp}	-	Cost of disposing of aircraft at the end of its service life
C_{dstr}	-	Development, support and testing cost during research

C(e+a)m	-	Cost of engines and avionics for the program
C(e+a)r	-	Cost of engines and avionics for research
Cfinm	-	Cost of financing manufacturing phase
Cfinr	-	Cost of financing RDTE phase
Cftar	-	Flight test costs for research aircraft
Cftom	-	Cost of flight test operations during manufacture
Cftor	-	Cost of flight test operations during research
Cintm	-	Cost of the interior
Clcc	-	Total life cycle cost
Cman	-	Total manufacturing costs
Cmanm	-	Manufacturing labour cost for the program
Cmanr	-	Manufacturing labour cost during research
Cmatm	-	Manufacturing material cost for the program
Cmatr	-	Research material costs
Cmisc	-	Cost of miscellaneous operating items
Cmperdir	-	Cost of direct maintenance personnel
Cops	-	Total operating cost
Cpersdir	-	Cost of direct personnel
Cpersind	-	Cost of indirect personnel
Cpol	-	Program cost of fuel, oil and lubricants
Cpro	-	Total profit margin
Cpror	-	RDTE profit
Cqcm	-	Quality control cost during manufacture
Cqcr	-	Quality control cost during research
Crdte	-	Research, development, test and evaluation cost
Cspares	-	Cost of spares
Ctoolm	-	Manufacturing tooling cost for the program
Ctoolr	-	Research tooling costs
Ctsfr	-	Test and simulation facility cost

Figure 1.1 'GEMINT'

2.1 This thesis covers 5 independent topics concerning the F93B aircraft

a. As group leader for the F93B CAD model the logistics of creating the CAD model are discussed in Chapter 2 along with the operational details of how it creates some the individual components and the stereo lithography of the model to create a physical model

b. Chapter 3 evaluates the overall performance of the F93B in its various phases: the take-off, climb, cruise, descent and landing phases of a mission profile.

CHAPTER 1

INTRODUCTION

1.1 The F93B is a large 4 engined aircraft designed to satisfy the general requirement of European Air Forces for a new, multi-purpose transport aircraft. Whilst the major need is for a replacement for the Lockheed C-130 Hercules, it is envisaged that the aircraft should be capable of fulfilling other roles. The 2 most important being anti-submarine maritime patrol and tanker missions.

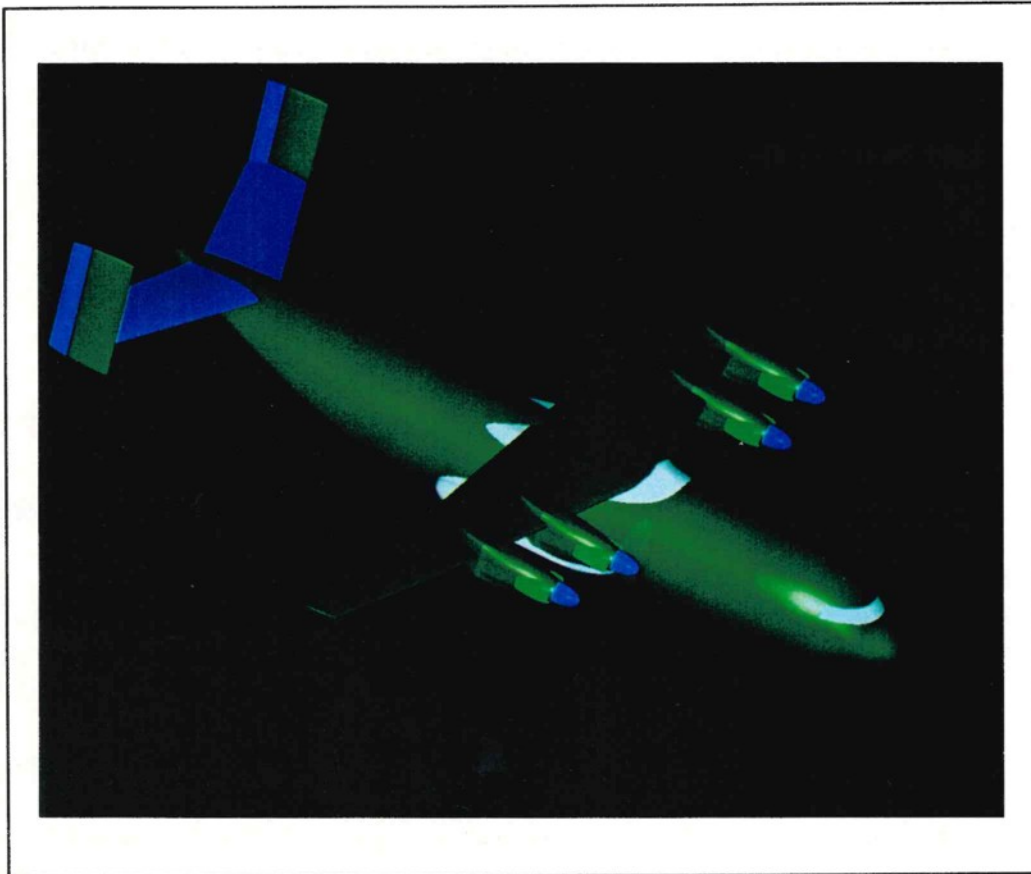


Figure 1.1 'GEMINI'

- 2.1 This thesis covers 5 independent topics concerning the F93B aircraft:
- a. As group leader for the F93B CAD model the logistics of creating the CAD model are discussed in Chapter 2 along with the operational details used to create some the individual components and the stereo lithographic process used to create a physical model.
 - b. Chapter 3 evaluates the overall performance of the F93B in several areas: the take-off ,climb, cruise, descent and landing phases of a mission profile and

assesses whether the F93B achieves the requirements laid down in the design specification Ref (1). The results of the performance evaluation are used to provide mission profile information for the generation of fatigue cycle data based on the flight pattern distribution specified in Ref (1).

c. In Chapter 4 the SPARV programme is used to confirm the basic wing and fuselage load distributions given in Ref (1) and on completion of the final CAD model it is intended that the aircraft dimensions are automatically fed into the SPARV programme to provide a more detailed assessment of the total aircraft loading.

d. The role conversion of the F93B into a maritime patrol aircraft is considered in Chapter 5 by looking at the additional equipment required and the resultant effect on the drag polar for the aircraft. The performance of the F93B maritime variant is also considered with a view to enhancing its maximum range and endurance capabilities.

e. In order to assess the competitiveness of the F93B a cost assessment is carried out in Chapter 6 which estimates the unit cost of the aircraft as well as the life cycle costs. Additional costs associated with the operation of the F93B maritime variant are also discussed.

DESIGN SPECIFICATION

Range

1.3 In the basic freighter role the aircraft is required to be capable of carrying a 20 tonne payload over a range of some 5500km and be able to operate from relatively short, semi-prepared strips. Over shorter ranges the payload requirement is 30 tonnes with sufficient dimensional clearance to accommodate a Eurocopter Super Puma.

Speed

1.4 The normal cruising speed requirement is for an improvement on the Mach 0.56 of the Hercules, but fuel economy is more important than a high subsonic Mach number. The design value of cruise speed (V_C) is Mach 0.7. The corresponding value of the design diving speed (V_D) is Mach 0.77.

Field Performance

1.5 The field length requirement is for operation from a normal runway of 1500m at an all up mass (AUM) of 101500kg and emergency operation from a 900m strip at reduced AUM. Although the 900m does not include the distance to and from the screen height.

DEFENCE STANDARD 00-970 REQUIREMENTS

1.6 As the F93B is a military transport aircraft its performance is required to satisfy Def Stan 00-970 rather than the civilian JARs. In particular this means that some of the requirements which civilian passenger and transport aircraft must achieve are relaxed slightly for military operations to allow for greater flexibility in aircraft design. For instance, the total take-off distance of an aircraft includes the horizontal distance travelled along the ground whilst airborne until the aircraft climbs to a safe height above the ground called the screen height. For most large civilian transport aircraft the screen height is 15m (50ft); for military aircraft the screen height is reduced to 10.7m (35ft).

1.7 According to Def Stan 00-970 leaflet 600/1 the F93B falls into group 3 for classification, that is a large/medium aircraft of low/medium manoeuvrability. Therefore the following specifications apply in particular to the F93B:

- The rotation speed V_R should be not less than:-
- The decision speed V_1
- The speed which allows the safety speed V_2 to be reached before reaching the screen height (10.7m).
- Take-off safety speed V_2 should be not less than:-
 - $1.15V_s$
 - The rotation speed (V_R), plus the speed increment attained while reaching a height of 10.7 (35ft) above the reference plane, having suffered an engine failure.

This Chapter outlines the construction techniques used to develop the components used for the F93B aircraft and discusses the process required to prepare the completed components for the rapid prototyping process called stereolithography.

Fuselage Model

2.3 The fuselage section was constructed in two parts: the nose and centre fuselage and merged together by the CAD group leaders. The geometry for the fuselage was taken from the design specification F93-3. Several points along the fuselage were defined using Cartesian co-ordinates which were then used to model B-spline curves. The

CHAPTER 2

F93B COMPUTER AIDED DESIGN MODEL

Introduction

2.1 The Computer Aided Design (CAD) group for the F93 project was formed to provide CAD support and develop a 3-dimensional CAD model for both the F93A and B aircraft using the McDonnell Douglas Unigraphics version 9.2 system. The dimensions of the resulting models are intended to be used to provide inputs for other processes, such as the SPARV, as well as providing the data for a physical model to be manufactured using the stereolithographic rapid prototyping technique.

2.2 To ensure correct assembly of the individual CAD model components a standardised file format was established. All model components were drawn in millimetres and a model datum was defined on the fuselage centreline at STN 0. The file co-ordinate system was defined as x being positive-rearward, y being positive-starboard and z being positive-upwards. Twenty layers were assigned to each component type, which consisted of separate layers for geometry, surfaces and dimensions to facilitate straightforward component assembly by the group leaders. In order to achieve a complete model those parts which were not drawn by individual students were the responsibility of the group leaders (F93A - Mr Buller, F93B - Mrs Gray). Individual components used in the construction of the F93B aircraft were produced by the following students:

- Mr Caron - Fin and Tailplane
- Mr McDougall - Centre and rear fuselage
- Miss Johansson - Nose fuselage
- Mr Buller - Wings
- Mr Roe + Mrs Gray - Engines
- Mr Genon - Main undercarriage
- Mr Hacker - Nose undercarriage

- Mr Buller + Mrs Gray - Fuselage to wing fillet
 - Undercarriage blister
 - Puma model

This Chapter outlines the construction techniques used to develop the components used for the F93B aircraft and discusses the process required to prepare the completed components for the rapid prototyping process called stereolithography.

Fuselage Model

2.3 The fuselage section was constructed in two parts: the nose and centre fuselage and merged together by the CAD group leaders. The geometry for the fuselage was taken from the design specification F93-3. Several frames along the fuselage were defined using Cartesian co-ordinates which were then used to model Bezier splines. The

splines were joined together using guide curves and the surface between the frames was created using ruled surfaces by using the the splines and curves as bounding entities. In most cases it was easier to create one half of the fuselage section and mirror image it over to create the other side rather than try to create the whole surface at one time. The nose fuselage was a little more difficult because of the windscreen section and reference should be made to Miss Johansson's project thesis for details of how the windscreen was constructed.

Wing Model

2.4 The wing model was constructed by defining the geometry of the supercritical aerofoil, developed at Cranfield, in Cartesian co-ordinates which were used to model the aerofoil curves in Bezier splines. The section was then transformed and scaled with the appropriate chord lengths and thickness ratios in accordance with Drawing F93-4. Ruled surfaces were used to model the wing surface as shown in Figure 2.1. The wing root section was also modelled using ruled surfaces between the port and starboard wing sections. The surface edit detail for the wing root is shown in Figure 2.2.

Wing Root/Fuselage Fillet

2.5 The wing root fillet is required to provide an aerodynamic interface between the wing root and the fuselage. In order to cope with the double curvature required the fillet was constructed in 6 separate segments. The extremities of the fillet were obtained from Drawing F93B-1 and the boundaries of the fairing were defined by projecting guide curves onto the fuselage and wing surfaces to ensure a blended interface. The surface edit detail for the fillet is shown in Figure 2.2 and the shaded image of the fillet is shown in Figure 2.3

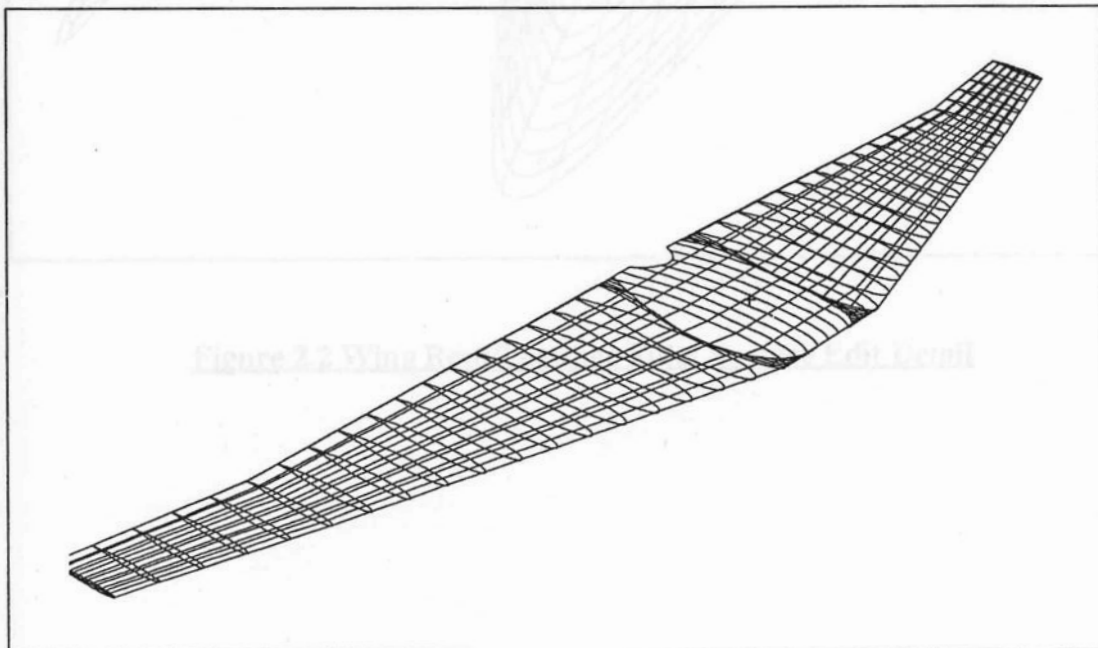


Figure 2.1 Wing Model Using Ruled Surfaces

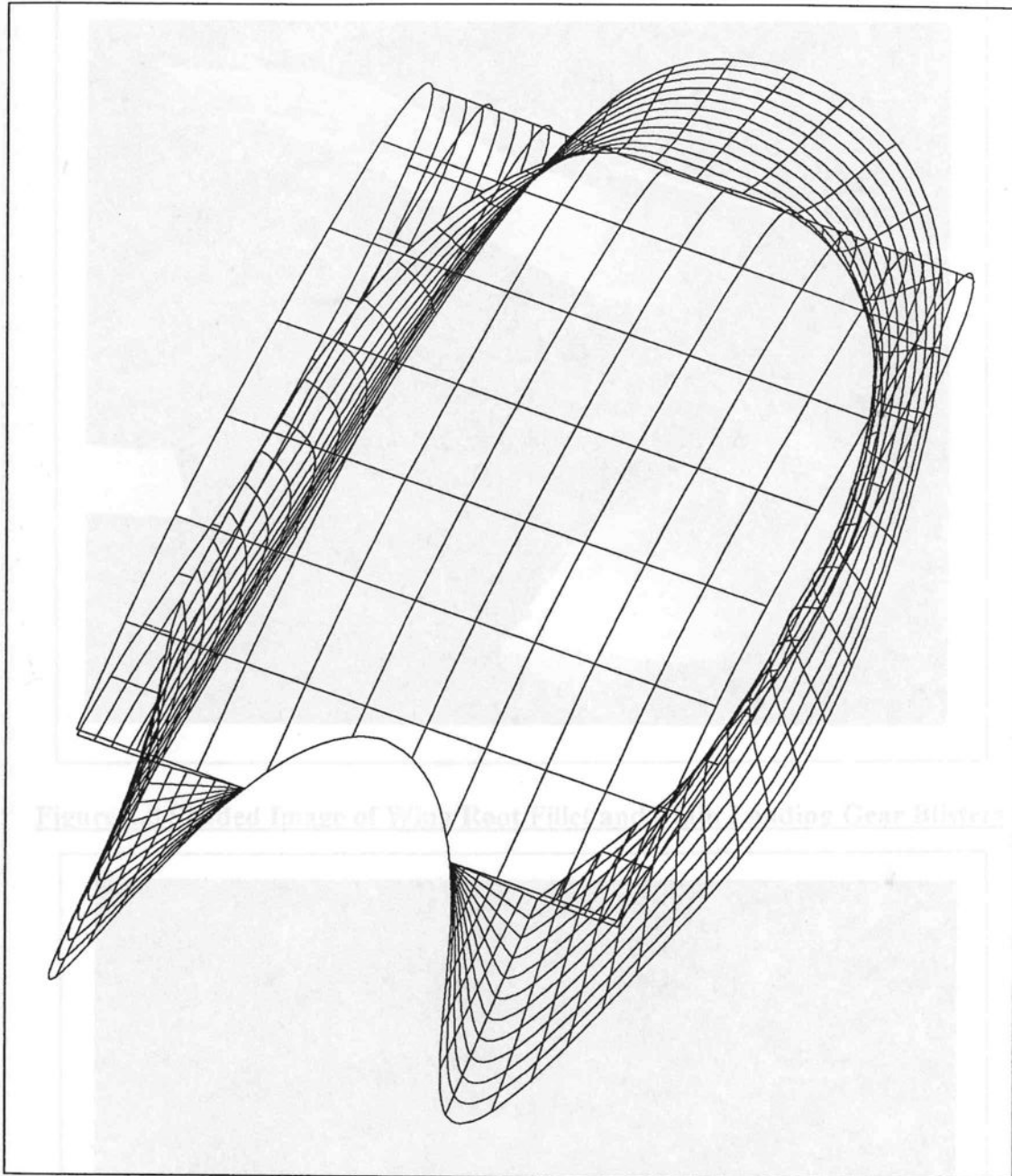


Figure 2.2 Wing Root/Fuselage Fillet Surface Edit Detail

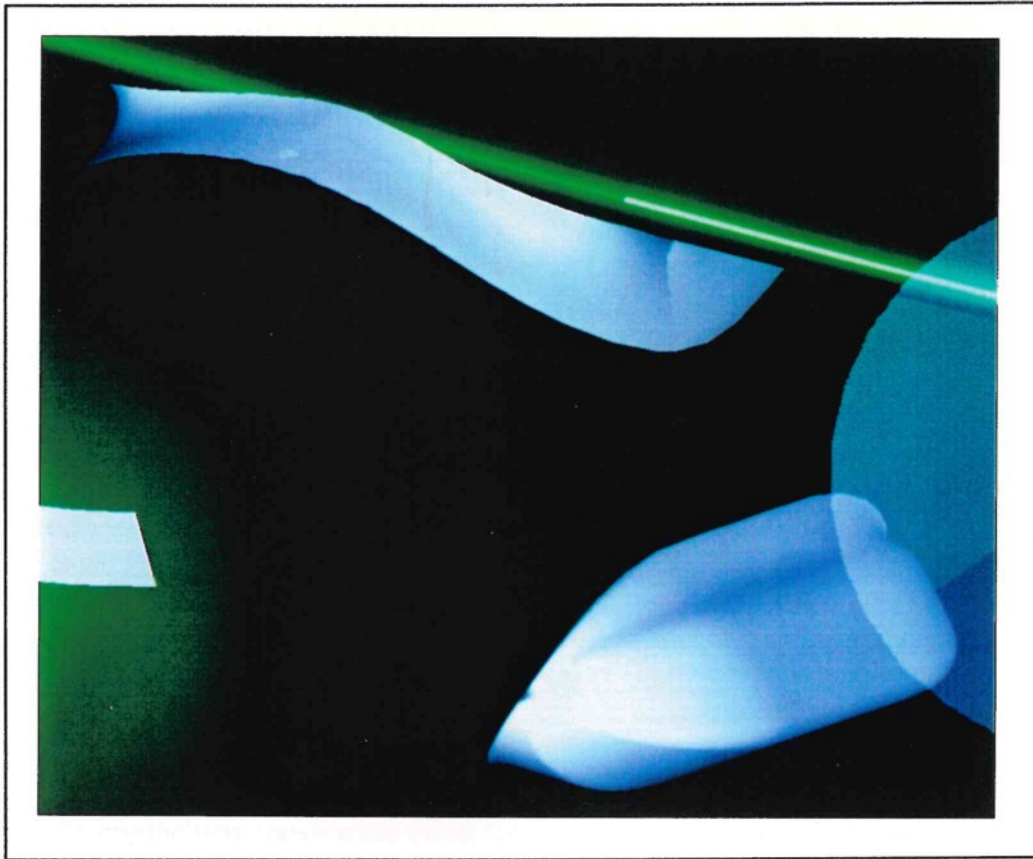


Figure 2.3 Shaded Image of Wing Root Fillet and Main Landing Gear Blisters

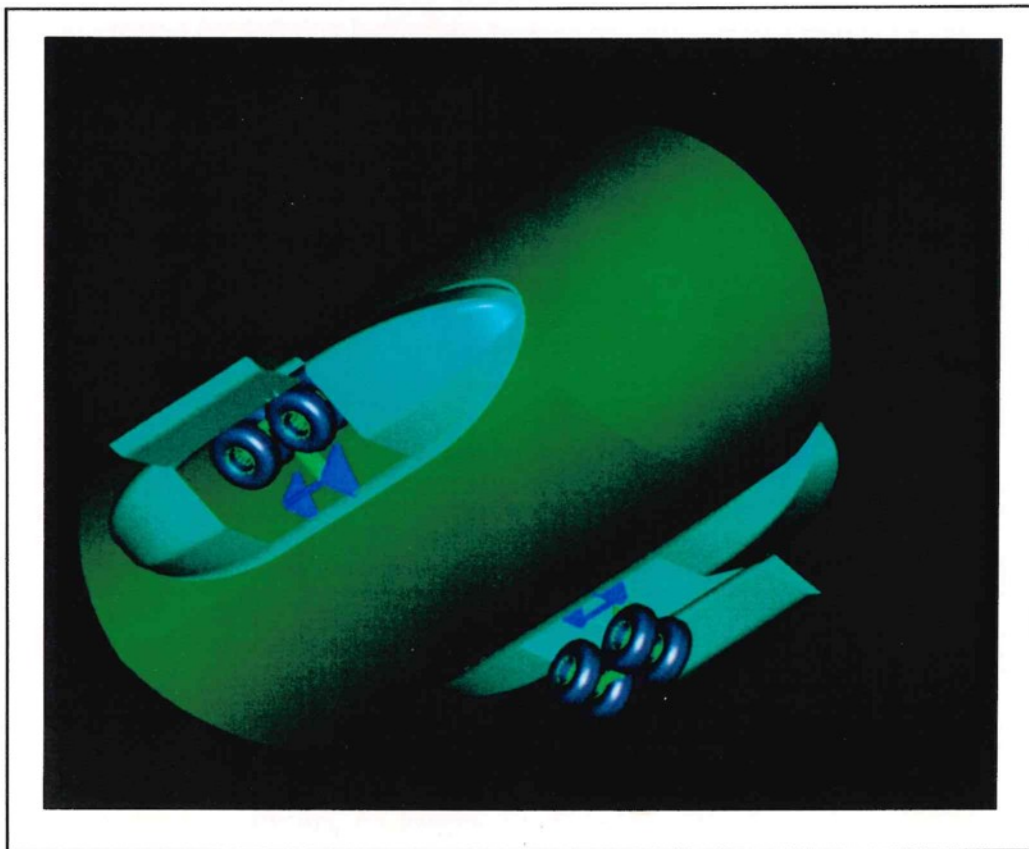


Figure 2.4 Main Landing Gear in Retracted and Extended Position

Main Landing Gear Blisters

2.6 The main landing gear blisters were constructed from a cross-sectional profile which was provided by Mr Genon. The extremities of the blisters were established using Drawing F93B-1 and elliptical profiles were used to simplify the surface modelling. By simplifying the profile curves lower order degree polynomials were used to model the surfaces which made the surfaces more pliable and able to follow the control surfaces more closely. The complex double curvature of the blisters meant that they had to be constructed in upper and lower surfaces using a curve mesh surface. Figure 2.3 and 2.4 show the shaded image of the complex double curvature surface required for the main landing gear blister. The main and nose landing gear models generated by Mr Hacker and Mr Genon respectively were constructed of solids; although, the files were not merged into the aircraft model because their size inhibited the processing speed the blister surfaces were edited to show the extended and retracted positions of the main landing gear. This also established that the basic fit of the main landing gear and the APU inside the blister was acceptable. Figure 2.4 shows the shaded image of the main landing gear in the extended and retracted position. Further details on the landing gear modelling can be obtained from Mr Hacker or Mr Genon.

Engines

2.7 The engine was constructed using cross-sections defined in Cartesian co-ordinates from Mr Roe. The Bezier splines joining the Cartesian co-ordinates formed the basis for the ruled surfaces. The engine was created in 5 seperate sections: the nosecone, the propeller disc, the engine intakes, the engine casing and the exhaust. Each section was created by defining half of the surface and producing a mirror image for the other side of the section. Figure 2.5 shows the surface edit detail for the engine and Figure 2.6 shows the shaded image of the engines.

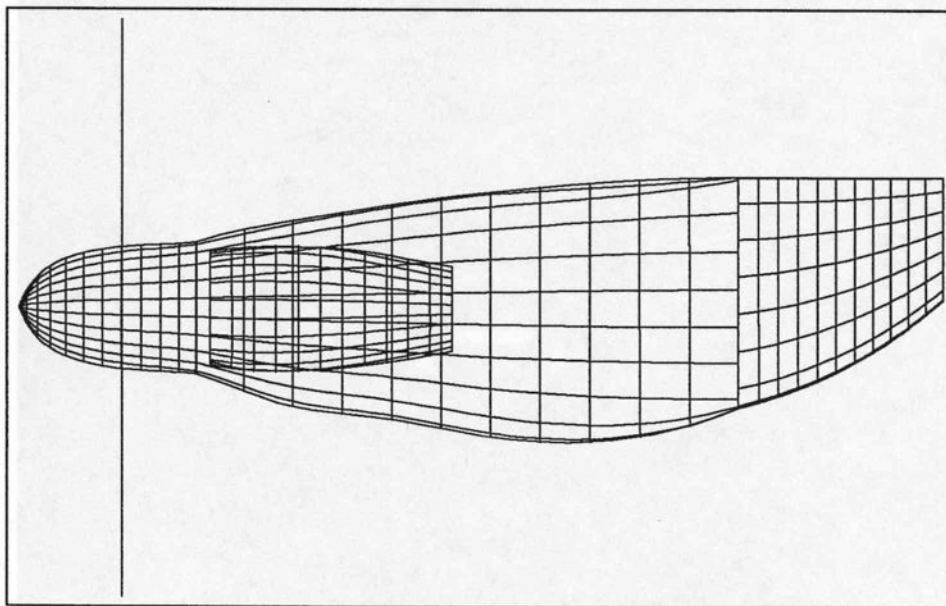


Figure 2.5 Engine Surface Edit Detail

Figure 2.7 Head on View of 'Gemini'

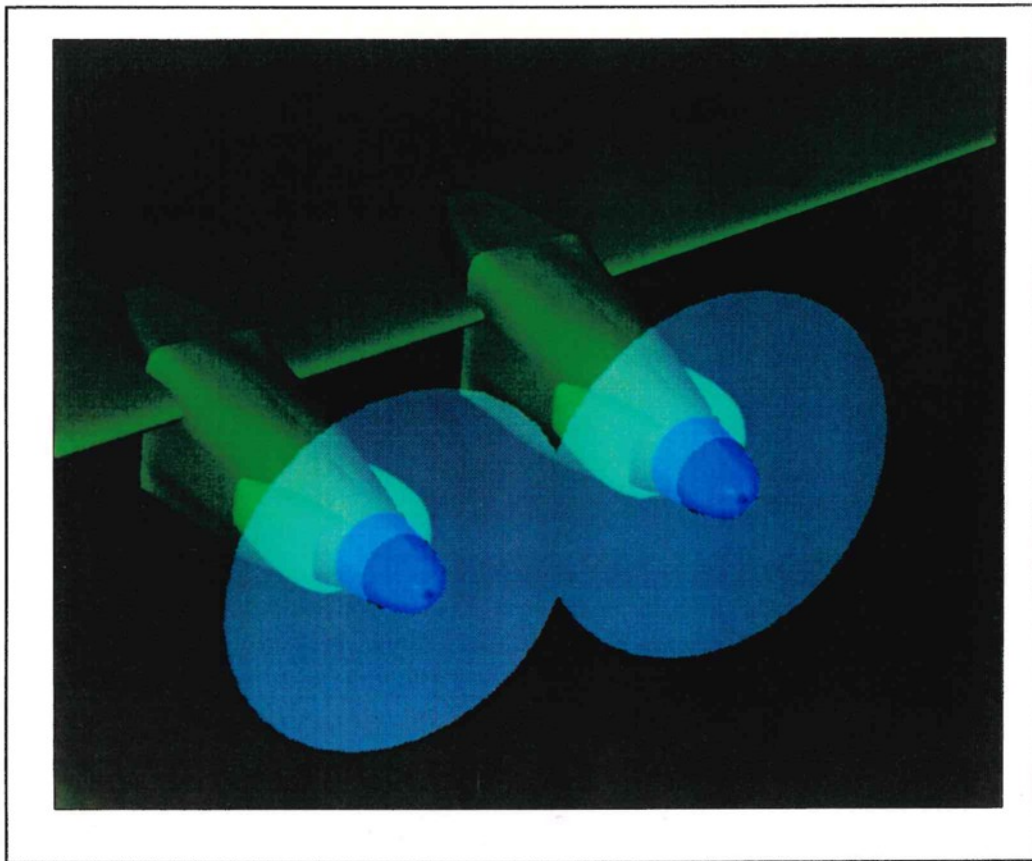


Figure 2.6 Shaded Image of the Engines

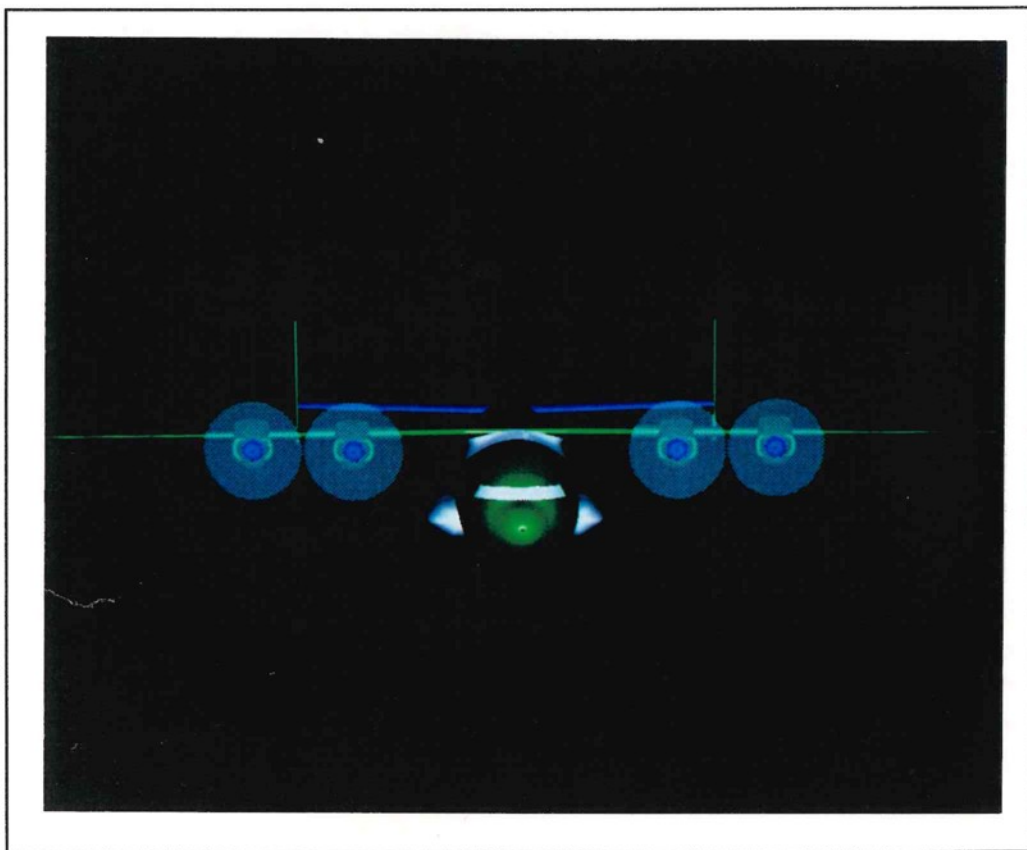


Figure 2.7 Head on View of 'Gemini'

Tailplane and Fins

2.8 The tailplane and fins were constructed in a similar manner to the wing using the Cartesian co-ordinates of a 10% NACA 63A010 aerofoil to model the aerofoil curves using Bezier splines and ruled surfaces. Figure 2.7 shows the breadth and height of the tailplane and fins compared with the wing structure.

Super Puma Model

2.9 One of the major design criteria was that the F93 should be able to carry a Eurocopter Super Puma without having to remove any of the major components. Therefore a Super Puma model was constructed to check the internal dimensions of the freight bay. The main body of the model was constructed by creating cross-sections using Cartesian co-ordinates from data defined in Ref (14). The surfaces were created as ruled surfaces and components such as the main rotor head, wheels, sponsons and the tailplane were created as separate surfaces. The model was inserted inside the F93B and Figure 2.8 shows that the length and breadth of the freight bay are adequate to carry the Super Puma. However Figure 2.9 shows that although there is adequate room beneath the wing root carry through box for the main rotor head, the tail rotor disc of the Super Puma protrudes through the roof of the freight bay. Therefore the tail rotors would have to be carefully aligned whilst manoeuvring the helicopter into place or the tail rotor blades would have to be removed altogether.

STEREOLITHOGRAPHY

2.10 Stereolithography (STL) is one of 3 rapid prototyping systems now used fairly widely within the aircraft industry. The other 2 being Selective Laser Sintering (SLS) and Laminated Object Modelling (LOM). The Stereolithography technique fabricates a scaled model of the aircraft by using a CAD/CAM interface called the Stereolithography Interface Apparatus (SLA) which is read by a converter that slices the model into a stack of 2 dimensional boundaries. A laser beam directed by the converter traces the 2 dimensional boundaries out on the surface of liquid epoxy resin. The layers are built up one by one and cured using an ultra-violet beam thus producing a 3 dimensional solid model of about 125mm x 125mm.

2.11 In order to use the SLA interface the aircraft model has to be converted into an SLA file. This is best achieved by splitting the aircraft into several components as the DEC stations produce memory exhaustion errors when the files are too large. Those components that could be used to produce mirror images for the other side of the aircraft such the right-hand wing, tailplane and fin, reduced the number of components required however, there were still more than 16 separate components required for the F93B aircraft. To create an SLA file from a component the surfaces must be changed into enclosed volumes so that a 3 dimensional boundary can be defined. Defining the volume involves specifying an adjacency and triangulation tolerance. The adjacency tolerance determines the accuracy within which the surfaces that make up the volume meet up. A tolerance of 0.1mm is used for this SLA. The triangulation tolerance

Figure 2.9 Tail Rotor Disc Protruding through the Freight Bay Ceiling

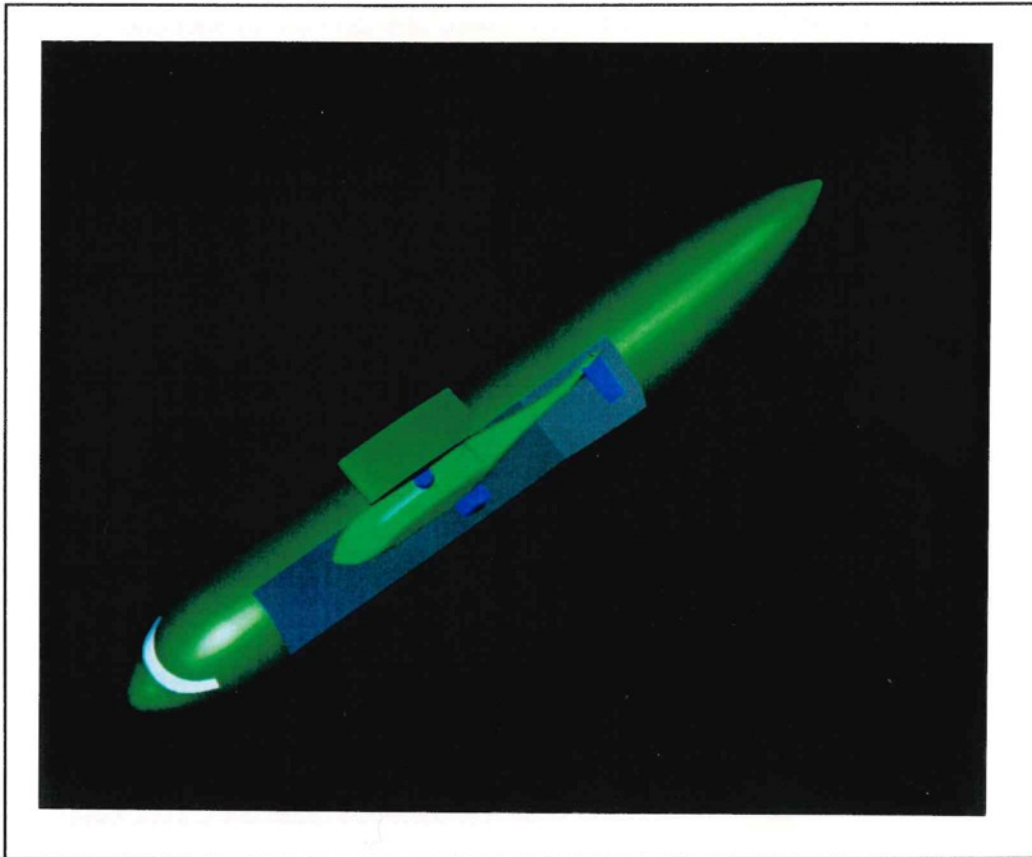


Figure 2.8 Shaded Image showing the Super Puma fitted into the Freight Bay

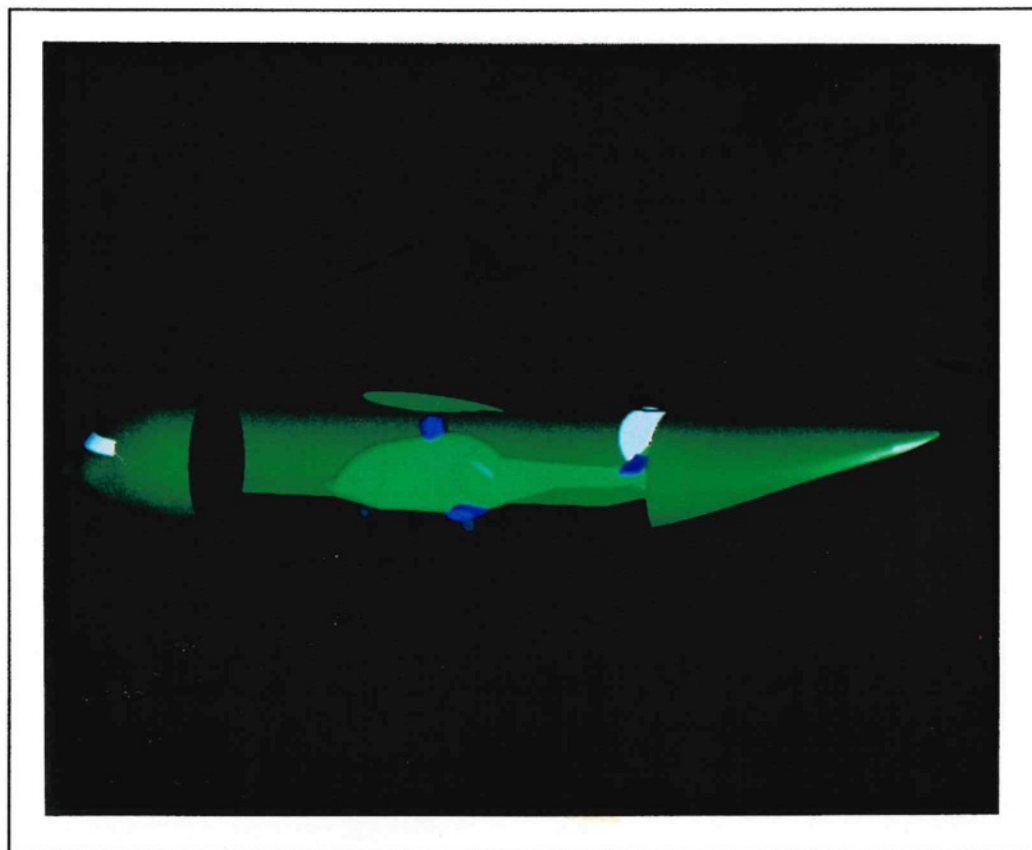


Figure 2.9 Tail Rotor Disc Protruding through the Freight Bay Ceiling

determines how accurately the surface triangles approximate the component surfaces. A triangulation tolerance of 0.05mm was used for most of the components but this was too fine for some of the engine components which were reduced to a tolerance of 0.5mm. Figure 2.10 shows the fuselage sections as enclosed volumes for the SLA interface.

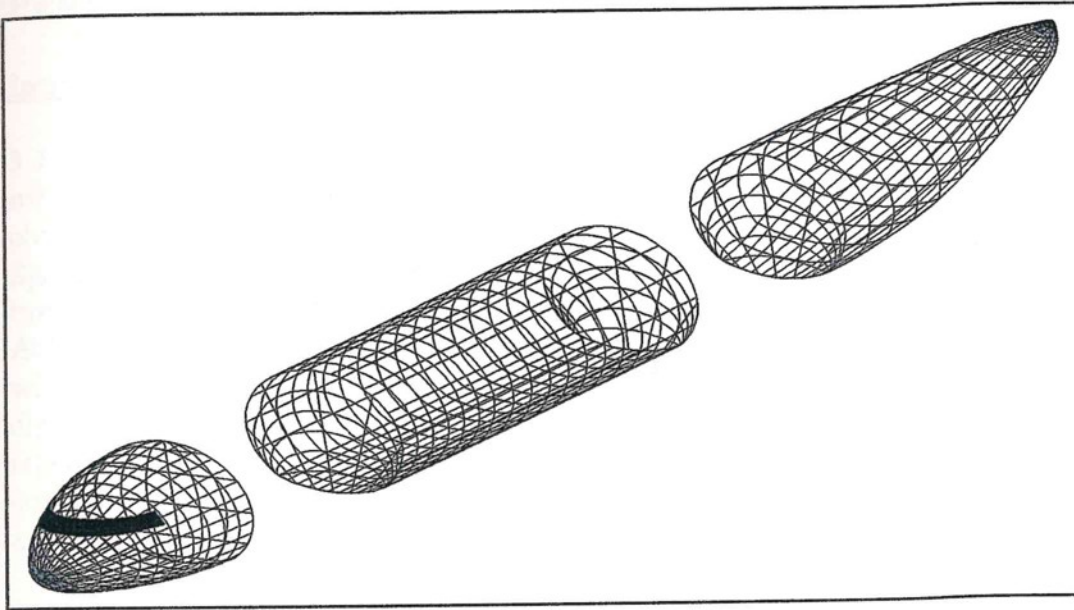


Figure 2.10 Fuselage Sections as Enclosed Volumes for SLA Interface

Figure 2.11 shows the resultant stereolithographic 3 dimensional aircraft model

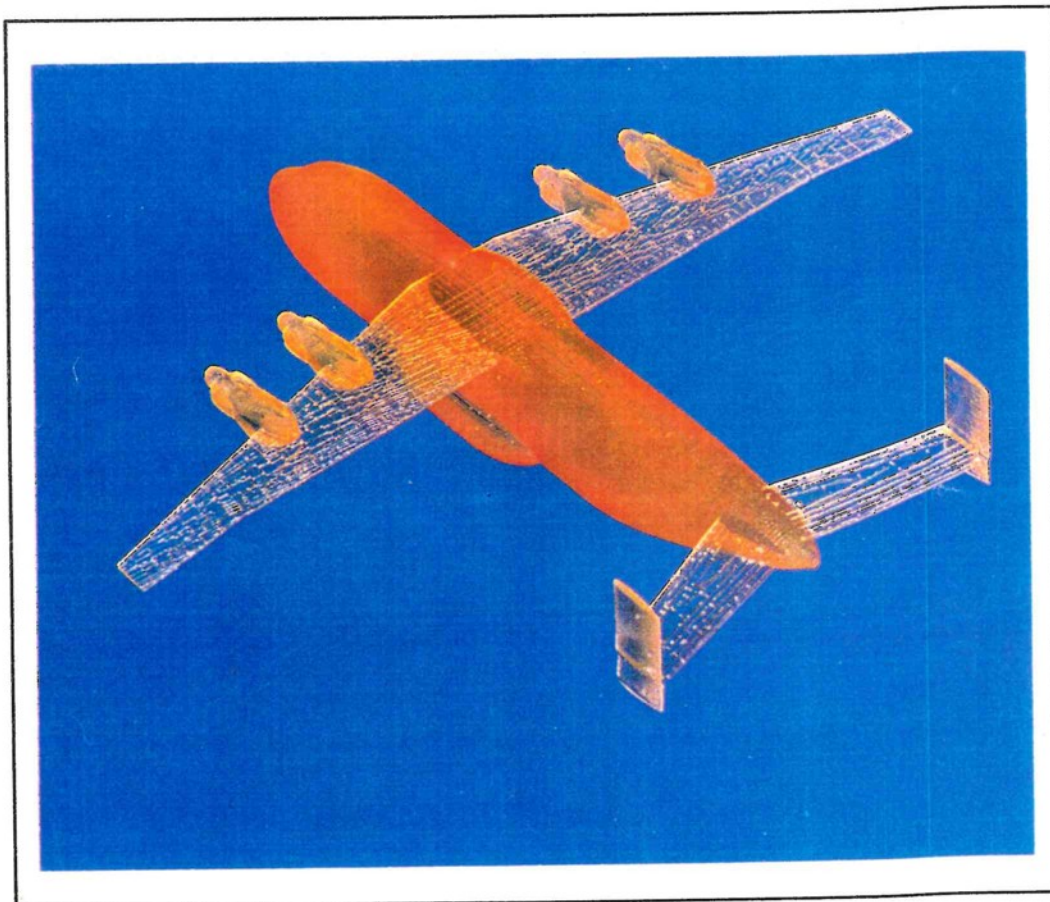


Figure 2.11 Stereolithographic 3 Dimensional Model

CHAPTER 3

PERFORMANCE

Introduction

3.1 This chapter assesses the performance of the F93B aircraft by looking at its take-off, landing, climb, cruise and descent capabilities. In particular, formulae are derived to check whether the F93B can achieve the design specifications required in Ref (1). Specifically, that it should be capable of taking off at maximum AUM from a 1500m runway as well as take-off from a semi-prepared strip 900m in length, albeit at reduced AUM. Additionally, the final payload/range diagram is established in order to assess whether the F93B can achieve the design requirement to carry 20000kg payload over a distance of 5500km with a 1250km reserve capability. As the F93B is intended as a replacement for the C-130 Hercules, where possible their performance figures are compared although consideration should be given to the fact that the C-130 Hercules is a smaller aircraft.

TAKE-OFF PERFORMANCE

3.2 Analysing the take-off distance requires an assessment of 3 different segments: the ground run distance, the distance between rotation and lift-off and the airborne distance to a screen height of 10.7m (35ft). An initial estimate of 720m for the ground run distance was established using Ref (2); however, this value neglects all drag and runway resistance which increases the ground run required. Therefore from Ref (3) Eqn (3.1) was used to calculate a more accurate figure by taking into account the thrust (T), drag (D), lift (L), weight (W) and runway resistance μ_R .

$$S_{GR} = \frac{1}{2} m (\Delta v)^2 \sum \frac{(2n-1)}{X_G} \quad \text{Eqn 3.1}$$

where $X_G = T - D - \mu_R(W - L)$

The revised value for the ground run distance at a maximum AUM of 101500kg is 858m which is a 20% increase in runway length. The distance from rotate to lift-off is based on the assumption that it takes approximately 3 seconds for an aircraft to achieve total lift-off. Therefore, by calculating the average velocity the distance at maximum AUM is 189m. The airborne distance to screen height was calculated using Eqn 3.2 from Ref (3).

$$S_a = (2Rh_{sc})^{\frac{1}{2}} \quad \text{Eqn 3.2}$$

where R is the radius of the circular arc described by the aircraft flight path during transition to its climb angle γ_C . The distance to screen height at maximum AUM is 328m.

Maximum Take-off Distance Required = 1376m

See Appendix A for the detailed calculations. Therefore the F93B satisfies the design requirement in that it requires less than 1500m to take-off at maximum AUM. Figure 3.1 shows the variation in take-off distance required for different AUM.

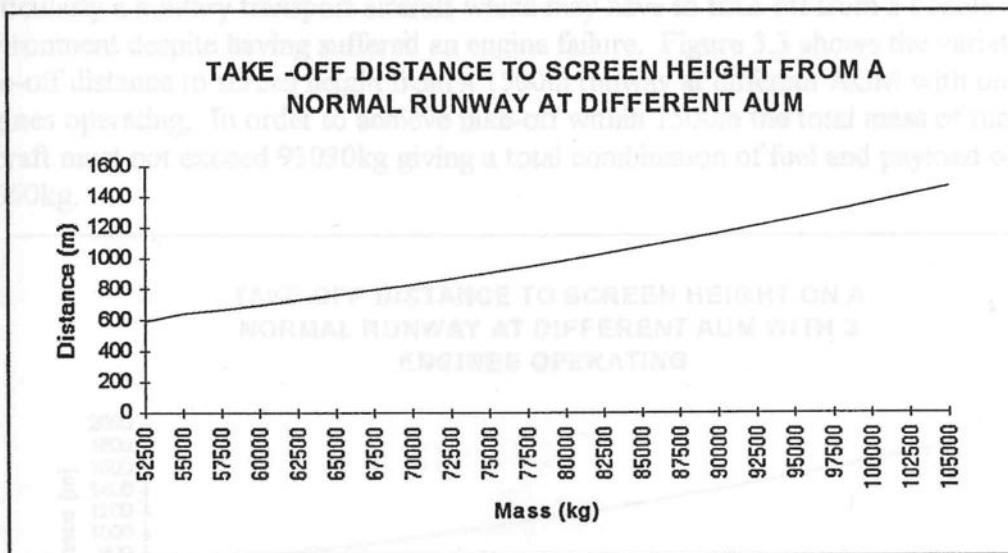


Figure 3-1

The additional requirement that the F93B should be able to take-off from a semi-prepared strip 900m in length is illustrated in Figure 3.2 which shows the variation in ground run, not including the distance to screen height, required for different AUM with an increased runway resistance of 0.03.

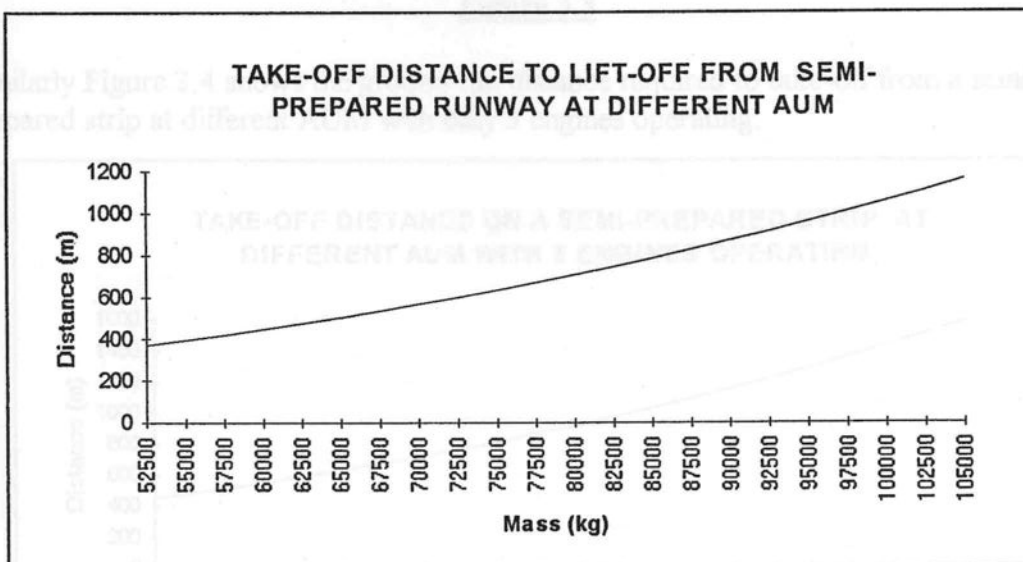


Figure 3.2

The maximum AUM permissible in order to lift-off within 900m is 91800 kg which means the maximum fuel and payload combination is 39430 kg.

3-Engine Performance

3.3 The 3 engine performance is an important characteristic of an aircraft, particularly a military transport aircraft which may have to take-off from a hostile environment despite having suffered an engine failure. Figure 3.3 shows the variation in take-off distance to screen height from a 1500m runway at different AUM with only 3 engines operating. In order to achieve take-off within 1500m the total mass of the aircraft must not exceed 93030kg giving a total combination of fuel and payload of 40660kg.

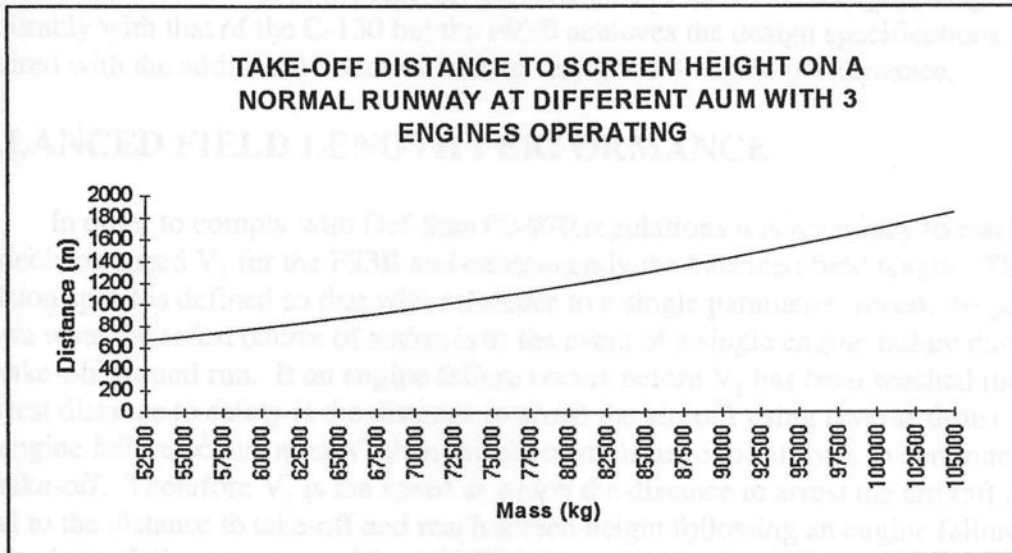


Figure 3.3

Similarly Figure 3.4 shows the ground run distance required to take-off from a semi-prepared strip at different AUM with only 3 engines operating.

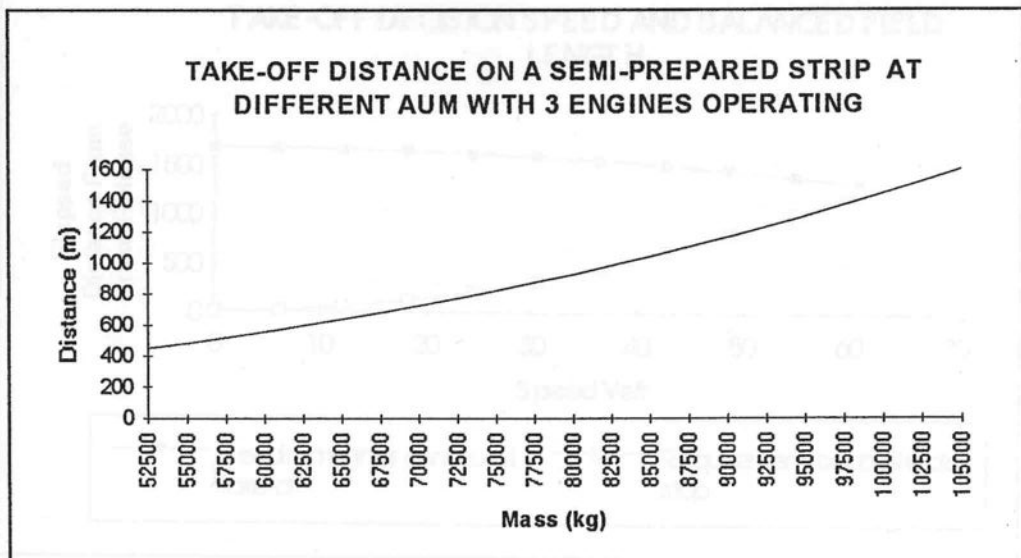


Figure 3.4

In order to take-off from a semi-prepared strip 900m in length, distance to screen height excluded, the aircraft AUM must not exceed 78500kg giving a total fuel and payload combination of 26130kg.

Comparison with C-130 Hercules

3.4 Although the C-130 Hercules is a smaller aircraft with a maximum take-off mass of 79380kg it requires a ground run of 1091m compared with 1047m for the F93B and the take-off distances to screen height at maximum AUM are 1573m and 1376m respectively. Not only does the take-off performance of the F93B compare very favourably with that of the C-130 but the F93B achieves the design specifications required with the additional bonus of having very good 3 engine performance.

BALANCED FIELD LENGTH PERFORMANCE

3.5 In order to comply with Def Stan 00-970 regulations it is necessary to establish the decision speed V_1 for the F93B and consequently the balanced field length. The decision speed is defined so that with reference to a single parameter, speed, the pilot knows what the safest course of action is in the event of a single engine failure during the take-off ground run. If an engine failure occurs before V_1 has been reached then the shortest distance to safety is the distance to arrest the aircraft using reverse thrust. If the engine failure occurs after V_1 then the shortest distance to safety is to continue with the take-off. Therefore V_1 is the speed at which the distance to arrest the aircraft is equal to the distance to take-off and reach screen height following an engine failure. The horizontal distance covered from brakes release to reaching safety, either by arresting the aircraft or continuing with the take-off, following an engine failure at V_1 is known as the balanced field length. Appendix A contains the detailed calculations required to establish Figure 3.5.

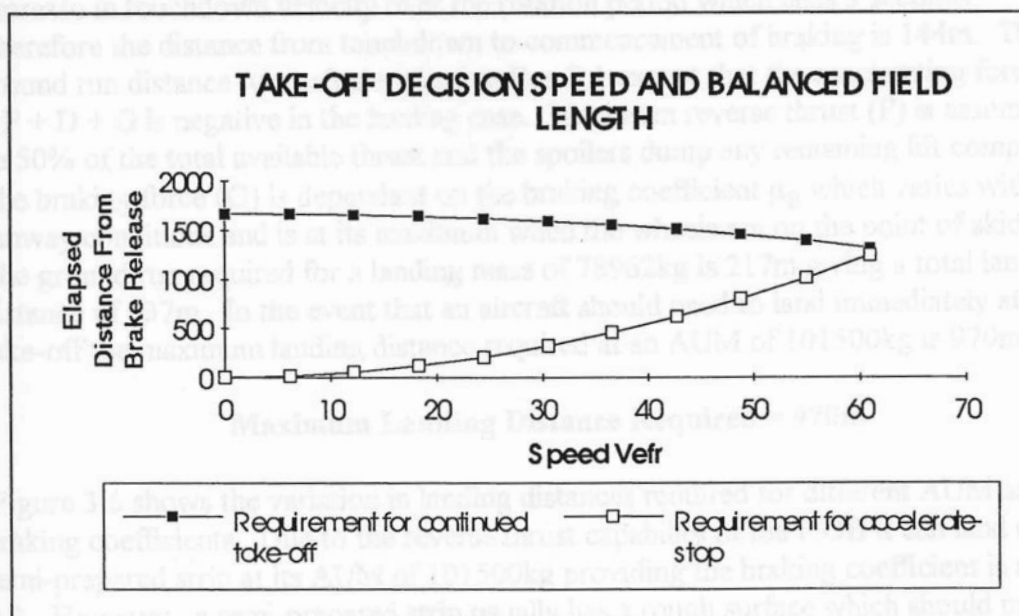


Figure 3.5

The decision speed and balanced field length occur where the graph for the continued take-off and the graph for the accelerate/stopping distance intersect. However, Figure 3.5 shows that in this case the graphs intersect at approximately the same speed as rotation occurs which is 61m/s. This is due partly to the assumption that the F93B has a 50% reverse thrust capability and also from neglecting the time delay between recognition of the engine failure and the reverse thrust becoming effective. Both factors would tend to increase the distance required to arrest the aircraft and cause the graphs to intersect at a lower speed. From Chapter 1 para 1.7 Def Stan 00-970 states that V_R should be not less than V_1 . Therefore the F93B is marginally acceptable regarding its decision speed taking into account the assumptions made during the calculations which would tend to reduce the decision speed, thus, making it more acceptable according to the Def Stan requirement.

LANDING PERFORMANCE

3.6 Analysing the landing performance also requires an assessment of 3 different segments similar to those assessed for the take-off performance: the landing approach at an angle (γ_A) of 3° to the horizontal which includes the flare in the flight path to reduce the vertical component of velocity just before touchdown, the period of rotation from touchdown of the main landing gear to touchdown of the nose landing gear and the ground run distance when reverse thrust and brakes are effective. Eqn 3.3 from Ref (3) was used to calculate the airborne distance from screen height to touchdown.

$$S_a = \frac{h_{sc}}{\gamma_A} + \frac{1}{2} R \gamma_A \quad \text{Eqn 3.3}$$

The airborne distance from screen height to touchdown at an AUM of 78962kg is 276m. The distance covered during rotation was calculated by assuming a 1.5% decrease in touchdown velocity over the rotation period which lasts 3 seconds. Therefore the distance from touchdown to commencement of braking is 144m. The ground run distance was calculated using Eqn 3.1 except that the accelerating force $X_G = F + D + G$ is negative in the landing case. Maximum reverse thrust (F) is assumed to be 50% of the total available thrust and the spoilers dump any remaining lift component. The braking force (G) is dependant on the braking coefficient μ_B which varies with the runway conditions and is at its maximum when the wheels are on the point of skidding. The ground run required for a landing mass of 78962kg is 217m giving a total landing distance of 637m. In the event that an aircraft should need to land immediately after take-off the maximum landing distance required at an AUM of 101500kg is 970m.

Maximum Landing Distance Required = 970m

Figure 3.6 shows the variation in landing distances required for different AUM and braking coefficients. Due to the reverse thrust capability of the F93B it can land on a semi-prepared strip at its AUM of 101500kg providing the braking coefficient is above 0.3. However, a semi-prepared strip usually has a rough surface which should prevent skidding occurring at a braking coefficient of less than 0.3.

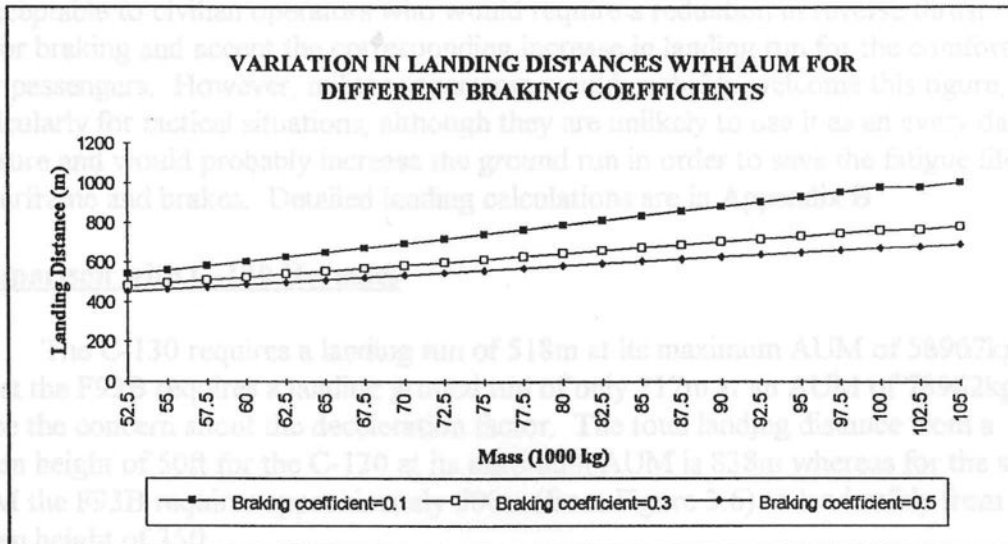
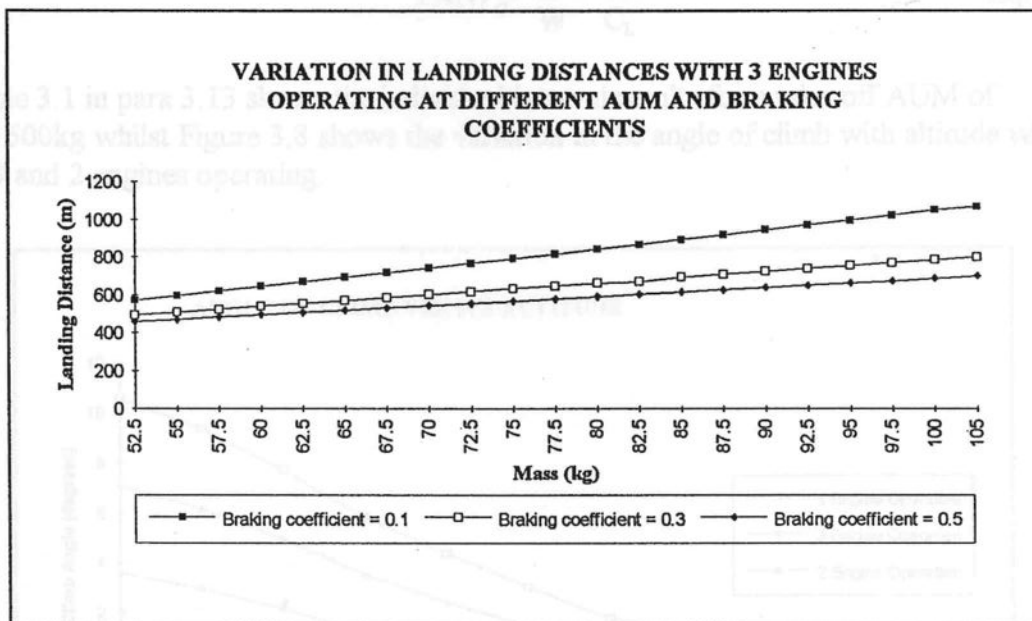
**Figure 3.6**

Figure 3.7 shows the landing distances required with only 3 engines operating in reverse thrust. The landing distances are slightly increased but are still within the take-off capabilities which means that if the F93B can take-off from the runway then it must be able to land back there if necessary.

**Figure 3.7**

Although there was no design specification laid down for the landing distances the short distances involved caused some concern over the deceleration experienced by the passengers and crew whilst braking the aircraft. At a point (0.7VB) during the ground run the deceleration was 5.08 m/s or 0.52g. It was felt that this deceleration may be

unacceptable to civilian operators who would require a reduction in reverse thrust and/or braking and accept the corresponding increase in landing run for the comfort of their passengers. However, military operators would probably welcome this figure, particularly for tactical situations, although they are unlikely to use it as an every day measure and would probably increase the ground run in order to save the fatigue life on the airframe and brakes. Detailed landing calculations are in Appendix B.

Comparison with C-130 Hercules

3.7 The C-130 requires a landing run of 518m at its maximum AUM of 58967kg whilst the F93B requires a landing ground run of only 217m at an AUM of 78962kg, hence the concern about the deceleration factor. The total landing distance from a screen height of 50ft for the C-130 at its maximum AUM is 838m whereas for the same AUM the F93B requires approximately 500m (from Figure 3.6) to land safely from a screen height of 35ft.

CLIMB PERFORMANCE

Climb Angle

3.8 The climb angle was calculated over 5000ft intervals using Eqn 3.4:

$$\tan \gamma_c = \frac{T}{W} - \frac{C_D}{C_L} \quad \text{Eqn 3.4}$$

Table 3.1 in para 3.13 shows the individual interval results for a take-off AUM of 101500kg whilst Figure 3.8 shows the variation in the angle of climb with altitude with 4, 3 and 2 engines operating.

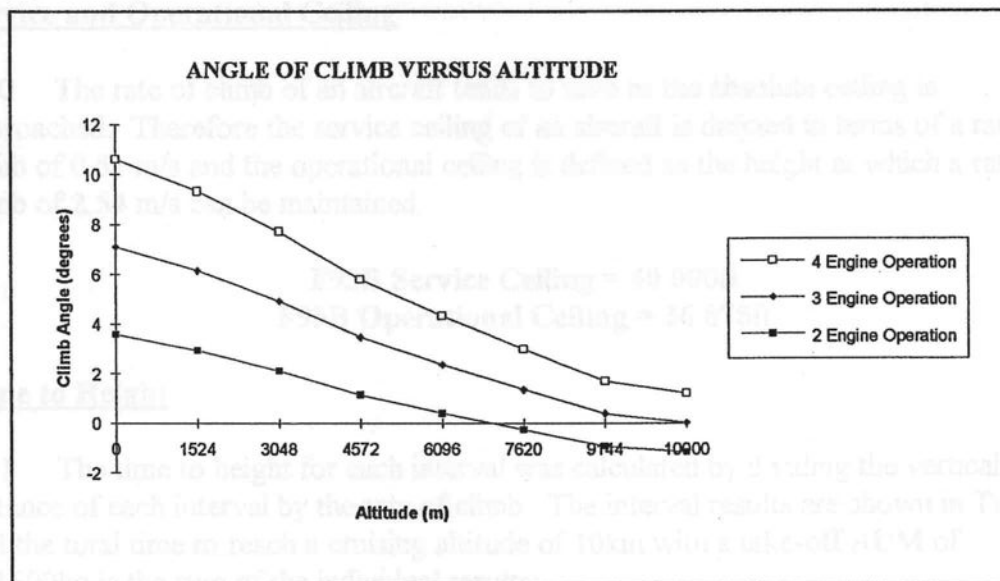


Figure 3.8

Rate of Climb

3.9 The rate of climb for each interval was calculated using Eqn 3.5:

$$V_c = \frac{(T-D)V_{TR}}{W} \quad \text{Eqn 3.5}$$

The individual interval rate of climb results for a take-off AUM of 101500kg are shown in Table 3.1 para 3.13 and Figure 3.9 shows the variation in the rate of climb with altitude with 4, 3 and 2 engines operating.

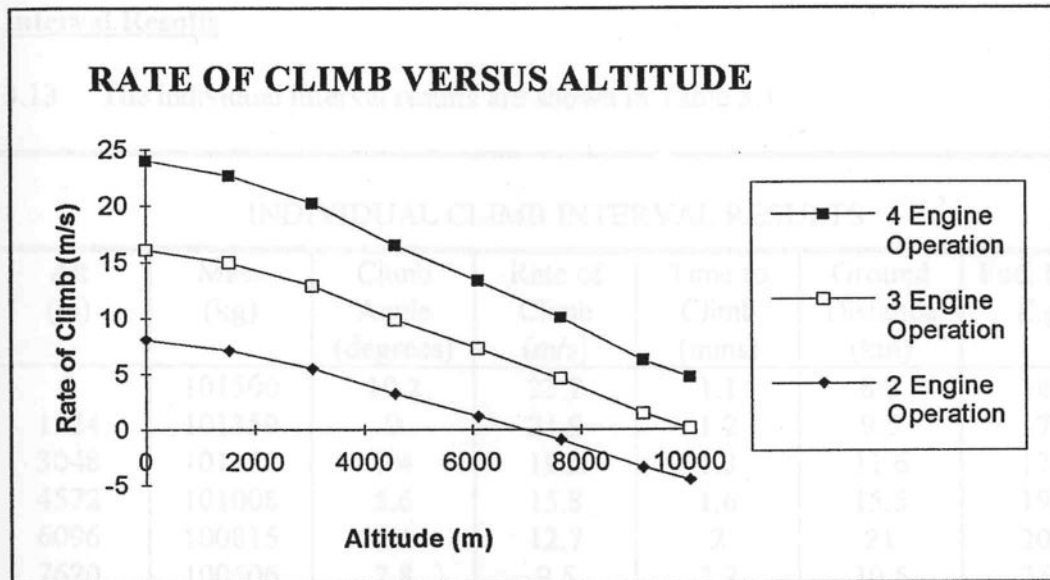


Figure 3.9

Service and Operational Ceiling

3.10 The rate of climb of an aircraft tends to zero as the absolute ceiling is approached. Therefore the service ceiling of an aircraft is defined in terms of a rate of climb of 0.51 m/s and the operational ceiling is defined as the height at which a rate of climb of 2.54 m/s can be maintained.

F93B Service Ceiling = 40 000ft
F93B Operational Ceiling = 36 875ft

Time to Height

3.11 The time to height for each interval was calculated by dividing the vertical distance of each interval by the rate of climb. The interval results are shown in Table 3.1 and the total time to reach a cruising altitude of 10km with a take-off AUM of 101500kg is the sum of the individual results:

Total Time to reach Cruising Altitude of 10km at AUM = 12.5 minutes

Fuel Consumption

3.12 The fuel consumed during each interval was the product of the SFC, thrust and the time period and the total fuel used during the climb was the sum of the results:

Fuel Consumed during Climb at 250KEAS = 1346kg

Therefore the mass of the F93B at the beginning of the cruise phase is 101500kg - 1346kg = 100154kg.

Interval Results

3.13 The individual interval results are shown in Table 3.1

INDIVIDUAL CLIMB INTERVAL RESULTS						
Alt (m)	Mass (kg)	Climb Angle (degrees)	Rate of Climb (m/s)	Time to Climb (mins)	Ground Distance (km)	Fuel Used (kg)
0	101500	10.2	23.2	1.1	8.3	141
1524	101359	9	21.9	1.2	9.5	177
3048	101182	7.4	19.5	1.3	11.6	174
4572	101008	5.6	15.8	1.6	15.5	193
6096	100815	4.1	12.7	2	21	209
7620	100606	2.8	9.5	2.7	30.5	253
9144	100353	1.6	5.8	2.4	31	199
10000	100154	-	-	-	-	-
TOTAL				12.3mins	127.4km	1336kg

Table 3.1

Comparison with C-130 Hercules

3.14 The C-130 has a maximum rate of climb at sea level of 9.65 m/s compared to 23.2 m/s for the F93B. This vast increase in climb capability is due to the modern engines installed on the F93B. The service ceiling for the C-130 at maximum AUM is 33 000ft whilst the F93B is capable of operating up to a service ceiling of 40 000ft but would not normally operate above 38 875ft which is quite a significant improvement on the Hercules and allows the F93B to operate in a wider variety of airways.

CRUISE PERFORMANCE

3.15 The cruise performance was based on a cruise speed of Mach 0.7 (210 m/s TAS). The design range for the F93B was 5500km + 1250km in reserve therefore the cruise distance was approximately 5500km - 127km = 5373km. In the cruise condition where thrust equals drag and lift equals weight the fuel consumed was calculated over

several intervals to take account of the change in drag, therefore thrust, due to the reduction in aircraft mass. See Appendix C para 8 for the detailed cruise calculations. The total fuel used over the design range is 20843kg.

Fuel consumed over Design Range(5373km) = 20 843kg

Therefore, the mass of the F93B at the start of the descent phase is 100154kg - 20843kg = 79311kg and the duration of the design cruise phase is 6.9 hrs.

DESCENT PERFORMANCE

3.16 The descent performance was analysed using exactly the same equations as the climb performance: however, the major difference in calculating the descent performance was that the engines no longer provide anywhere near maximum thrust. During the descent phase the engines are usually throttled back to provide only enough power for the secondary systems which was assumed to require as little as 2% of the maximum available thrust. It was also assumed that the F93B descent was performed at a constant 250KEAS. The individual descent interval results are shown in Table 3.2.

INDIVIDUAL DESCENT INTERVAL RESULTS						
Alt (m)	Mass (kg)	Descent Angle (degrees)	Rate of Descent (m/s)	Time to Descend (mins)	Ground Distance (km)	Fuel used (kg)
10000	79311	3.8	13.1	1.1	13.7	14
9144	79297	3.8	13.8	1.8	23	30
7620	79267	3.7	12.5	2	23.2	38
6096	79229	3.7	11.4	2.2	23.4	47
4572	79182	3.7	10.4	2.4	23.7	59
3048	79123	3.7	9.5	2.7	24	72
1524	79051	3.6	8.7	2.9	24.2	89
0	78962	-	-	-	-	-
TOTAL				18.4mins	155.2km	349kg

Table 3.2

Therefore for the design mission of 5500km the F93B requires a total of 22 538 kg of fuel and the mission duration is a total of 7.4 hours.

FATIGUE CYCLE DATA

3.17 In order to assess the structural fatigue of individual components certain performance characteristics were required as input variables to a fatigue cycle analysis method used by various structural groups. Ref (1) defined a flight pattern distribution containing 8 different mission lengths ranging from 1000km to 8000km. Following on

from the intervals used to calculate the climb, cruise and descent statistics the same intervals were used to derive the necessary information for the fatigue cycle information. There were 15 flight stages for each mission: 7 climb stages, 1 cruise stage and 7 descent stages. The altitude and mass of the F93B at the mid-position of each stage was required plus the horizontal distance covered along the ground. It was assumed that all the aircraft profiles were flown at a 250 KEAS climb and descent speed and a cruising speed of Mach 0.7. The detailed flight profiles are in Appendix D and for further information on the component structural fatigue the reader is recommended to those members of the F93B group project who designed a structural component.

OPERATIONAL CONFIGURATIONS

3.18. Table 3.3 shows various operational statistics for 3 different aircraft missions. The design configuration with 20000kg of payload and the maximum permissible fuel of 29130kg gives a range of 5975km without using the reserve fuel and a maximum of 7225km using all the reserve fuel. The design specification, Ref (1), requires the design payload to be carried a distance of 5500km with an additional 1250km reserve capability. Therefore the F93B not only achieves the design requirement but can carry the design payload an additional 475km.

AIRCRAFT CONFIGURATIONS			
	Design Payload and Fuel	Maximum Payload	Maximum Fuel
TAKE-OFF			
Payload	20000 kg	30000 kg	9130 kg
Fuel	29130 kg	19130 kg	40000 kg
AUM	101500 kg	101500 kg	101500 kg
RANGE			
Climb	127 km	127 km	127 km
Cruise	5848 km	3113 km	8809 km
Reserve	1250 km	1250 km	1250 km
Total	7225 km	4490 km	10186 km
DURATION			
Without using Reserves	8 hrs	4.3 hrs	11.8 hrs
Using Reserves	9.7 hrs	6 hrs	13 5 hrs

Table 3.3

The other configurations represent the 2 extremes of carrying the maximum payload or the maximum fuel load with a very small payload. When carrying its maximum payload the F93B can achieve a maximum range of 4490km which when compared with the C-130, currently in use with the RAF, which can carry a maximum payload of

20412kg a distance of approximately 4000km the F93B is a significant improvement. The second configuration is effectively the ferry range of the aircraft which is 10186km for the F93B whilst the C-130 has a ferry range of approximately 8750km.

RANGE VERSUS PAYLOAD DIAGRAM

3.19 The range versus payload diagram for the F93B is shown in Fig 3.10.

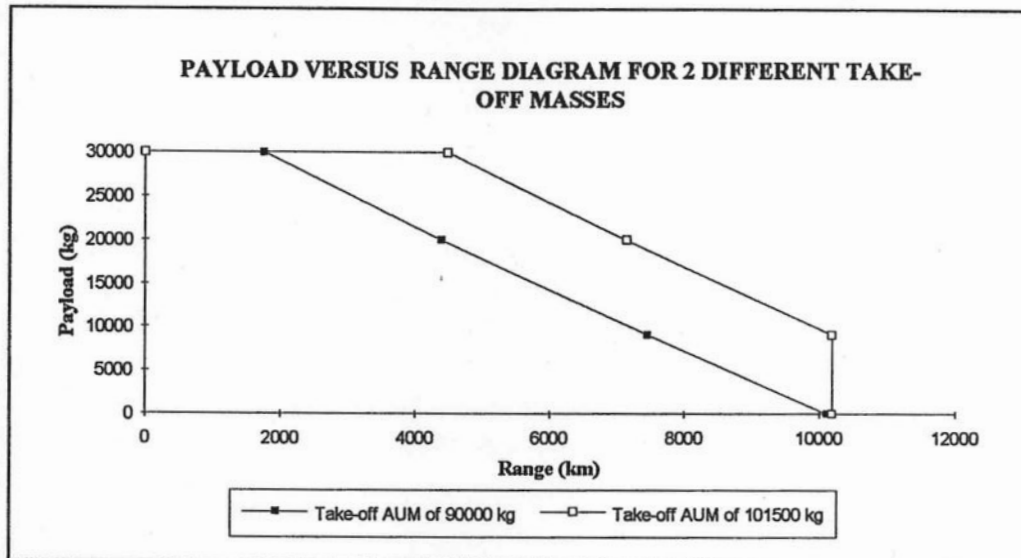


Figure 3.10

Ref (1) initially estimated that for a take-off mass of 90000kg the maximum payload could not be carried any distance. However Fig 3.10 shows that in fact the maximum payload can be carried just under 2000km.

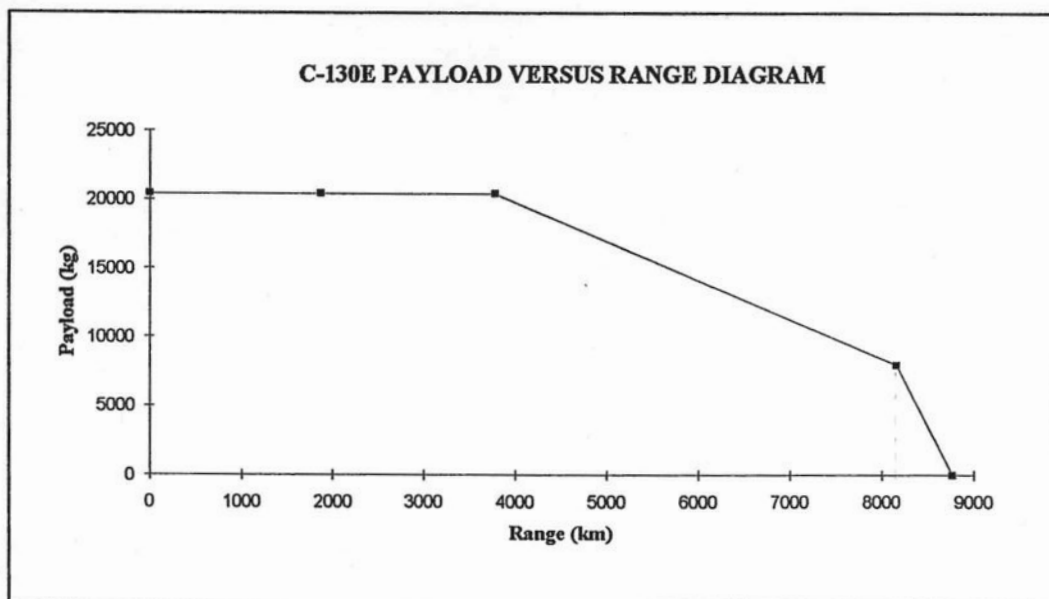


Figure 3.11

Figure 3.11 shows the range versus payload diagram for the C-130 Hercules which when compared with Figure 3.10 shows the F93B in a favourable light. However, Figure 3.11 does not include a reserve factor of 5% of the initial fuel plus 30 minutes flying time, which means that the ranges quoted for the C-130 are not the total ranges and the distances will be underestimated compared to the F93B figures which include the reserve factor of 1250km in the diagram.

4.1 The Sparv programme is an aerodynamic panel programme available at the College of Aeronautics which computes the pressure and load force distribution on a component in both viscous and non-viscous modes. The analysis required 3 stages, the creation of an input model, running SPARV and interpretation of the results.

SPARV Input Models

4.2 There is no dedicated pre or post processor for the SPARV, therefore students are normally required to write their own pre-processor. Fortunately for the F93 project we were able to use a pre-processor developed by Mr Djaffri, a PhD student. The x-pot data file for the wing component was created by manually inputting the x and y co-ordinates for the upper and lower surfaces of the supercritical airfoil, whilst the four-upt data file was created by manually reporting characteristics such as the aircraft height and length.

Running SPARV

4.3 Although the SPARV programme can now be run on the SUN system thus saving a considerable amount of processing time SPARV is not a particularly user friendly system and it is recommended that students take time to discuss operation of the SUN computer with Mr Oswald. Apart from obtaining a password for the SUN Mr Oswald will be able to increase file quotas to accommodate the large file space that SPARV requires. Once the component data files have been generated a pre-SPARV exe file is created which not only selects the input method for the wing data, (ie I - Interactive method or F - file input method) which was the file input method for the F93B project, but also requests information on what mode the programme should be run in and at what angles of attack. Once the pre-SPARV exe file is completed the SPARV can be run on the SUN computer. For a complete briefing on the programme capabilities the reader is recommended to the user manual or for specific programme commands to the F93 project Mr Chu's thesis.

SPARV Results

4.4 Wing Component The load distribution was considered for several angles of attack ranging from -40° to $+40^\circ$. Appendix E contains the post processed results for the wing component and Figure 4.1 shows the information in graph format. The load distribution generated by the SPARV programme followed the expected prediction given in Ref (1) fairly closely; however, the results pertained to the inviscid mode only which does not allow for any inaccuracies due to interference at the wing-fuselage joint. Despite this the results appeared to be fairly accurate.

CHAPTER 4

SPARV ANALYSIS

Introduction

4.1 The Sparv programme is an aerodynamic panel programme available at the College of Aeronautics which computes the pressure and load/force distribution on a component in both viscous and non-viscous modes. The analysis required 3 stages: the creation of an input model, running SPARV and interpretation of the results.

SPARV Input Models

4.2 There is no dedicated pre or post processor for the SPARV, therefore students are normally required to write their own pre-processor. Fortunately for the F93 project we were able to use a pre-processor developed by Mr Djafri, a PHD student. The input data file for the wing component was created by manually inputting the x and y coordinates for the upper and lower surface of the supercritical aerofoil, whilst the fuselage data file was created by manually inputting characteristics such as the aircraft height and length.

Running SPARV

4.3 Although the SPARV programme can now be run on the SUN system thus saving a considerable amount of processing time, SPARV is not a particularly user friendly system and it is recommended that students take time to discuss operation of the SUN computer with Mr Oswald. Apart from obtaining a password for the SUN Mr Oswald will be able to increase file quotas to accommodate the large file space that SPARV requires. Once the component data files have been generated a pre-SPARV.exe file is created which not only selects the input method for the wing data, (ie. I - Interactive method or F - file input method) which was the file input method for the F93B project, but also requests information on what mode the programme should be run in and at what angles of attack. Once the pre-SPARV.exe file is completed the SPARV can be run on the SUN computer. For a complete briefing on the programme capabilities the reader is recommended to the user manual or for specific programme commands for the F93 project Mr Chu's thesis.

SPARV Results

4.4 Wing Component The load distribution was considered for several angles of attack ranging from -4° to $+4^{\circ}$. Appendix E contains the post processed results for the wing component and Figure 4.1 shows the information in graph format. The load distribution generated by the SPARV programme followed the expected prediction given in Ref (1) fairly closely; however, the results pertained to the inviscid mode only which does not allow for any inaccuracies due to interference at the wing-fuselage joint. Despite this the results appeared to be fairly accurate.

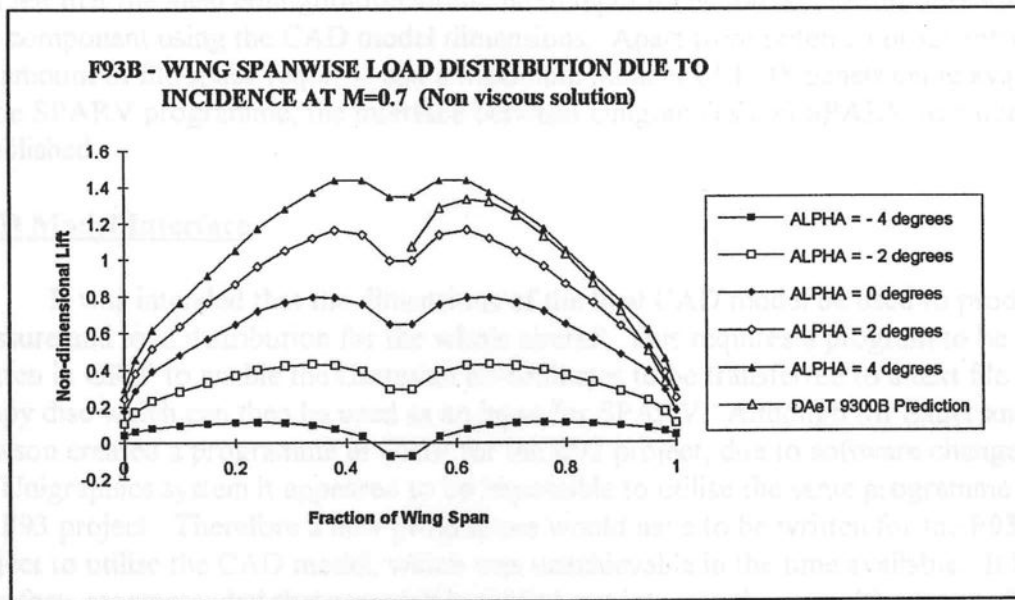


Figure 4.1

4.5 Fuselage Component The load distribution was considered for a simple cigar shaped fuselage. The detailed results are in Appendix E and Figure 4.2 shows the SPARV results compared with DAeT 9300B prediction for the aerodynamic loading in a graphical format.

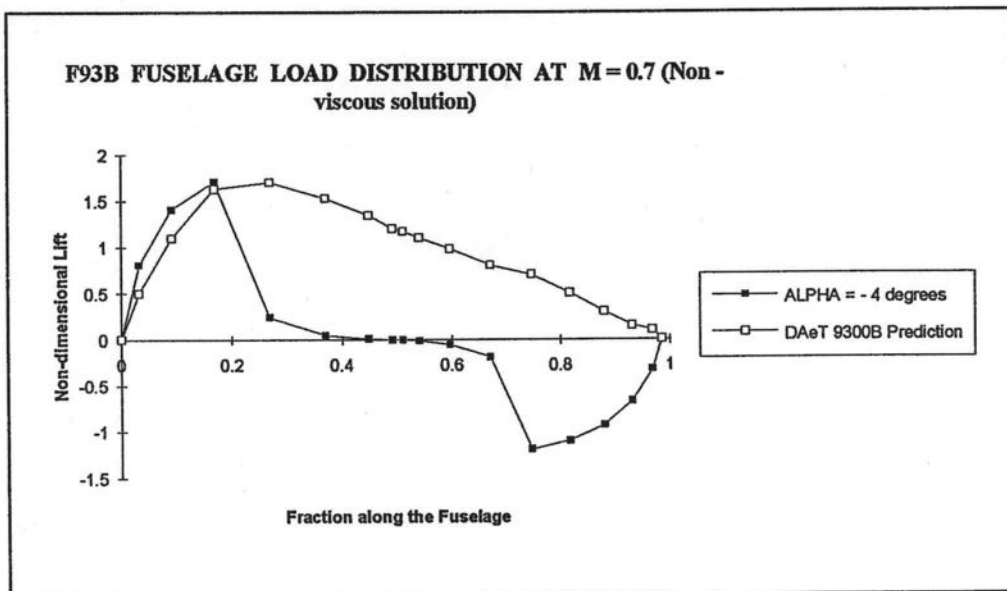


Figure 4.2

Because of the simplified shape of the fuselage input model and the fact that the programme was run in the inviscid mode there was no benefit gained from the shape of the rear fuselage ramp area. Therefore the SPARV results for the fuselage component

were not considered to be very accurate. A more accurate analysis was required and it was felt that the ideal configuration would be to input the whole aircraft into SPARV as one component using the CAD model dimensions. Apart from potential problems with the amount of file space required and a maximum number of 1248 panels being available in the SPARV programme, the interface between Unigraphics and SPARV was not fully established.

CAD Model Interface

4.6 It was intended that the dimensions of the final CAD model be used to produce a pressure and load distribution for the whole aircraft. This requires a program to be written in GRIP to enable the Cartesian co-ordinates to be transferred to a text file on a floppy disc which can then be used as an input for SPARV. Although Mr Sauer and Mr Hewson created a programme in GRIP for the E92 project, due to software changes in the Unigraphics system it appeared to be impossible to utilise the same programme for the F93 project. Therefore a new programme would have to be written for the F93 project to utilise the CAD model, which was unachievable in the time available. It is, therefore, recommended that research is carried out into whether a multi-purpose GRIP programme could be written for use on either the version 9.2 or 10 Unigraphics system to allow future students easy access to any CAD model data for use in the SPARV programme.

ADDITIONAL MARITIME PATROL EQUIPMENT

5.2 Due to the short undercarriage on the F93B all of the additional equipment required for the maritime role had to be accommodated externally or on underwing pylons. This severely restricts the number of anti-submarine weapons that can be carried by the F93BMPA as these weapons would normally be carried in a fuselage bay.

Internal Equipment

5.3 The additional maritime equipment, which is located internally includes: central operators positions, electronic surveillance equipment, sonobuoy equipment, crew rest areas, life rafts, radar, search light, low light television, forward looking infrared as well as the 6 additional crew members required to operate the equipment. An additional requirement which must be specified in the F93B basic transport aircraft design is: 2 observation windows which can be easily fixed and removed at the aircraft transition between roles. The justification for the equipment is also detailed in Appendix F and the role conversion drawing F93B P-01 is attached at the end of Appendix F. Due to the large amount of space available inside the F93B there was sufficient room for a 2,000 gallon internal fuel tank to be carried providing the APU and main engines were removed and the freight bay

CHAPTER 5

MARITIME ROLE CONVERSION

Introduction

5.1 In order to increase the potential market for the F93B Ref (1) required that consideration be given to role conversion of the F93B into a maritime surveillance aircraft. Aircraft used in ASW operations require a wide range of special equipment and weapons. The major problems are the initial detection of a submerged submarine from the air and the subsequent pin-pointing of its position with sufficient accuracy to permit effective attack. To help counter the modern submarine, various electronic, sonar and other equipment has been developed, but its performance depends greatly on weather and water conditions. Thus, the chances of successful detection are increased by the variety of equipment carried, each type of equipment providing some information which, when fitted into the whole picture, results in detection of the submarine. Nevertheless, visual search is still important; even nuclear submarines may from time to time expose aials to transmit or receive information or to gain attack intelligence. Moreover, the conventional type of submarine still has to snort for up to one hour in every 8 hours to charge its batteries. To operate efficiently, an aircraft may therefore maintain a visual search as well as exploiting its other detection devices. This chapter discusses the location of all the additional equipment required to perform a maritime role and the drag penalty incurred from carrying external stores. Finally the performance characteristics of the F93B Maritime Patrol Aircraft (MPA) are assessed and compared with the latest RAF specification.

ADDITIONAL MARITIME PATROL EQUIPMENT

5.2 Due to the short undercarriage on the F93B all of the additional equipment required for the maritime role had to be accommodated internally or on underwing pylons. This severely restricts the number of anti-submarine weapons that can be carried by the F93BMPA as these weapons would normally be carried in a bomb bay.

Internal Equipment

5.3 The additional maritime equipment which is located internally includes: tactical operators positions, electronic surveillance equipment, sonobuoy equipment, crew rest areas, life rafts, radar, search light, low light television, forward looking infra-red as well as the 6 additional crew members required to operate the equipment. An additional requirement which must be specified in the F93B basic transport aircraft design are 2 observation windows which can be easily fitted and removed when the aircraft converts between roles. The justification for the equipment location is detailed in Appendix F and the role conversion drawing F93B P-01 is situated at the end of Appendix F. Due to the large amount of space available inside the F93B there was sufficient room for an internal fuel tank to be carried providing the AUM limit was relaxed and the freight bay

floor was capable of withstanding the point loading. The dimensions and weight of the extra equipment are specified in Table 5.1 at para 5.5.

External Equipment

5.4 The carriage of external equipment on the aircraft is very important in determining the drag characteristics of the aircraft. For the F93BMPA external weapons in the form of anti-submarine torpedoes or air-to-surface missiles are required depending on the particular target. Active self defence measures include AIM-9 missiles whilst passive self defence measures include chaff and flare cartridges. Surveillance equipment is carried in 2 wing tip pods plus one fin pod as well as a Magnetic Anomaly Detector (MAD) in a rear boom. Appendix F describes the external equipment fitted and drawing F93B P-02 at the end of the Appendix shows the location of the external equipment. Table 5.1 details the variety of external stores which could be carried by the F93BMPA.

Equipment Dimensions

5.5 Table 5.1 details the dimensions of the additional equipment.

MARITIME PATROL EQUIPMENT DIMENSIONS				
EQUIPMENT (QTY)	LENGTH (m)	WIDTH (m)	HEIGHT (m)	WEIGHT (kg)
INTERNAL				
Operators Consoles	6.04	1.6	2.1	400
Electronic Equipment	2.9	2.6	-	1140
Sonobuoy Racks	3.3	1.6	1.5	2286
Sonobuoy Chute	1.3	0.8	1	50
Rest Area	3.4	1.8	-	-
Sleeping Bunks	6	1	2	-
Radar	-	1.14	0.4	237
Searchlight	0.7	0.5	-	10
LLTV	0.25	0.25	0.5	6
FLIR	0.5	0.5	0.25	10
SAR Rafts	1	0.5	0.8	90
Extra Internal Fuel	3	1.5 radius	-	14293
Additional Crew	-	-	-	540
Observation Windows	0.6	0.3 dia	-	-
EXTERNAL				
MAD	0.84	0.18 dia	-	16
ESM Pods (3)	2	0.46 dia	-	60
AIM-9 missiles (2)	2.87	0.127 dia	-	174
Exocet missiles (2)	4.7	0.35 dia	-	670
Harpoon missiles (2)	3.9	0.343 dia	-	1060
Stingray Torpedoes (2)	2.6	0.324 dia	-	530

Table 5.1

DRAG ESTIMATION

5.5 The increase in drag from each individual external store was calculated manually using Ref (4), (8) and (9), in Appendix G, which resulted in a total increase in the zero lift coefficient of drag (C_{D0}) of 0.00123 whilst the coefficient of lift (C_L) is assumed to be unaffected by the addition of external stores. Therefore the new drag polar for the F93BMPA at an altitude of 10km and Mach 0.7 is:

$$C_D = 0.0222 + 0.39C_L^2$$

However, there was insufficient information available in Ref (4), (8) and (9) to establish the change in the zero lift coefficient of drag when the aircraft is being operated at its patrol height of 1000ft (305m) and at slow speeds. Therefore, in order to establish a drag polar for the patrol conditions the College of Aeronautics DELTADD programme was utilised.

5.6 DELTADD Programme The input data required for the DELTADD programme is detailed in Appendix G Table G-1 and the case data requested is shown in Table 5.2.

CASE DATA			
CASE NO	MACH NO	ALTITUDE (m)	AIRCRAFT MASS (kg)
1	0.7	10000	100085.8
2	0.3	305	93043.3
3	0.35	305	93043.3
4	0.4	305	93043.3

Table 5.2

A reference case at an altitude of 10km and Mach 0.7 was specified in order to check the manual calculations and 3 cases at various patrol speeds were specified so that the most suitable patrol speed could be established. Table 5.3 shows the DELTADD results. The DELTADD results for the reference case were surprisingly close to the manual results, a difference of 0.0001, in that DELTADD estimated a drag polar for the F93BMPA at an altitude of 10km and Mach 0.7 of :

$$C_D = 0.0223 + 0.039C_L^2$$

DELTADD DRAG ESTIMATE RESULTS				
CASE DATA	DRAG POLAR	CASE C_L	CASE C_D	BUFFET ONSET C_L
1	$C_D = 0.0223 + 0.039C_L^2$	0.5592	0.0329	0.857
2	$C_D = 0.0233 + 0.039C_L^2$	0.7658	0.0451	1.415
3	$C_D = 0.0229 + 0.039C_L^2$	0.5626	0.035	1.345
4	$C_D = 0.0225 + 0.039C_L^2$	0.4307	0.0285	1.287

Table 5.3

At Mach 0.3 (200 knots) the coefficient of lift = 0.77 which was considered to be too high to maintain comfortably especially when the surveillance equipment requires a steady platform. At Mach 0.35 (230 knots) the coefficient of lift = 0.56 which is a more realistic value to maintain; whilst at Mach 0.4 (264 knots) the coefficient of lift = 0.43 but it is doubtful whether the RB509-18 engine can achieve this speed at low level. Therefore, a patrol speed of Mach 0.35 (230 knots) was considered to be a satisfactory balance between a low patrol speed and a low C_L value. Therefore, the following drag polars were used to assess the performance of the F93BMPA; although the increase in drag caused by the external stores carried by the F93BMPA is actually very small. This is thought to be because of the large wing area of the F93B unlike a smaller fighter aircraft with a small wing area which would be affected to a much greater extent by carrying external stores.

Drag Polar during Cruise at 10km Altitude and Mach 0.7 = $C_D = 0.0223 + 0.039C_L^2$

Drag Polar during Patrol at 305m and Mach 0.35 = $C_D = 0.0229 + 0.039C_L^2$

F93BMPA PERFORMANCE

5.7 The F93BMPA performance was assessed using the same equations generated for the F93B transport variant in Appendices A, B and C except for the following differences: revised drag polars developed in Appendix G were utilised, the 3g manoeuvre requirement placed on the F93B was relaxed to allow for an increased maximum AUM due to the carriage of additional fuel in an internal tank and the possibility of shutting down 2 engines during the patrol phase was considered in order to increase patrol endurance.

Take-off Performance

5.8 The maximum AUM of 113162kg occurs when the F93BMPA is carrying Exocet missiles and the maximum internal fuel load of 14293kg. The ground run required is 1084m and the distance to rotate is 198m. The distance from lift-off to a screen height of 10.7m (35ft) is 350m giving a total of 1632m.

Maximum Take-off Distance required for the F93BMPA = 1632m

This take-off distance is outside the original design limitation of 1500m and may be a factor that future buyers would wish to consider if both variants are intended to operate from the same base. Figure 5.1 shows the variation in take-off distance required at different AUM.

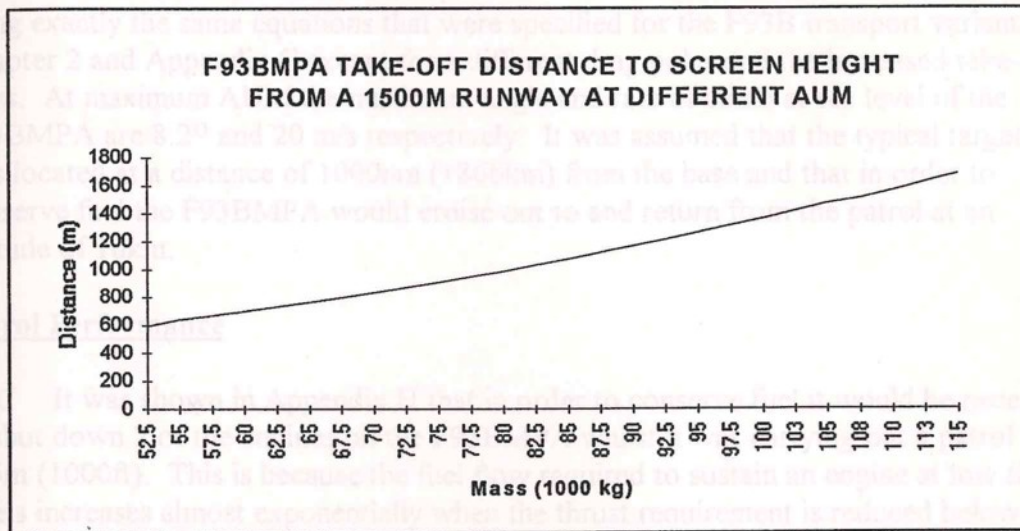


Figure 5-1

In order to take-off within 1500m the F93BMPA must reduce its maximum AUM to 107200kg which gives a maximum internal fuel load of 9141kg which is still almost a 25% increase on the basic fuel load. Figure 5.2 shows the 3 engine capability of the F93BMPA from a normal runway. The additional requirement to take-off from a semi-prepared strip has not been considered for the F93BMPA as it unlikely to occur.

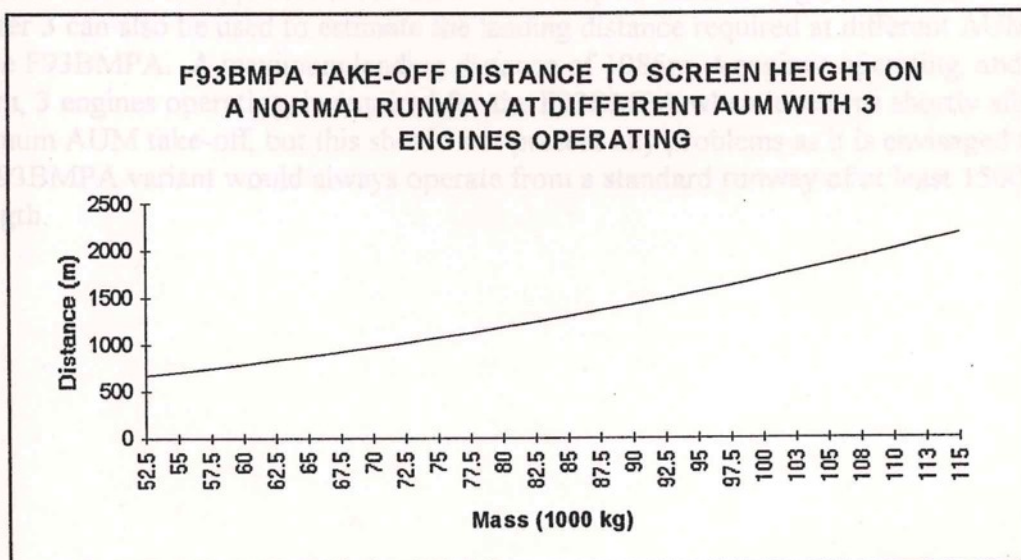


Figure 5-2

Climb, Cruise and Descent Performance

5.9 The climb, cruise and descent performance for the F93BMPA was calculated using exactly the same equations that were specified for the F93B transport variant in Chapter 2 and Appendix C except for a different drag polar and the increased take-off mass. At maximum AUM the maximum angle and rate of climb at sea level of the F93BMPA are 8.2° and 20 m/s respectively. It was assumed that the typical target area was located at a distance of 1000nm (1866km) from the base and that in order to conserve fuel the F93BMPA would cruise out to and return from the patrol at an altitude of 10km.

Patrol Performance

5.10 It was shown in Appendix H that in order to conserve fuel it would be necessary to shut down 2 of the engines on the F93BMPA whilst it was carrying out a patrol at 305m (1000ft). This is because the fuel flow required to sustain an engine at low thrust levels increases almost exponentially when the thrust requirement is reduced below 1/3 of the maximum available thrust. Therefore in order to maintain the individual engine thrust requirements above 1/3 of their maximum 2 of the engines must be shut down. With only 2 engines operating the fuel flow at 305m was assumed to be 3/4 of the maximum fuel flow available at that altitude.

Landing Performance

5.11 The landing performance for the F93BMPA can be calculated using the same equations derived in Appendix B for the F93B transport variant. Figures 3.7 and 3.8 in Chapter 3 can also be used to estimate the landing distance required at different AUM for the F93BMPA. A maximum landing distance of 1085m, 4 engines operating, and 1152m, 3 engines operating, is required for the F93BMPA when it returns shortly after a maximum AUM take-off, but this should not present any problems as it is envisaged that the F93BMPA variant would always operate from a standard runway of at least 1500m in length.

Mission Profiles

5.12 Appendix H contains the detailed breakdown of the equipment used on different mission profiles whilst Table 5.4 shows the range and endurance figures for several different aircraft configurations.

MISSION PROFILES					
CONFIGURATION	1	2	3	4	5
Total MPA Equipment	6219kg*	2509kg**	5689kg***	6499kg****	5689kg***
Total Fuel Load	40000kg	52321kg	49141kg	48331kg	54293kg
TOTAL TAKE-OFF AUM	98589kg	107200kg	107200kg	107200kg	112352kg
Patrol Duration	6.2hrs	9.2hrs	8.4hrs	8.2hrs	9.7hrs
TOTAL DURATION	11.1hrs	14.2hrs	13.4hrs	13.2hrs	14.6hrs
Patrol Range	2620km	3915km	3575km	3485km	4100km
TOTAL RANGE	6233km	7546km	7215km	7127km	7758km

* 2 x Harpoon missiles + 2 x AIM-9

** Nil weapons or Sonobuoys

*** 2 x Stingray missiles + 2 x AIM-9

**** 2 x Exocet missiles + 2 x AIM-9

Table 5.4

Comparison of F93BMPA with Future RAF Specification

5.13 Ref (15) states that the RAF intends to extend the specification for its next generation of maritime patrol aircraft up to an unrefuelled mission endurance of 14 hours from the Nimrod's current 8 hour endurance in the anti-submarine role. Whilst the F93BMPA is capable of achieving the endurance specification it lets itself down, like smaller MPA aircraft such as the Fokker 50 and the CN-235, with its lack of ability to carry large quantities of torpedoes or anti-ship missiles to counter potential threats. To improve the offensive capabilities of the F93BMPA further research is required on methods of transporting greater quantities of weapons. In its present form the F93BMPA would be an excellent maritime surveillance aircraft capable of staying on station for long time periods but it may require assistance with firepower once a target had been detected.

CHAPTER 6

COST PREDICTION

Introduction

6.1 In order to sell the F93B the unit cost of the aircraft must be competitive compared with other aircraft in the same field such as the C-130J which costs approximately US\$ 45 million. Initial estimates based on comparisons of existing aircraft on the market revealed that a realistic price for the F93B based on its maximum AUM would be between US\$ 50 and 52 million. The unit cost depends to a large extent on the number of aircraft manufactured therefore the variation in unit cost with production numbers was established by analysing the research, development, test, evaluation and acquisition costs (Appendix I). Another important cost required for budgeting purposes is the life cycle cost which includes not only the unit cost of the aircraft but also the direct operating costs and disposal costs (Appendix J). The increase in unit cost and life cycle costs for the F93B maritime patrol aircraft are also considered.

UNIT COST

Research, Development, Test and Evaluation Cost (Crdte)

6.2 The total cost for research, development, test and evaluation is US\$ 920.8 million which is broken down as follows:

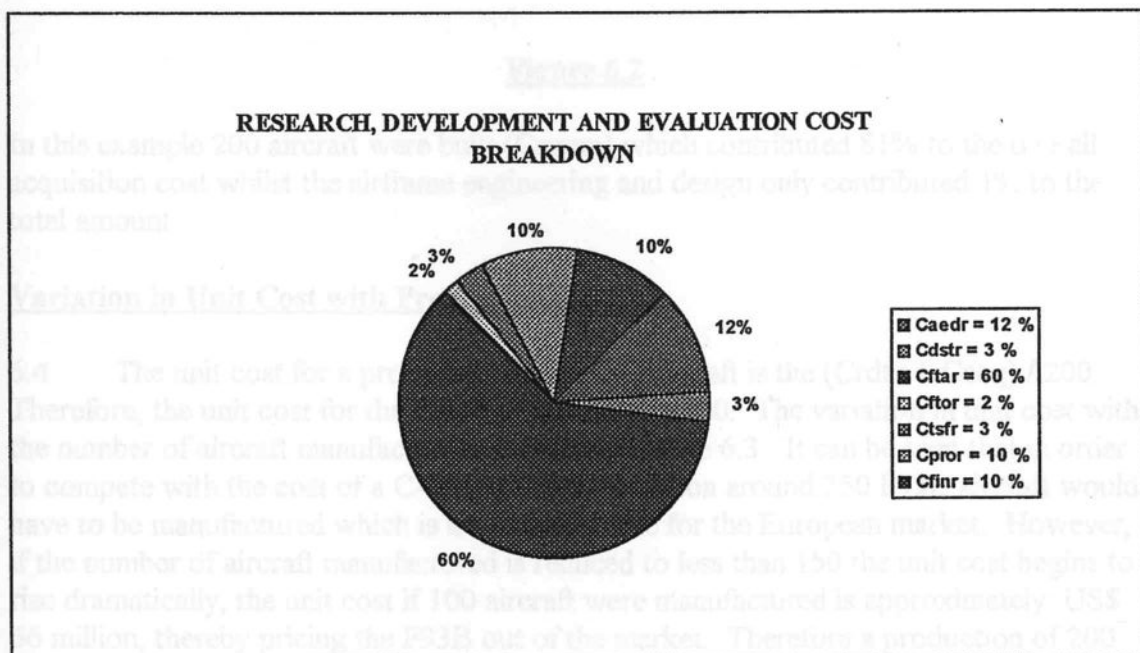


Figure 6.1

The dominant contributor to Crdte was the number of prototype flight test aircraft built (Cftar). In this case 4 aircraft were built, one for static testing and 3 for flight testing. At 60 % of the total cost careful consideration should be given to the number of prototype aircraft that are required.

Acquisition Costs (Cacq)

6.3 The total acquisition cost is US\$ 9304.8 million which is broken down as follows:

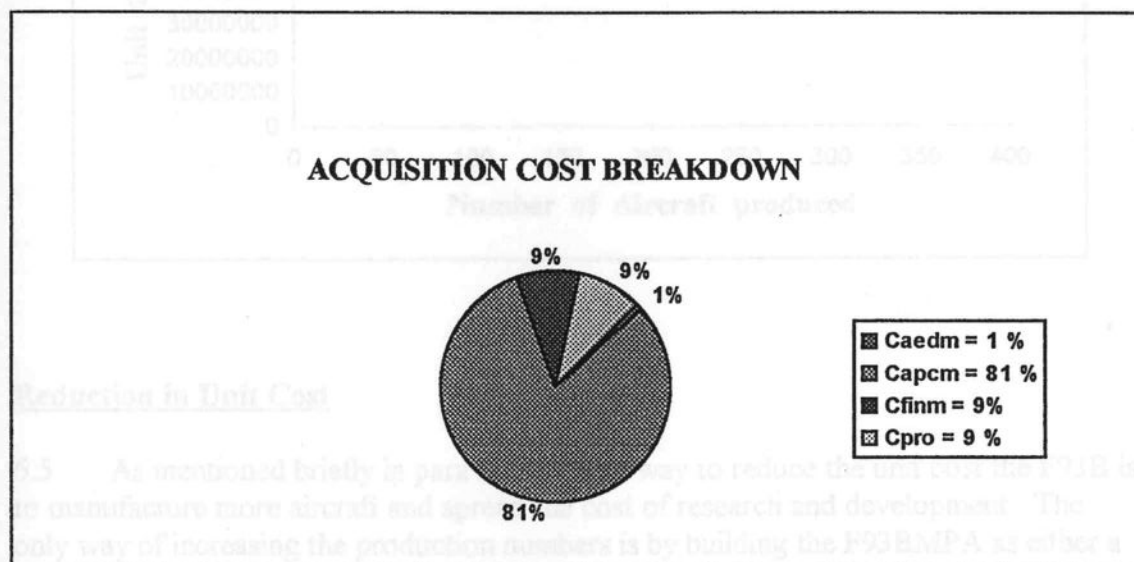


Figure 6.2

In this example 200 aircraft were built (Capcm) which contributed 81% to the overall acquisition cost whilst the airframe engineering and design only contributed 1% to the total amount.

Variation in Unit Cost with Production Run

6.4 The unit cost for a production run of 200 aircraft is the $(Crdte + Cacq) / 200$. Therefore, the unit cost for the F93B is US\$ 51 128 220. The variation in unit cost with the number of aircraft manufactured is shown in Figure 6.3. It can be seen that in order to compete with the cost of a C-130J at US\$ 45 million around 350 F93B aircraft would have to be manufactured which is a little optimistic for the European market. However, if the number of aircraft manufactured is reduced to less than 150 the unit cost begins to rise dramatically, the unit cost if 100 aircraft were manufactured is approximately US\$ 66 million, thereby pricing the F93B out of the market. Therefore a production of 200 aircraft is a suitable number which allows the F93B to be competitively priced at just over US\$ 51 million.

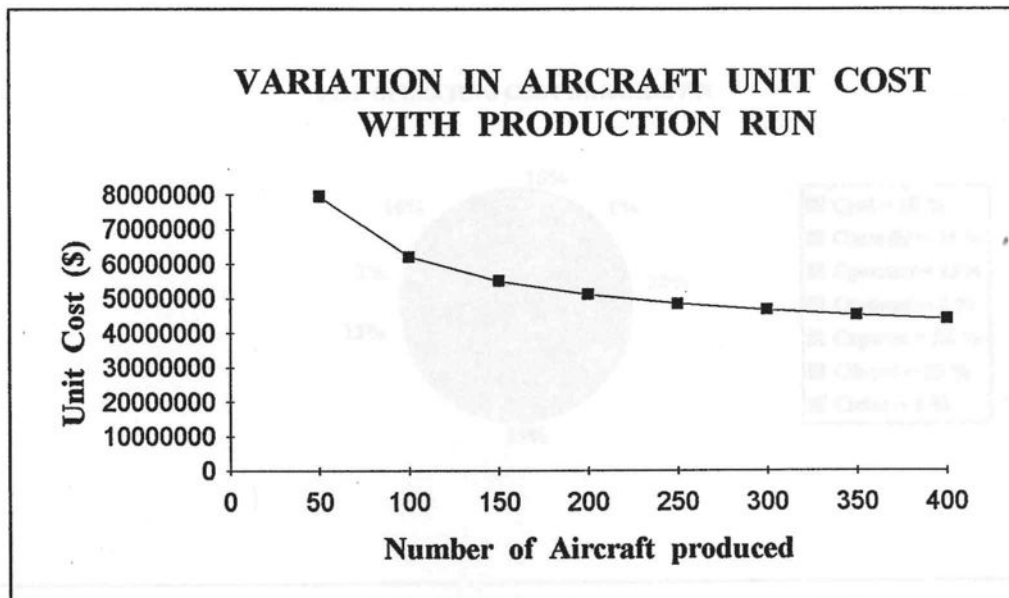


Figure 6.3

Reduction in Unit Cost

6.5 As mentioned briefly in para 6.4 the only way to reduce the unit cost the F93B is to manufacture more aircraft and spread the cost of research and development. The only way of increasing the production numbers is by building the F93BMPA as either a European or even better a Far Eastern joint collaborative venture. This would increase the potential market (not including USA or Russia) to around 350 aircraft, Ref (4), which is currently being served by a combination of PC3, Altantique, Nimrod, Bear and ex-US Navy S2 aircraft.

LIFE CYCLE COST (Clcc)

F93B Operating Costs (Cops)

6.6 The operating costs for the F93B were calculated on a fleet size of 60 aircraft with an average annual utilisation of 800hrs and 30 years in service. The total operating cost for the F93B is US\$ 3 997 876 776 which is broken down in Figure 6.4. At 33% the largest contributor to the operating costs is the cost of direct personnel, the cost of crew and maintenance personnel, which remains constant unless the aircraft are grounded. However the 20% contribution to the operating costs made by the cost of fuel, oil and lubricants (Cpol) will vary according to the mission lengths and the price of fuel. Therefore there is likely to be some variation to the total operating costs due to Cpol. For a fleet of 60 aircraft the operating cost per flight hour is US\$ 3381.

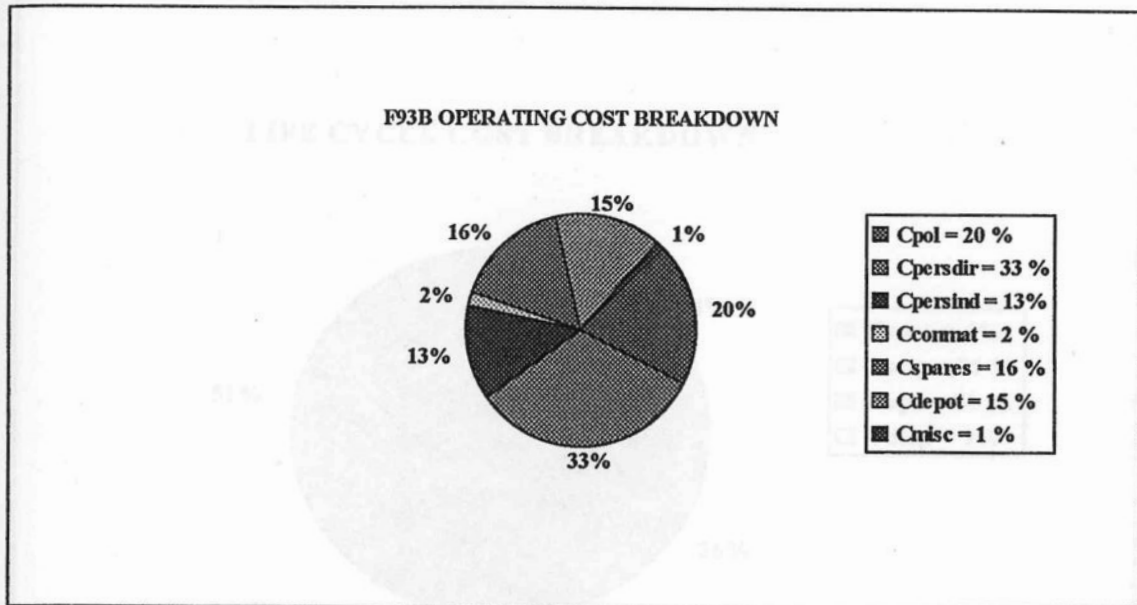


Figure 6.4

Disposal Costs (Cdisp)

6.7 The disposal cost for both variants of the F93B is estimated to be 0.01 Clcc from Ref (12) and therefore is US\$ 71 million.

F93B Life Cycle Cost (Clcc)

6.8 The total life cycle cost is $Crdte + Cacq + Cops + Cdisp$. Therefore assuming the RAF purchase 60 airframes to replace their transport fleet then the total $Crdte$ and $Cacq$ cost is the unit cost $\times$ the number of aircraft bought. The total life cycle cost = $(US\$ 51.1 \text{ million} \times 60 + 4000 \text{ million}) / (1 - 0.01) = US\7137 million and the unit life cycle cost is US\$119 million. The breakdown of the F93B life cycle cost is shown in Figure 6.5. It can be seen that the main contributors to the life cycle cost are the operating cost and the manufacturing cost with the research and development costs accounting for less than the disposal cost.

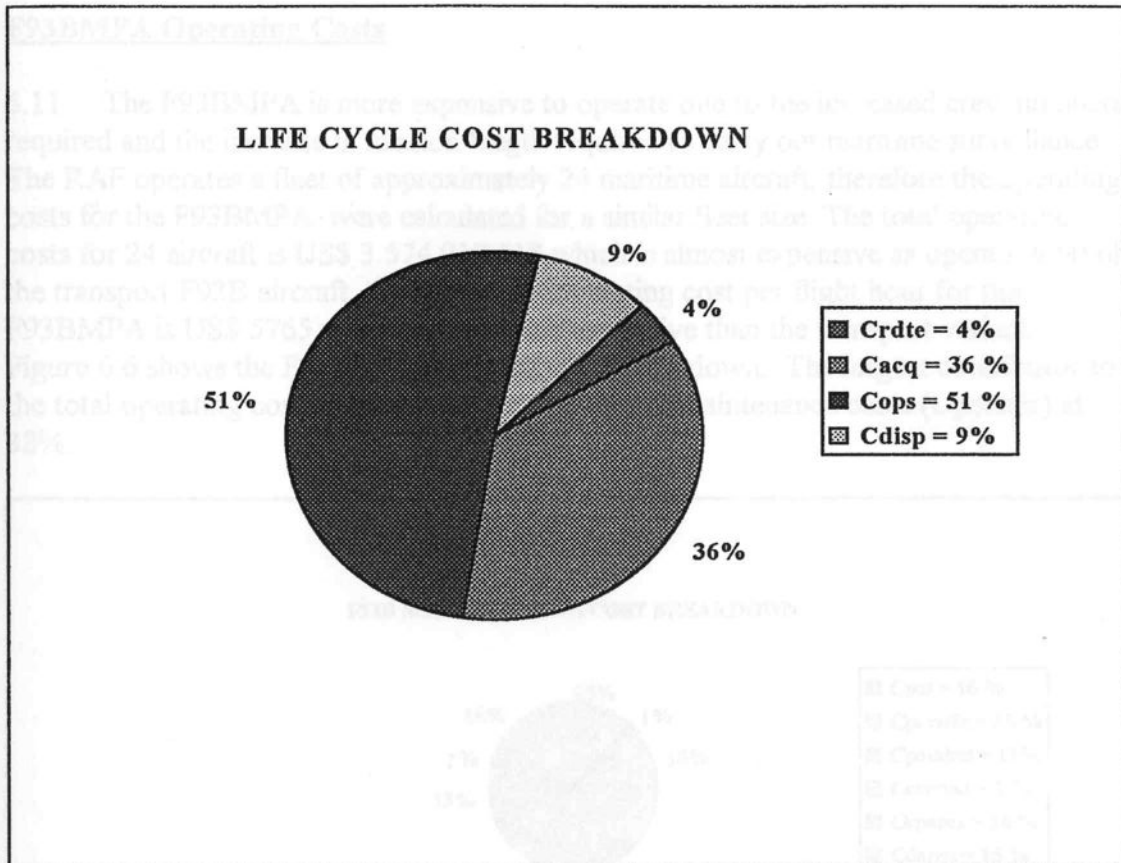


Figure 6.5

F93BMPA COSTS

6.9 Although the basic airframe for the F93BMPA costs the same as the transport variant. There are several additional costs associated with maritime surveillance aircraft.

Unit Cost of F93BMPA

6.10 Ref (4) stated that a complete suite of updated mission equipment capable of carrying out maritime surveillance including anti-submarine warfare would cost in the region of US\$ 11.8 million. Therefore the final unit cost for the F93BMPA is approximately US\$ 63 million. A recent sale of 8 PC3 aircraft to South Korea for a unit cost of US\$ 75 million, quoted in Ref (14), implies that the F93B MPA unit cost is very competitive. However the F93BMPA is designed as a role conversion from the military transport version of the F93B and therefore customers would not expect to pay as much for the role conversion as they would for a purpose designed maritime surveillance aircraft. This is primarily because the F93BMPA can only carry minimal weapons for self defence and anti-submarine warfare. Despite its one shortcoming the F93BMPA's dual capability should be an attractive selling point.

F93BMPA Operating Costs

6.11 The F93BMPA is more expensive to operate due to the increased crew numbers required and the increase in mission length required to carry out maritime surveillance. The RAF operates a fleet of approximately 24 maritime aircraft, therefore the operating costs for the F93BMPA were calculated for a similar fleet size. The total operating costs for 24 aircraft is US\$ 3 574 917 518 which is almost expensive as operating 60 of the transport F93B aircraft. Therefore the operating cost per flight hour for the F93BMPA is US\$ 5765 which is 70% more expensive than the transport variant. Figure 6.6 shows the F93BMPA operating cost breakdown. The largest contributor to the total operating cost is the yet again the crew and maintenance costs (Cpersdir) at 38%.

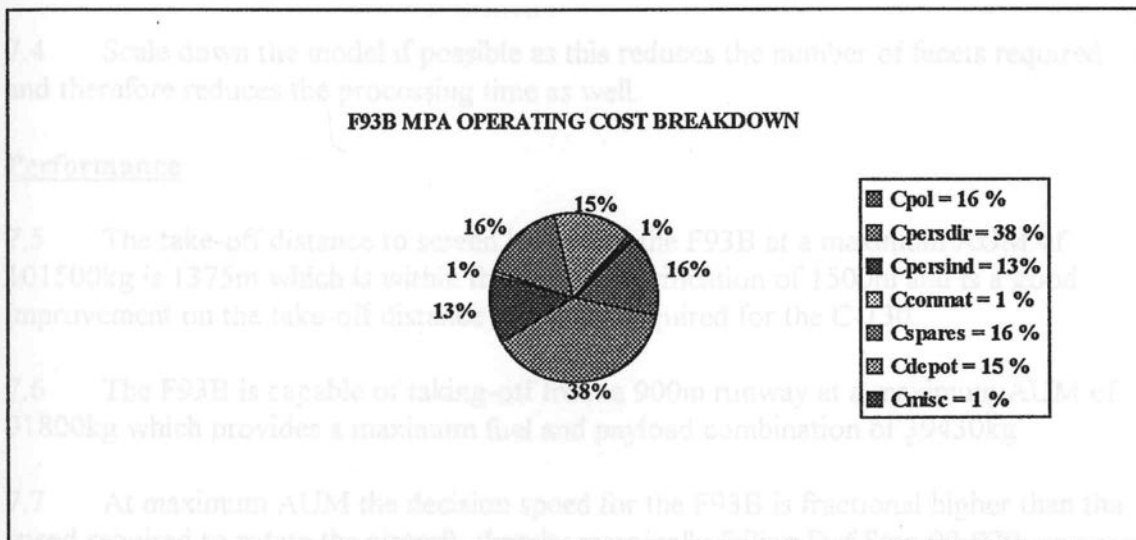


Figure 6.6

F93B MPA Life Cycle Cost

6.12 The disposal cost for 24 maritime variants is US\$ 51.3 million and the total life cycle cost is US\$ 5136.5 million giving a unit life cycle cost of US\$ 214 million.

CHAPTER 7

CONCLUSIONS

Cad Model

- 7.1 STL file size should be kept to a maximum of 3 Mb.
- 7.2 When creating STL files it is better not to use appended files.
- 7.3 If the model is too large separate it into smaller components and ensure that each sub part is fully enclosed.
- 7.4 Scale down the model if possible as this reduces the number of facets required and therefore reduces the processing time as well.

Performance

- 7.5 The take-off distance to screen height for the F93B at a maximum AUM of 101500kg is 1375m which is within the design specification of 1500m and is a good improvement on the take-off distance of 1573m required for the C-130.
- 7.6 The F93B is capable of taking-off from a 900m runway at a maximum AUM of 91800kg which provides a maximum fuel and payload combination of 39430kg.
- 7.7 At maximum AUM the decision speed for the F93B is fractional higher than the speed required to rotate the aircraft, thereby marginally failing Def Stan 00-970. Therefore when using 50% reverse thrust the F93B does not have a balanced field length as it will always be quicker to stop the aircraft if it suffers an engine failure during the ground run.
- 7.8 The F93B maximum AUM for a 3 engine take-off from a 1500m runway is 93030kg giving a maximum fuel and payload combination of 40660kg.
- 7.9 The F93B maximum AUM for a 3 engine take-off from a semi-prepared strip 900m long is 78500kg giving a fuel and payload combination of 26130kg.
- 7.10 The landing distance required for the F93B at an AUM of 78962kg (20000kg over 5500km range) is 637m giving a maximum deceleration of 0.52g.
- 7.11 The payload/range requirement was exceeded by 475km (8.6%).
- 7.12 The maximum range of the F93B is 10186km whilst the maximum range of the C-130 is 8750km, a 16 % improvement.

7.13 The maximum rate of climb for the F93B at sea level is 23 m/s which is considerably faster than the C-130 which has a maximum rate of climb of 9.65 m/s at sea level.

7.14 Time to height for the F93B at maximum AUM is 12.5 minutes.

7.15 The service ceiling for the F93B is 40000ft compared to 33000ft for the C-130.

Maritime Patrol Role Conversion

7.16 The maximum additional fuel which can be carried in an internal fuel tank is 14293kg.

7.17 The zero lift component in the F93B drag polar is increased by 0.00123 due to the carriage of external stores at an altitude of 10km.

7.18 The maximum take-off distance to screen height required for the F93BMPA at an AUM of 113162kg is 1632m.

7.19 The maximum AUM of the F93BMPA which can take-off from a 1500m runway is 107200kg.

7.20 The maximum patrol duration achieved by the F93BMPA is 9.7 hrs and the patrol range is 4100km. The total sortie duration is 14.6 hrs and covers 7758km which satisfies the specification of 14 hrs mission endurance required for the next generation of MPA aircraft required by the RAF.

Cost Estimation

7.21 The unit cost of the F93B aircraft is US\$ 51 128 220 providing 200 aircraft are produced.

7.22 The unit cost of the F93BMPA is US\$ 63 million.

7.23 Operating costs per flight hour for the F93B are US\$ 3381.

7.24 Unit life cycle cost for the F93B is US\$ 118 948 989.

7.25 Operating costs per flight hour for the F93BMPA are US\$ 5765.

7.26 Unit life cycle cost for the F93BMPA is US\$ 214 023 350.

CHAPTER 8

RECOMMENDATIONS

Cad Model/SPARV Interface

8.1 A general use GRIP programme for interfacing between the Unigraphics CAD system and the SPARV aerodynamic programme is required.

8.2 Unigraphics version 10 is much more user friendly and should be installed as soon as possible.

Performance

8.3 If possible, future engine data should be more comprehensive to avoid making general assumptions concerning fuel flow and SFC and various altitudes.

Maritime Patrol Role Conversion

8.4 Flight trials are recommended to establish the radar and FLIR coverage achieved on the F93BMPA.

8.5 Detailed loading calculations are required on the freight bay floor to establish whether the maximum amount of fuel (14293kg) can be carried in the internal fuel tank.

8.6 Further research into methods of increasing the weapons load that can be carried by the F93BMPA is required.

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TAKE-OFF PERFORMANCE

The take-off distance is characterized in 3 separate segments: the run out run distance from brake release to the velocity at rotation (V_R), when the aircraft leaves the runway, the distance travelled during V_R to V_{Lift} , when the main gear is rotated finally leaves the runway, and the airborne distance from V_{Lift} to a safety height 10.67m (35ft). In order to calculate the take-off distance for any aircraft, cross wing, simple sub-coefficients, thrust and lift slope equations have been derived from the engines manufacturers information and Ref (1).

Thrust Equations

Fig A-1 shows the limited data available from Rolls Royce for the RTVDA-13 engine.

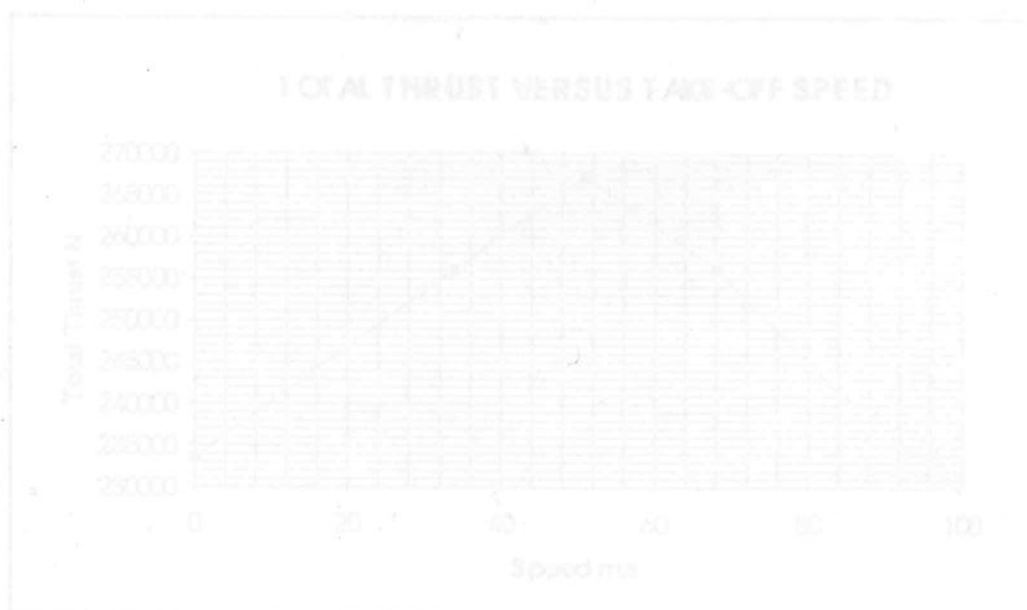


Figure A.1

APPENDIX A

TAKE-OFF AND BALANCED FIELD LENGTH PERFORMANCE

Introduction

1. The take-off performance for the F93B is focused around the take-off distance to screen height, which is the total distance from release of brakes through lift-off to a height of 10.67m (35ft) above the ground. The design specification at Ref (1) stated that the F93B aircraft should be able to take-off at full weight from a 1500m runway. The balanced field length performance calculates the maximum distance required to either take-off safely or stop following an engine failure during the take-off run.

TAKE-OFF PERFORMANCE

2. The take-off distance is calculated in 3 separate segments; the ground run distance from brake release to the velocity at rotation (V_R), when the nose wheel leaves the runway; the distance travelled during V_R to V_{LOF} , when the main undercarriage finally leaves the runway, and the airborne distance from V_{LOF} to a screen height of 10.67m (35ft). In order to calculate the take-off distance for any aircraft mass, using simple sub-routines, thrust and lift curve slope equations have been derived from the engines manufactures information and Ref (1).

Thrust Equations

3. Fig A-1 shows the limited data available from Rolls Royce for the RB509-18 engine.

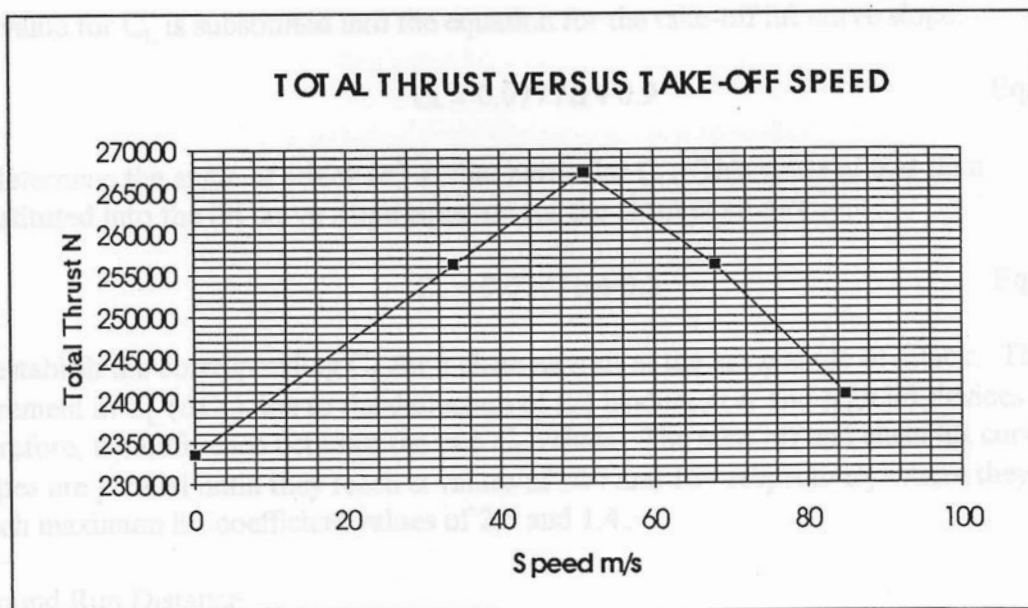


Figure A-1

The graph is unusual in that it increases in thrust from zero velocity to 51m/s and then decreases in thrust from 51m/s onwards. This is thought to be a result of propeller blade stall at the lower speeds which prevents the engines providing their full thrust capability. Without blade stall the graph would be extrapolated upwards and backwards from 51m/s producing significantly higher thrust values at lower speeds. Due to the nature of the graph two equations are required; from zero to 51m/s

$$T = 660.67V + 233563 \quad \text{Eqn (1)}$$

and 51m/s upwards

$$T = 267500 - 743.05(V - 51) \quad \text{Eqn (2)}$$

Therefore thrust data can be calculated for any instantaneous velocity during take-off.

Wing Lift Curve Slope Equations

4. Ref (1) details the wing lift curve slopes for the landing, take-off and clean configurations of the F93B aircraft. These slopes are used primarily to establish ΔC_L , the increment in C_L due to the deflection of landing gear and high lift devices, which is required to determine C_D via the drag polars given in Ref (1).

C_L is calculated at a particular weight and speed using:

$$C_L = \frac{W}{\frac{1}{2} \rho V_s^2 S} \quad \text{Eqn (3)}$$

the value for C_L is substituted into the equation for the take-off lift curve slope:

$$C_L = 0.0777\alpha + 0.9 \quad \text{Eqn (4)}$$

to determine the angle of attack (α) at that particular C_L . This value of α is then substituted into the lift curve slope equation for the clean configuration:

$$C_L = 0.0777\alpha + 0.1 \quad \text{Eqn (5)}$$

to establish the corresponding C_L for a clean aircraft at the same angle of attack. The increment in C_L (ΔC_L) due to the deflection of the landing gear and high lift devices is, therefore, the difference between the two C_L values. The take-off and clean lift curve slopes are parallel until they reach α values of 20° and 16° respectively where they reach maximum lift coefficient values of 2.5 and 1.4.

Ground Run Distance

5. Using Newtons equation of motion:

$$v = u + at$$

Eqn (6)

where $u = 0$ at start of ground roll. Therefore:

$$v = at$$

Eqn (7)

or

$$t = \frac{v}{a}$$

Eqn (8)

A first approximation for the ground run during take-off can be found by calculating the area under the graph of velocity / acceleration against velocity from Ref (?). The area under the graph $= 1/2(v/a)v$. The acceleration is calculated using $F=ma$ where m is the take-off mass of the aircraft and F is the instantaneous engine thrust at the mid position of each velocity interval. In order to calculate the necessary velocity intervals and corresponding engine thrusts the rotation speed V_R is required. This is a function of the stalling speed V_S . From Def Stan 00-970 the take-off safety speed (V_2) must be a minimum of $1.15V_S$ for military aircraft. Therefore, extrapolating backwards it is assumed that V_{LOF} is approximately $1.1V_S$ and V_R is $1.05V_S$ for the first iteration.

V_S occurs when the take-off lift coefficient (C_L) is at its maximum value of 2.5. At AUM:

$$M = 101\,500 \text{ kg}$$

$$\rho = 1.225$$

$$S = 193.3$$

Then rearranging Eqn (3):

$$V_S = \sqrt{\frac{101500 \times 9.81}{\frac{1}{2} \times 1.225 \times 193.3 \times 2.5}}$$

$$V_S = 58 \text{ m/s and } V_R = 1.05 \times 58 = \underline{60.9 \text{ m/s}}$$

Using 10 intervals of $V_R = 6.09 \text{ m/s}$ each and taking the mid position of each interval the thrust settings are shown in Table A-1.

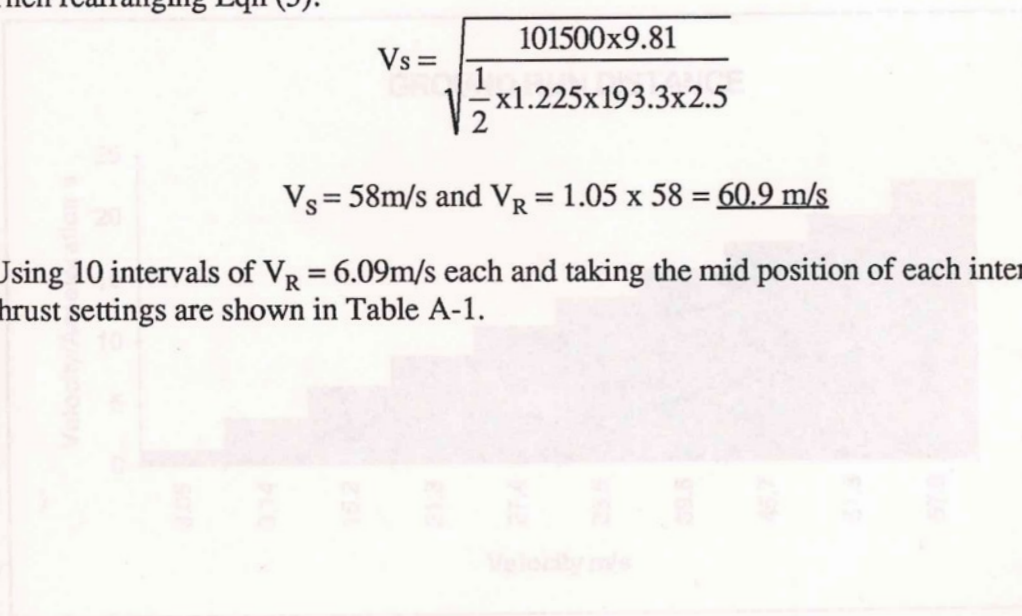


Figure A-3

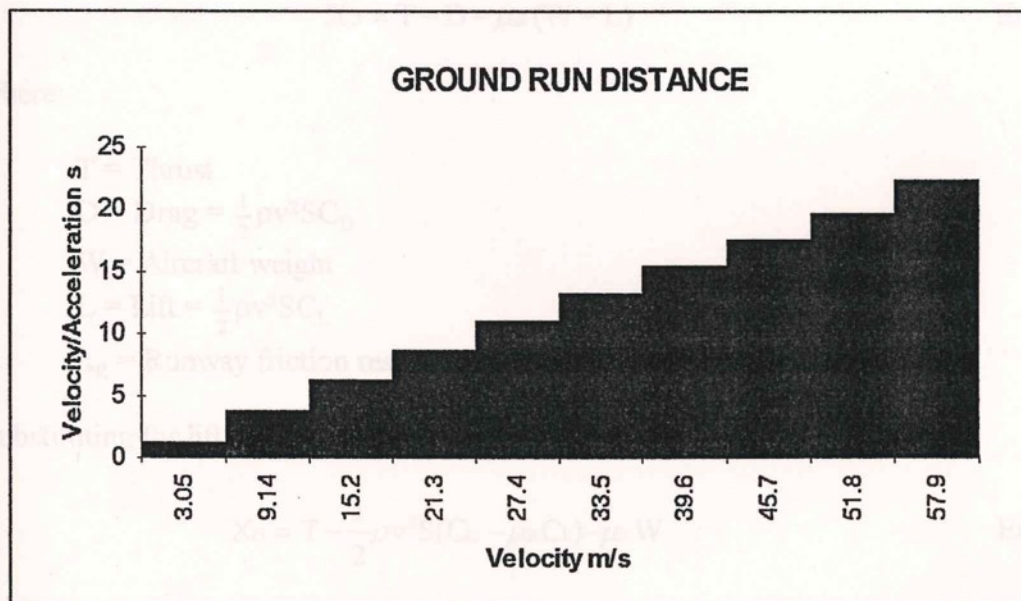
VELOCITY AT MID INTERVAL POSITIONS	THRUST AT MID INTERVAL POSITIONS
3.045 m/s	235574.7 N
9.135 m/s	239598.2 N
15.225 m/s	243621.7 N
21.315 m/s	247645.2 N
27.405 m/s	251668.7 N
33.495 m/s	255692.2 N
39.585 m/s	259715.7 N
45.675 m/s	263739.1 N
51.765 m/s	266931.5 N
57.855 m/s	262406.3 N

Table A-1

Fig A-2 shows v/a against v over 10 intervals between $v = 0$ and $v = 61$ m/s. The area under the graph which equates to the ground run distance to V_R is

$$\frac{1}{2} \times 23.6 \times 61 = 720 \text{ m}$$

However this result neglects all drag and runway resistance which will tend to increase the length of runway required for take-off. Therefore a more precise calculation is required.

**Figure A-2**

Using Newtons equation for distance:

$$S_{GR} = ut + \frac{1}{2}at^2 \quad \text{Eqn (9)}$$

Substitute for t^2 from Eqn (8) and $ut = 0$ at the start of the take-off run.

$$S_{GR} = \frac{1}{2}a \frac{v^2}{a^2} = \frac{1}{2} \frac{v^2}{a} \quad \text{Eqn (10)}$$

Substituting for acceleration in Eqn (10) using $a = F/m$

$$S_{GR} = \frac{1}{2}m \frac{v^2}{F} \quad \text{Eqn (11)}$$

where $F = X_G =$ accelerating force.

Therefore over several increments of velocity from 0 to V_R the ground distance is:

$$S_{GR} = \frac{1}{2}m(\Delta v)^2 \sum \frac{(2n-1)}{X_G} \quad \text{Eqn (12)}$$

where Δv is the velocity interval and $\frac{1}{2}(2n-1)\Delta v$ is the mean velocity during each interval.

The accelerating force X_G depends on several factors:

$$X_G = T - D - \mu_R(W - L) \quad \text{Eqn (13)}$$

where:

$T =$ Thrust

$D =$ Drag $= \frac{1}{2}\rho v^2 S C_D$

$W =$ Aircraft weight

$L =$ Lift $= \frac{1}{2}\rho v^2 S C_L$

$\mu_R =$ Runway friction resistance

Substituting for lift and drag gives:

$$X_G = T - \frac{1}{2}\rho v^2 S(C_D - \mu_R C_L) - \mu_R W \quad \text{Eqn (14)}$$

Therefore

$$X_G = a - bv^2 \quad \text{Eqn (15)}$$

where $a = T - \mu_R W$ and $b = \frac{1}{2}\rho S(C_D - \mu_R C_L)$

Therefore a can be calculated at the mid-interval velocities using a typical value of $\mu_R = 0.02$ and the AUW of 995715 N/m/s².

$$a = T_{\Delta v} - 19914.3 \quad \text{Eqn (16)}$$

To calculate b , the coefficient of drag, C_D must be established.

Using Eqns (4) and (5) the maximum C_L attainable for a wing body incidence of +4 degrees with the flaps in the take-off setting of 8 degrees is 1.2 and the maximum C_L for a clean configuration is 0.4. Therefore $\Delta C_L = 0.8$. Using the drag polar in Ref (1) for take-off at sea level with landing gear and high lift devices extended:

$$C_D = 0.043 + 0.0384(C_L)^2 + 0.02(\Delta C_L)^2 \quad \text{Eqn (17)}$$

Substituting for C_L and ΔC_L gives $C_D = 0.111$ and:

$$b = \frac{1}{2} \times 1.225 \times 193.3 \{0.111 - (0.02 \times 1.2)\} = 10.3$$

Therefore:

$$X_G = (T_{\Delta v} - 19914.3) - 10.3 \Delta v^2 \quad \text{Eqn (18)}$$

Calculating $\sum \frac{2n-1}{X_G}$ over the whole interval from $v = 0$ to $v = 60.9$ m/s = 0.000456

Therefore using Eqn (12) at AUM = 101500 kg

$$S_{GR} = \frac{1}{2} (101500) (6.09)^2 (0.000456)$$

$$\underline{S_{GR} = 858.23\text{m}}$$

The ground run distance has increased from 720m to 858m due to drag and runway resistance which is a 20% increase in runway length.

Distance between V_R and V_{LOF}

6. In order to calculate the distance between the nosewheel lifting off and the main wheels lifting off, the acceleration of the aircraft must be established:

$$\text{Accelerating Force} = X_G = ma.$$

Therefore substituting into Eqn (14):

$$a = \frac{T - \mu_R W - \frac{1}{2} \rho v^2 S (C_D - \mu_R C_L)}{m} \quad \text{Eqn (19)}$$

At $V_R = 60.9$ m/s, using Eqns (3),(4) and(5), $C_L = 2.27$ and $\Delta C_L = 0.872$ and using Eqn (17) $C_D = 0.256$. As V_{LOF} is unknown V_R is used in Eqn (19) to find a first approximation for the acceleration between V_R and V_{LOF} . Using Eqn (2) the thrust at $v = 60.9$ m/s = 260143.3 N. Therefore:

$$a = \frac{260143.7 - (0.02 \times 101500 \times 9.81) - \frac{1}{2} \times 1.225 \times (60.9)^2 \times 193.3 (0.256 - (0.02 \times 2.27))}{101500}$$

$$a = 1.46 \text{ ms}^{-2}$$

It is assumed that it takes approximately 3 seconds to transition from V_R to V_{LOF} , therefore $V_{LOF} = 60.9 + (3 \times 1.46) = 65.27$ m/s. A second iteration is performed using the mean velocity between V_R and V_{LOF} to find a mean acceleration and therefore a new value for V_{LOF} .

Using $V_{\text{mean}} = 63.09$ m/s, $T = 256896$ N, $C_L = 2.11$ and $\Delta C_L = 0.8$. Therefore $C_D = 0.227$ and:

$$a = 1.48 \text{ ms}^{-2}$$

and $V_{LOF} = 60.9 + (3 \times 1.475) = 65.33$ m/s. Therefore:

$$\underline{V_{LOF} = 65.33 \text{ m/s}}$$

The distance covered between V_R and $V_{LOF} = 3 \times \text{mean velocity} = 3 \times \frac{1}{2}(60.9 + 65.33) = 3 \times 63.115 = 189.34$ m.

$$\underline{S_{VR-VLOF} = 189.3 \text{ m}}$$

Distance from V_{LOF} to Screen Height

7. There is a transition phase after lift-off during which the flight path is curved upwards and the lift is greater than the weight. After transition the aircraft climbs steadily at an angle γ_C which is assumed to be constant for at least a few seconds. The point at which the screen height (h_{SC}) is reached may be either before or after the end of the curved transition and the horizontal distance to this point from the point of lift-off is the airborne distance (S_a) which is required to determine the total take-off distance $S_g + S_a$ where $S_g = S_{GR} + S_{VR-VLOF}$.

During transition there is some increase in speed and also a variation of lift coefficient, so that the drag does not remain constant. However for the purpose of estimating the distance S_a , the motion of the aircraft can be represented adequately by specifying mean values of the speed and lift coefficient and regarding these as constants.

The velocity at screen height (V_3) must be greater than V_2 , which is the minimum allowable velocity at screen due to engine failure on take-off. In accordance with Def

Stan 00-970 $V_2 \geq 1.15V_s$. Therefore if $V_s = 58$ m/s the minimum velocity for $V_2 = 66.7$ m/s and a reasonable approximation for $V_3 = 1.25V_s = 72.5$ m/s.

Therefore the mean speed during transition $V_{\text{mean}} = \frac{1}{2}(V_{\text{LOF}} + V_3) = 68.9$ m/s and the C_L at that speed = 1.77. Ref (3) provides 2 formula for establishing the airborne distance, S_a , depending on whether the screen height is reached before or after the end of the curved transition.

When the screen height is reached before the end of the curved transition:

$$\gamma_c > (2h_{sc}/R)^{\frac{1}{2}} \quad \text{Eqn (20)}$$

where the climb angle $\gamma_c = (f - \beta)$ if the angle is assumed to be small ($f = T/W$ and $\beta = C_D/C_L$) and R is the radius of the circular arc described by the aircraft flight path during transition.

If the condition in Eqn (20) is satisfied then Eqn (21) is used to calculate the airborne distance to screen height:

$$S_a = (2Rh_{sc})^{\frac{1}{2}} \quad \text{Eqn (21)}$$

However, when the screen height is reached after the end of the curved transition and Eqn (22) is satisfied:

$$\gamma_c < (2h_{sc}/R)^{\frac{1}{2}} \quad \text{Eqn (22)}$$

Then Eqn (23) is used to calculate the airborne distance to screen height:

$$S_a = \frac{1}{2}\gamma_c R + \frac{h_{sc}}{\gamma_c} \quad \text{Eqn (23)}$$

Therefore in order to calculate S_a , γ_c and R must be determined.

If $V_{\text{mean}} = 68.9$ m/s, $T = 254189$ N, $W = 995715$ N, $C_L = 1.77$, $\Delta C_L = 0.8$ and

$C_D = 0.176$. Then $f = T/W = 254189 / 995715 = 0.255$ and $\beta = C_D/C_L = 0.176/1.77 = 0.0995$.

Therefore $\gamma_c = 0.255 - 0.0995 = 0.1558$

R is calculated using the following equation from Ref (3):

$$R = \frac{2m}{(\rho S \Delta C_L)} \quad \text{Eqn (24)}$$

where $S = 193.3 \text{ m}^2$ and $m = 101500 \text{ kg}$.

ΔC_L for a maximum effort take-off can also be found empirically from Ref (3) by using:

$$\Delta C_L = [(V/V_s)^2 - 1][C_{L_{\max}}\{(V_s/V)^2 - 0.53\} + 0.38] \quad \text{Eqn (25)}$$

Practical experience has shown that in a normal take-off the ΔC_L employed by the pilot is likely to be about half the value obtained in a maximum effort take-off. Therefore it is assumed that the value of ΔC_L given by Eqn (25) is divided by 2 for a normal take-off.

If $V_s = 58 \text{ m/s}$, $C_{L_{\max}} = 2.5$ and $V = 68.9 \text{ m/s}$ then using Eqn (25) $C_L = 0.34$. Therefore:

$$R = \frac{2 \times 101500}{1.225 \times 193.3 \times 0.17} = 5042.24$$

and

$$(2RhSC)^{\frac{1}{2}} = \sqrt{\frac{(2 \times 10.67)}{5042.24}} = 0.065$$

Therefore $\gamma_c > (2h_{sc}/R)^{\frac{1}{2}}$ as $0.1558 > 0.065$ and Eqn (21) is used to calculate the airborne distance to screen height:

$$S_a = (2 \times 5042.24 \times 10.67)^{\frac{1}{2}} = 328.03 \text{ m}$$

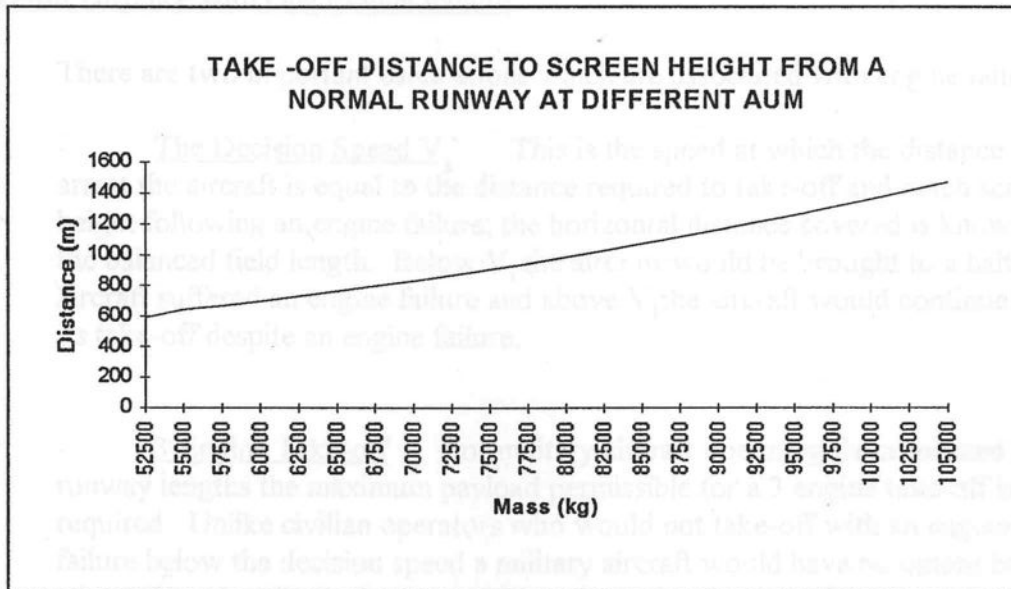
$$\underline{S_a = 328\text{m}}$$

Total Take-off Distance

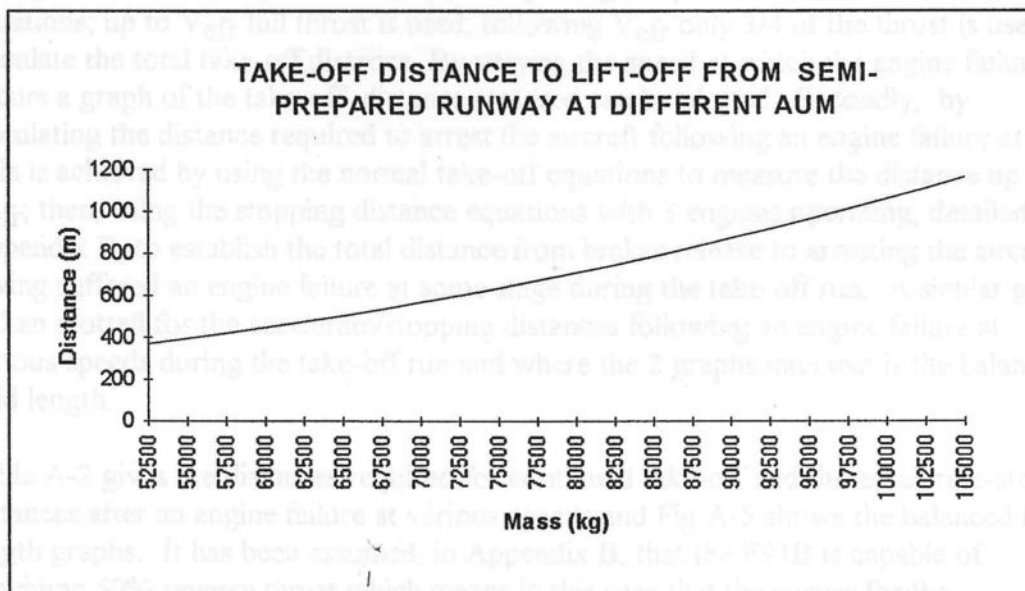
8. Total take-off distance required to reach screen height $S_{SC} = S_{GR} + S_{VR-LOF} + S_a$
 $= 858.23 + 189.34 + 328.03 = 1375.59 \text{ m}$

$$\underline{S_a = 1375.6\text{m}}$$

The design specification Ref (1) required a total take-off distance to screen height of 1500m for normal operations with 4 engines operating. At its heaviest possible AUM of 101500 kg the F93B is capable of satisfying that requirement with 124m to spare. Fig A-3 shows the total take-off distance to screen height required at different AUM. Although the F93B achieves its take-off design specification with distance to spare the take-off distance required will vary not only with weight but also with altitude, temperature and the runway resistance due to the environment. Therefore further calculations would be required before it could be said that the F93B could take-off from a 1500m runway at its AUM no matter what the conditions.

**Figure A-3**

The additional take-off requirement laid down in Ref (1) is that for emergency operations the F93B should be able to take-off from a semi-prepared strip of 900m, although the 900m does not include the horizontal distance to screen height. Fig A-4 shows the ground run required for different AUM from a semi-prepared runway with an increased runway resistance of 0.03.

**Figure A-4**

The maximum AUM permissible in order to lift-off within 900m is 91800 kg which means the maximum fuel and payload combination is 39430 kg.

Take-off Distance with One Engine Failure

9. There are two important calculations which are associated with engine failure:

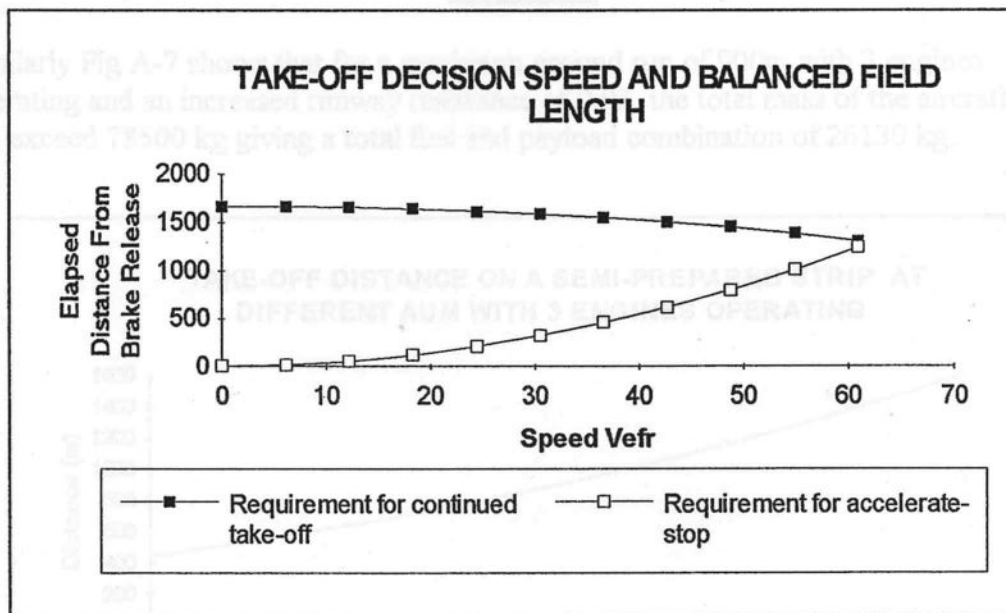
- The Decision Speed V_1 This is the speed at which the distance to arrest the aircraft is equal to the distance required to take-off and reach screen height following an engine failure; the horizontal distance covered is known as the balanced field length. Below V_1 the aircraft would be brought to a halt if the aircraft suffered an engine failure and above V_1 the aircraft would continue with its take-off despite an engine failure.
- 3 Engine Take-off For military aircraft operating from reduced runway lengths the maximum payload permissible for a 3 engine take-off is required. Unlike civilian operators who would not take-off with an engine failure below the decision speed a military aircraft would have no option but to take-off in a hostile environment. Therefore an indication of the maximum payload permissible in order to achieve a successful take-off with an engine failure is extremely useful.

Balanced Field Length

10. The balanced field length is calculated in 2 parts. Firstly, by calculating the distance required to take-off despite suffering an engine failure (V_{eff}) at some stage during the take-off run. This is calculated by using the previously established take-off equations; up to V_{eff} full thrust is used, following V_{eff} only 3/4 of the thrust is used to calculate the total take-off distance. By varying the speed at which the engine failure occurs a graph of the take-off distance required can be plotted. Secondly, by calculating the distance required to arrest the aircraft following an engine failure at V_{eff} . This is achieved by using the normal take-off equations to measure the distance up to V_{eff} ; then, using the stopping distance equations with 3 engines operating, detailed in Appendix B, to establish the total distance from brakes release to arresting the aircraft having suffered an engine failure at some stage during the take-off run. A similar graph is then plotted for the accelerate/stopping distances following an engine failure at various speeds during the take-off run and where the 2 graphs intersect is the balanced field length.

Table A-2 gives the distances required for continued take-off and the accelerate-stop distances after an engine failure at various speeds and Fig A-5 shows the balanced field length graphs. It has been assumed, in Appendix B, that the F93B is capable of providing 50% reverse thrust which means in this case that the curves for the requirement for continued take-off and the requirement for accelerate-stop distance do not actually intersect below V_R , therefore under normal operating conditions the F93B would always brake to a halt having suffered an engine failure during the take-off ground run.

BALANCED FIELD LENGTH DATA		
Engine Failure Speed (V _{eff})	Requirement for Continued Take-off	Requirement for Accelerate-Stop
0	1668.1m	0
6.1 m/s	1664.7m	13.3m
12.2 m/s	1655m	52.6m
18.3 m/s	1638.8m	117.3m
24.4 m/s	1616m	206.6m
30.5 m/s	1586.5m	320m
36.6 m/s	1549.6m	456.7m
42.7 m/s	1504.7m	616.7m
48.8 m/s	1451m	799.6m
54.9 m/s	1386.9m	1006.2
61 m/s	1307.6m	1242.7m

Table A-2**Figure A-5**

3-Engine Take-off (Maximum Payload)

11. Using the take-off equations already established the take-off distance with only 3 engines operating is calculated using only 3/4 of the available thrust. Fig A-6 shows the take-off distance required with only 3 engines operating at different AUM. In order to take-off from a 1500m runway the total mass of the aircraft must be no more than 93 030 kg giving a maximum combination of fuel and payload of 40660 kg. Therefore, the

loss of an engine does not appear to be a great penalty for the F93B as regards loss of capability.

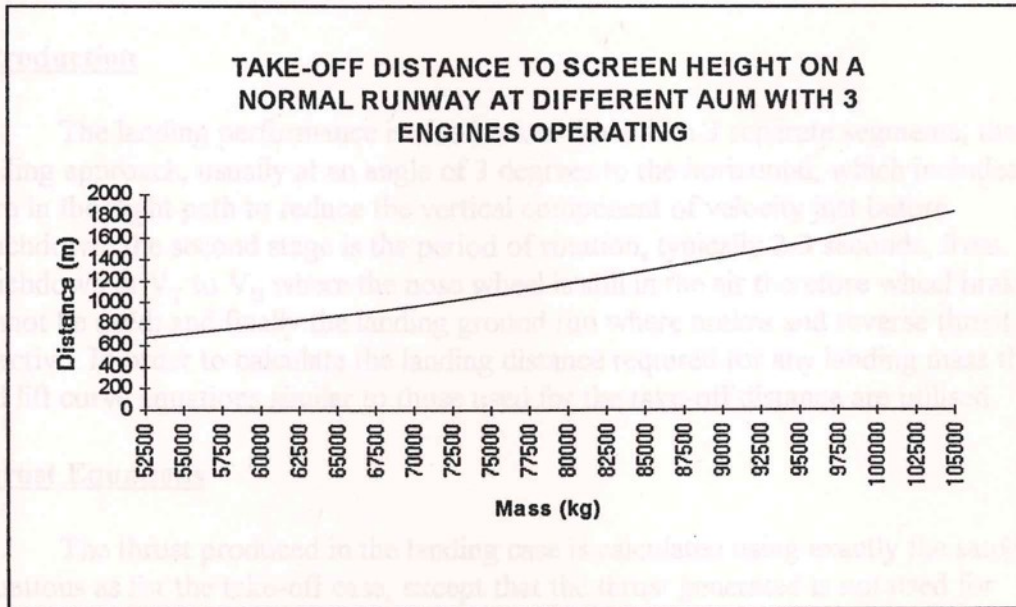


Figure A-6

Similarly Fig A-7 shows that for a maximum ground run of 900m with 3 engines operating and an increased runway resistance of 0.03, the total mass of the aircraft must not exceed 78500 kg giving a total fuel and payload combination of 26130 kg.

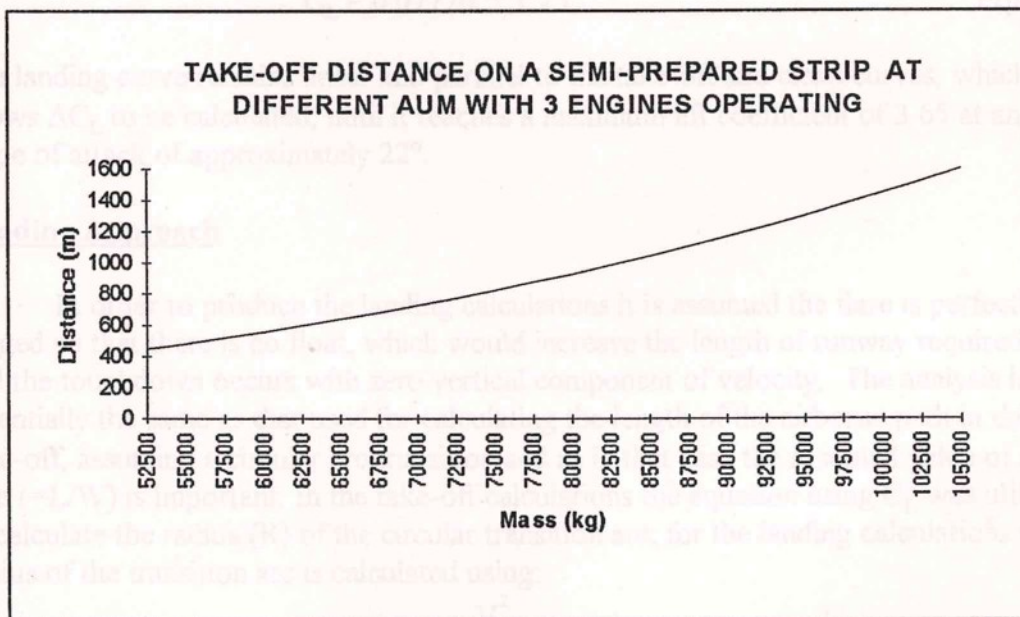


Figure A-7

APPENDIX B

LANDING PERFORMANCE

Introduction

1. The landing performance is also broken down into 3 separate segments; the landing approach, usually at an angle of 3 degrees to the horizontal, which includes the flare in the flight path to reduce the vertical component of velocity just before touchdown; the second stage is the period of rotation, typically 2-3 seconds, from touchdown at V_T to V_B where the nose wheel is still in the air therefore wheel brakes cannot be used; and finally the landing ground run where brakes and reverse thrust are effective. In order to calculate the landing distance required for any landing mass thrust and lift curve equations similar to those used for the take-off distance are utilised.

Thrust Equations

2. The thrust produced in the landing case is calculated using exactly the same equations as for the take-off case, except that the thrust generated is not used for forward thrust but required for reverse thrust. It has been assumed that the F93B can utilise 50% of its generated thrust in a reverse thrust capability.

Lift Curve Slope Equation

3 The lift curve slope for the landing configuration is:

$$C_L = 0.0777\alpha + 1.75 \quad \text{Eqn (1)}$$

The landing curve remains linear and parallel to the take-off and clean curves, which allows ΔC_L to be calculated, until it reaches a maximum lift coefficient of 3.65 at an angle of attack of approximately 22° .

Landing Approach

4. In order to produce the landing calculations it is assumed the flare is perfectly judged so that there is no float, which would increase the length of runway required, and the touchdown occurs with zero vertical component of velocity. The analysis is essentially the same as that used for calculating the length of the airborne path in the take-off, assuming a circular arc transition and as in that case the assumed value of ΔC_L or n ($=L/W$) is important. In the take-off calculations the equation using C_L was utilised to calculate the radius (R) of the circular transition arc; for the landing calculations the radius of the transition arc is calculated using:

$$R = \frac{V^2}{g(n-1)} \quad \text{Eqn (2)}$$

where Ref (3) recommends that n should be taken as 1.1 and $V=V_F$, which is the mean speed in the flare:

$$V_F = \frac{1}{2}(V_A + V_T) \quad \text{Eqn (3)}$$

where V_A is the approach speed and V_T is the touchdown speed.

V_A is usually taken as $1.3V_S$ and V_S occurs at max C_L on approach to landing (flaps at 12°) = 3.65. Therefore at the design landing mass of 78962 kg (Take-off mass of 101 500 kg fuel used over range of 5500km) $V_A = 55 \text{ ms}^{-1}$. V_T must be less than V_A , therefore assuming $V_T = 0.9V_A = 49.5 \text{ ms}^{-1}$

$$V_F = \frac{1}{2}(55 + 49.54) = 52.3 \text{ m/s}$$

Substituting for V_F in Eqn (2)

$$R = \frac{52.3^2}{9.81(1.1 - 1)} = 2787 \text{ m}$$

The height at the flare point is h_f where

$$h_f = \frac{1}{2} R \gamma_A \quad \text{Eqn (4)}$$

and in the usual case where $h_f < h_{sc}$ the total airborne distance from the threshold (screen height) to touchdown is:

$$S_a = \frac{h_{sc}}{\gamma_A} + \frac{1}{2} R \gamma_A \quad \text{Eqn (5)}$$

In the exceptional case where $h_f > h_{sc}$ which may occur when a steep approach angle is used (This approach is more likely to be used in a military situation due to the tactical advantages of remaining out of range for as long as possible).

$$S_a = (2Rh_{sc})^{\frac{1}{2}} \quad \text{Eqn (6)}$$

For the F93B at an approach angle (γ_A) of 3 degrees = 0.05236 radians.

$$h_f = \frac{1}{2} \times 2787 \times (0.05236)^2 = 3.82 \text{ m}$$

Therefore $h_f < h_{sc}$ and

$$S_a = \frac{10.67}{0.05236} + \frac{1}{2} \times 2787 \times 0.05236 = 277 \text{ m}$$

$$\underline{S_a = 277 \text{ m}}$$

The first approximation for V_T can be improved by iteration, providing a reasonable estimation of the mean net drag ($D-F$) can be made, by using:

$$(V_A)^2 - (V_T)^2 = 2g[(\beta - f)S_a - h_{SC}] \quad \text{Eqn (7)}$$

where $\beta = C_D/C_L$

At $V_F = 52.3 \text{ ms}^{-1}$, $C_L = 2.39$ and $\Delta C_L = 1.65$. Therefore using the drag polar for the landing configuration:

$$C_D = 0.11 + 0.037(C_L)^2 + 0.02(\Delta C_L)^2 = 0.38$$

Therefore $\beta = 0.38/2.39 = 0.157$

To calculate $f = T/W$ a value for the thrust at $V = 52.3 \text{ ms}^{-1}$ must be calculated.

However, unlike the take-off condition where the thrust is found using Eqn (1) or (2) in Appendix A, in the landing condition the engines are virtually at idle and are providing very little forward thrust. It is assumed in this case that the engines provide 1% forward thrust when at idle. Therefore at $V = 52.3 \text{ ms}^{-1}$ 1% of $T = 266543 \text{ N}/100 = 2665.43 \text{ N}$.

At the landing mass of 78 962 kg $f = 2665.43/774617 = 0.00344$.

Substituting into Eqn (7):

$$(55)^2 - (V_T)^2 = 2 \times 9.81[(0.157 - 0.00344)277 - 10.67]$$

$$V_T = 49.02 \text{ m/s}$$

Distance from Touchdown to Commencement of Braking (S_{T-B})

6. The distance travelled from the point of touchdown at V_T to the point at which braking can commence V_B , usually the point at which the nose wheel has touched down, is calculated by assuming a decrease in velocity of 1.5% per second. The rotation period lasts on average 3 seconds therefore if $V_T = 49.02 \text{ ms}^{-1}$ then $V_B = 46.82 \text{ ms}^{-1}$. The distance covered during this period is the average velocity multiplied by 3. $V_{\text{average}} = 47.92 \text{ m/s}$, therefore, $S_{T-B} = 3 \times 47.92 = 143.7 \text{ m}$.

$$S_{T-B} = 143.7 \text{ m}$$

Ground Run Distance (S_{GR})

7. The equation used for calculating the length of the ground run after the start of braking is essentially the same as the equation used for take-off.

$$S_{GR} = \frac{1}{2} m (\Delta V)^2 \sum \frac{(2n-1)}{X_G} \quad \text{Eqn (8)}$$

except that the accelerating force X_G is negative in this case.

$-X_G = F + D + G$ where F is the reverse thrust and it is assumed in this case that the reverse thrust is applied simultaneously with the spoilers and brakes at V_B . D is determined by

a nearly constant C_D throughout the ground run. The coefficient is considerably higher than it is in the take-off ground run because of the greater flap angle. The other important term contributing to X_G is the braking force G which is defined as the component of the force exerted by the ground on the wheels. The braking coefficient $\mu_B = G/Z$ where Z is the load on the wheels normal to the runway, i.e the weight of the aircraft. The maximum value of μ_B occurs when the tyre is on the point of skidding. Ref (3) shows that the average maximum value of μ_B decreases with increasing speed and that there is a significant difference between braking on a dry and a wet runway. At $V=70\text{ms}^{-1}$ the maximum braking coefficient on a dry runway is 0.5, on a wet runway μ_B is 0.175. In order to calculate the landing run it is assumed in this case that the runway is dry and that μ_B is a constant value of 0.3 which should also allow for any brake torque limit.

To calculate X_G

$$X_G = T + \frac{1}{2}\rho V^2 S C_D + \mu_B W \quad \text{Eqn (9)}$$

Unlike the take-off case where there would be a lift component in Eqn (9) the spoilers are used to dump all the lift.

Therefore

$$X_G = a + bV^2 \quad \text{Eqn (10)}$$

where $a = T + \mu_B W$ and $b = \frac{1}{2}\rho S C_D$

Using 10 intervals of $\Delta V_B = 4.68\text{ms}^{-1}$ each and taking the mid position the thrust settings are detailed in Table 2.

Therefore (a) can be calculated at the mid-interval velocities using:
 $\mu_B = 0.3$ and an AUW of $78962 \times 9.81 = 774617.2\text{N}$

$$a = T_{RAV} + 232385 \quad \text{Eqn (11)}$$

Total Landing Distance

8. The total landing distance from a screen height of 10.67m is $S_L = S_G + S_{FB} + S_{CL}$
 $= 275.7 + 143.8 + 216.8 = 637.3\text{m}$

$$S_L = 637.3\text{m}$$

VELOCITY AT MID INTERVAL POSITION	REVERSE THRUST AT MID INTERVAL
44.48 m/s	13141474.1 N
39.79 m/s	129927.5 N
35.11 m/s	128380.9 N
30.43 m/s	126834.4 N
25.75 m/s	125287.7 N
21.07 m/s	123741.1 N
16.39 m/s	122194.5 N
11.70 m/s	120647.9 N
7.02 m/s	119101.4 N
2.34 m/s	117554.8 N

Table B-1

To calculate b , C_D must be established.

From V_B to $V = 0$ the maximum C_L attainable, for a wing body incidence of +4 degrees with the flaps in the landing setting of 12 degrees, is 2.061, $\Delta C_L = 1.65$ and $C_D = 0.322$.

Therefore

$$b = \frac{1}{2} \times 1.225 \times 193.3 \times 0.322 = 38.12$$

and:

$$X_G = T_{RAV} + 232385 + 38.12(\Delta V)^2 \quad \text{Eqn (12)}$$

Over the whole interval from $V_B = 46.8$ m/s to $V = 0$

$$\sum \frac{(2n-1)}{X_G} = 0.0002505 \quad \text{Eqn (13)}$$

Therefore at the design mass of 78962 kg using Eqn (8):

$$S_{GR} = \frac{1}{2} (78962)(4.68)^2(0.0002505)$$

$$\underline{S_{GR} = 217\text{m}}$$

Total Landing Distance

8. The total landing distance from a screen height of 10.67m is $S_L = S_a + S_{T-B} + S_{GR}$
 $= 276.7 + 143.8 + 216.8 = 637.3\text{m}$

$$\underline{S_L = 637\text{m}}$$

Fig B-1 shows a graph of landing distances required at various landing masses and braking coefficients with 4 engines in reverse thrust.

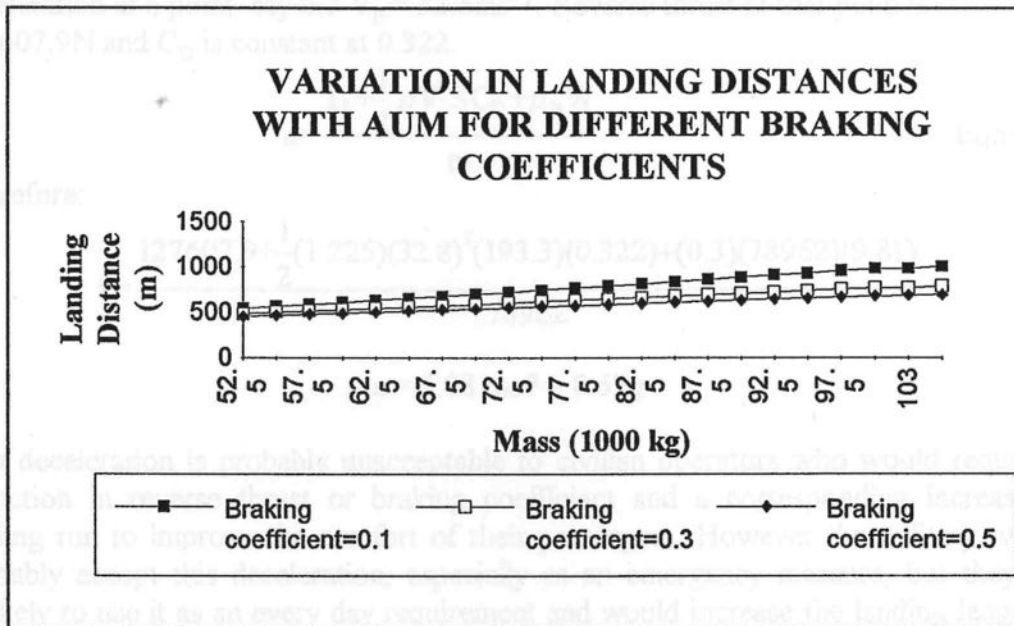


Figure B-1

Fig B-2 shows the landing distances required with only 3 engines operating in reverse thrust.

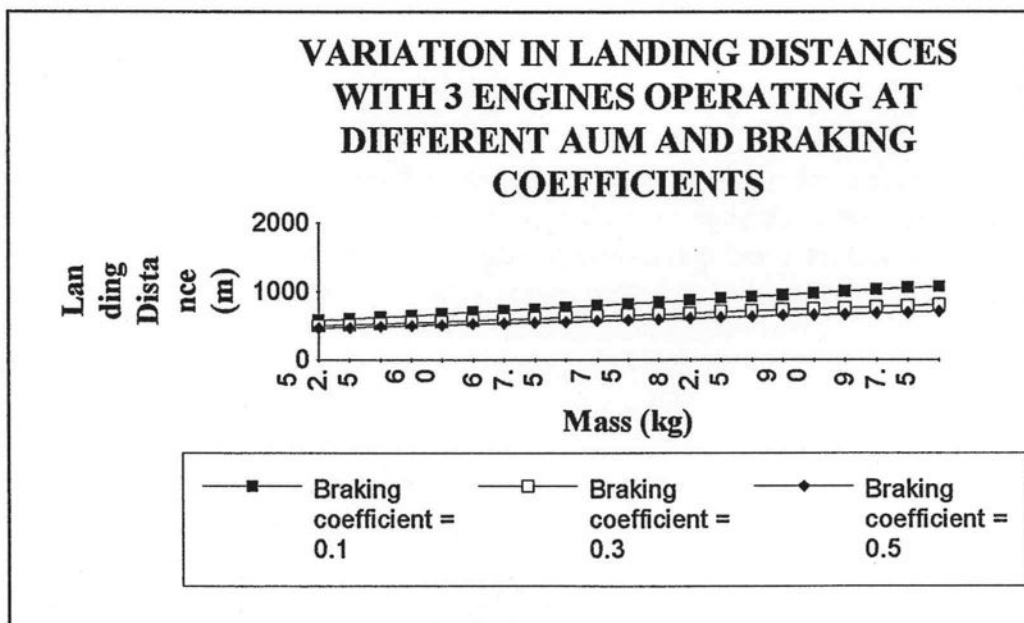


Figure B-2

Although there is no design requirement for a particular landing field length the F93B has a fairly short landing run due to the significant reverse thrust capability. A civilian

operator would probably be concerned with the deceleration experienced by the passengers and crew during the landing run. Using Eqn (14) to calculate the deceleration at a point, say $0.7 V_B = 32.8 \text{ ms}^{-1}$. Reverse thrust at that point is 127607.9 N and C_D is constant at 0.322 .

$$a = \frac{T_R + \frac{1}{2} \rho V^2 S C_D + \mu_B W}{m} \quad \text{Eqn (14)}$$

Therefore:

$$a = \frac{127607.9 + \frac{1}{2} (1.225) (32.8)^2 (193.3) (0.322) + (0.3) (78962) (9.81)}{78962}$$

$$a = 5.08 \text{ ms}^{-2} = 0.52g$$

This deceleration is probably unacceptable to civilian operators who would require a reduction in reverse thrust or braking coefficient and a corresponding increase in landing run to improve the comfort of their passengers. However the military would probably accept this deceleration, especially as an emergency measure, but they are unlikely to use it as an every day requirement and would increase the landing length in order to save the fatigue life on the airframe and brakes.

Climb Angle

The climb angle (γ) is calculated using:

$$\tan \gamma = \frac{T}{W} - \frac{C_D}{C_L} \quad \text{Eqn (1)}$$

where T is the engine thrust according to the manufacturer data for that particular altitude and speed. Although the manufacturer data covering the climb phase contained limited thrust information, which necessitated interpolation to provide engine thrust data at the required heights, it was considered more practical to use the manufacturer details than try and determine a thrust equation similar to the ones generated for the take-off performance. The engine thrust data provided is in lbf, therefore weight thrust is $\text{lbf} \times 4.448 \times 4 \times 1.7$ where 1.7 is the safety factor required, specified in Ref (1). In order to calculate C_L and C_D for each interval the equivalent airspeed of 250 knots must be converted into true airspeed:

$$V_{TA} = \frac{V_e}{\sqrt{\sigma}} \quad \text{Eqn (2)}$$

where σ is the relative density at altitude $= \rho/\rho_0$.

APPENDIX C

CLIMB, CRUISE AND DESCENT PERFORMANCE

Introduction

1. In order to establish the landing weight, the fuel used during the climb, cruise and descent segments must be established. This is calculated by multiplying the specific fuel consumption (SFC) by the thrust and the time period ($SFC \times T \times t$). However the SFC and thrust vary throughout the mission as the weight of the aircraft decreases and, therefore, must be calculated over several intervals.

CLIMB PERFORMANCE

2. The climb up to 10km has been calculated over intervals of 1524m (5000ft) until the last interval which is 856m (2800ft). For each interval the angle of climb, rate of climb, ground distance covered and time to climb are calculated. The SFC for each section is also determined in order to adjust the weight of the aircraft for the next sections calculations. It is assumed that the aircraft climbs at a max climb speed of 250 knots EAS.

Climb Angle

3. The climb angle (γ) is calculated using:

$$\tan \gamma_c = \frac{T}{W} - \frac{C_D}{C_L} \quad \text{Eqn (1)}$$

where T is the engine thrust according to the manufacturers data for that particular altitude and speed. Although the manufacturers data covering the climb phase contained limited thrust information, which sometimes required interpolation to provide engine thrust data at the required heights, it was considered more practical to use the manufacturers details than try and determine a thrust equation similar to the ones generated for the take-off performance. The engine thrust data provided is in lbf, therefore total thrust is $\text{lbf} \times 4.448 \times 4 \times 1.7$ where 1.7 is the scaling factor required, specified in Ref (1). In order to calculate C_L and C_D for each interval the equivalent airspeed of 250 knots must be converted into true airspeed:

$$V_{TR} = \frac{V_e}{\sqrt{\sigma}} \quad \text{Eqn (2)}$$

where σ is the relative density at altitude = ρ/ρ_0 .

Therefore:

$$C_L = \frac{W}{\frac{1}{2} \rho V_{TR}^2 S} \quad \text{Eqn (3)}$$

and using the drag polar given in Ref (1) for the clean configuration:

$$C_D = 0.21 + 0.39(C_L)^2 \quad \text{Eqn (4)}$$

Climb Angle for the first interval:

$$\text{Thrust} = 7955 \text{ lbf} \times 4 \times 4.448 \times 1.7 = 240610 \text{ N}$$

$$\text{Weight} = 101500 \times 9.81 = 995715 \text{ N}$$

$$\text{Using Eqn (2) and (3): } C_L = \frac{995715}{\frac{1}{2} \times 1.225 \times (128.5)^2 \times 193.3} = 0.51$$

$$\text{Using Eqn (4): } C_D = 0.21 + 0.39(0.51)^2 = 0.031$$

Therefore using Eqn (1)

$$\tan \gamma_c = \frac{(240610)}{(995715)} - \frac{(0.031)}{(0.51)} = 0.1805$$

$$\text{Climb Angle during First Interval} = 10.2^\circ$$

Detailed results of all the intervals are shown in para 15 Table C-1. Similar results can be obtained for 3 and 2 engine operation by using 3/4 and 1/2 of the total thrust available. Fig C-1 shows the climb angle for the F93B with 4, 3 and 2 engines operating.

$$\text{Thrust} = 7955 \text{ lbf} \times 4 \times 4.448 \times 1.7 = 240610 \text{ N}$$

$$\text{Weight} = 101500 \times 9.81 = 995715$$

$$\text{Using Eqn (2) } V_{TR} = 128.5 \text{ m/s}$$

$$\text{Using Eqn (3) } C_L = 0.51$$

$$\text{Using Eqn (4) } C_D = 0.031$$

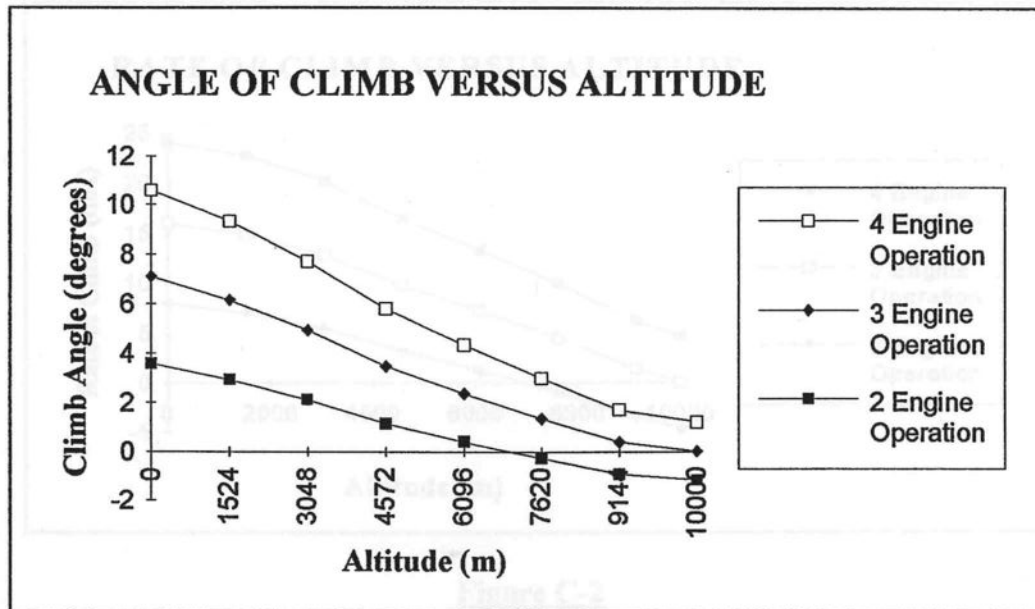
$$\text{Therefore Drag} = \frac{1}{2} \times 1.225 \times (128.5)^2 \times 193.3 \times 0.031 = 60833 \text{ N}$$

Substituting into Eqn (5):

$$V_c = \frac{(240610 - 60833)128.5}{995715} = 23.2 \text{ m/s}$$

$$\text{Rate of Climb during First Interval} = 23.2 \text{ m/s}$$

Table C-1 at para 8 shows the individual interval results and Fig C-2 shows the rate of climb during the climb with 4, 3 and 2 engines operating.

**Figure C-1****Rate of Climb**

4. The rate of climb (V_C) is calculated using:

$$V_C = \frac{(T-D)V_{TR}}{W} \quad \text{Eqn (5)}$$

where the aircraft drag (D) = $\frac{1}{2}\rho(V_{TR})^2 S C_D$.

Rate of climb during the first interval:

$$\text{Thrust} = 7955 \text{ lbf} \times 4 \times 4.448 \times 1.7 = 240610 \text{ N}$$

$$\text{Weight} = 101500 \times 9.81 = 995715$$

$$\text{Using Eqn (2)} \quad V_{TR} = 128.5 \text{ m/s}$$

$$\text{Using Eqn (3)} \quad C_L = 0.51$$

$$\text{Using Eqn (4)} \quad C_D = 0.031$$

$$\text{Therefore Drag} = \frac{1}{2} \times 1.225 \times (128.5)^2 \times 193.3 \times 0.031 = 60833 \text{ N}$$

Substituting into Eqn (5):

$$V_C = \frac{(240610 - 60833)128.5}{995715} = 23.2 \text{ m/s}$$

Rate of Climb during First Interval = 23.2 m/s

Table C-1 at para 8 shows the individual interval results and Fig C-2 shows the rate of climb during the climb with 4,3 and 2 engines operating.

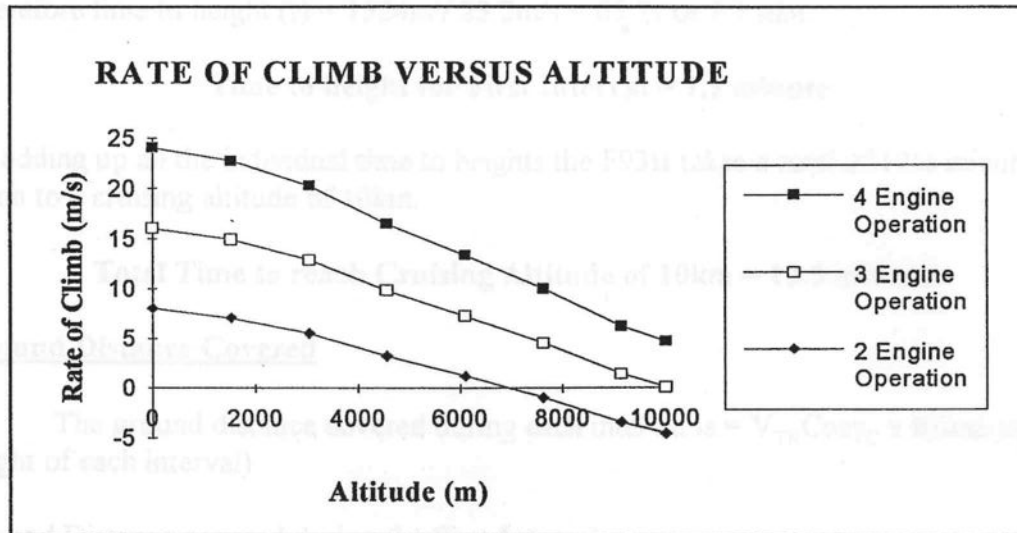


Figure C-2

Service and Operational Ceiling

4.1 The service and operational ceiling of an aircraft may be limited by the low air density at high altitudes and cannot be increased indefinitely by increasing the available thrust. Since the rate of climb of an aircraft tends to zero as the absolute ceiling is approached a more useful indication of the practicable upper limit is the maximum height at which a specified rate of climb can be maintained in steady flight. The service ceiling for an aircraft is defined in terms of a rate of climb of 0.51 m/s and the operational ceiling is defined as the height at which a rate of climb of 2.54 m/s can be maintained. For the F93B aircraft the operational and service ceilings are calculated by continuing on with the climb intervals until the required rates of climb have been reached. The manufacturers engine data, Ref (2) was limited to a height of 40000ft (12192m) at which point the rate of climb of the F93B is 0.536 m/s. Therefore, the service ceiling of the F93B can be assumed to be close enough to 40000ft (12192m). A rate of climb of 2.54 m/s occurs when the F93B is at a height of 36 875ft (11240m), therefore the operational ceiling for the F93B is 38875ft (11240m)

F93B Service Ceiling = 40 000ft
F93B Operational Ceiling = 36 875ft

Time to Height

5. The time to climb to height for each interval is calculated by dividing the vertical distance of each interval by the rate of climb, see para 4.

Time to height for the first interval:

Each interval is 5000ft (1524m) except the last interval which is only 2800ft (856m)

Therefore time to height (t) = $1524\text{m} / 23.2\text{m/s} = 65.7\text{s}$ or 1.1 min.

Time to height for First Interval = 1.1 minute

By adding up all the individual time to heights the F93B takes a total of 12½ minutes to climb to a cruising altitude of 10km.

Total Time to reach Cruising Altitude of 10km = 12.5 minutes

Ground Distance Covered

6. The ground distance covered during each interval is = $V_{TR} \cos \gamma_C \times t$ (time to height of each interval)

Ground Distance covered during the First Interval

If $V_{TR} = 128.5 \text{ m/s}$, $\gamma_C = 10.2^\circ$ and $t = 65.7\text{s}$ then :

The ground distance covered = $128.5 \times \cos 10.2^\circ \times 65.7 = 8306\text{m}$

Ground Distance Covered during the First Interval = 8.3km

Individual interval results are shown in para 8. By summing the individual results the total ground distance covered during the climb is established.

Total Ground Distance Covered during the Climb = 127.4km

The distance travelled during take-off and the climb is subtracted from the desired mission length to establish the remaining distance to be covered and hence the fuel consumption. The design range for the F93B is 5500km plus an additional 1275km capability in reserve. The ground distance covered during take-off and climb is 1.37km and 127.4km respectively giving a total of 129km. Therefore the remaining range required is 6621km.

Fuel Consumption

7. The fuel consumed during each interval is the product of the SFC, thrust and the time period ($\text{SFC} \times T \times t$).

Fuel consumed during the First Interval

Total thrust = 240610 N

The SFC for that thrust from the engine data sheets = $8.92\text{E-}06 \text{ kg/N/s}$

Duration of the first interval = 65.7 seconds

Therefore, fuel consumption = $240610 \times 8.92\text{E-}06 \times 65.7 = 141 \text{ kg}$

Fuel Consumption during the First Interval = 141 kg

The individual interval results are shown in para 15. The weight of the aircraft at the start of the next interval is, therefore, the original weight minus the fuel used over the preceding interval. The total fuel used during the climb is 1346 kg. Therefore the aircraft weight at the start of the cruise phase is $101500 - 1346 = 100154 \text{ kg}$.

CRUISE PERFORMANCE

8. In the cruise condition where thrust equals drag and lift equals weight, the fuel consumed over the design range is calculated over 9 intervals of 675km plus the last interval of $6621 - (9 \times 675) = 546 \text{ km}$. The F93B design cruising speed specified in Ref (1) is Mach 0.7 (210 ms^{-1} TAS). Therefore the time period for the first 9 intervals is $675000/210 = 3214 \text{ s}$ and $546000/210 = 2600 \text{ s}$ over the last interval.

In order to estimate the thrust, and therefore the SFC used during each interval the drag must be calculated using Eqn (3) and (4) and knowing $D = \frac{1}{2} \rho V^2 S C_D = T$. The manufacturers data was interpolated for each interval thrust setting to provide the SFC during each interval and therefore the fuel used is the SFC x thrust x t and thus the total fuel consumed is the sum of all the interval estimates.

Fuel used during the First Interval of the Cruise Segment

The weight of the aircraft at the start of the cruise = $100154 \times 9.81 = 982514 \text{ N}$ and the cruising velocity is 210 m/s TAS . Therefore at a height of 10 km where $\rho = 0.4105 \text{ kg/m}^3$:

$$C_L = \frac{982514}{\frac{1}{2}(0.4105)(210)^2(193.3)} = 0.56$$

and:

$$C_D = 0.021 + 0.039(0.56)^2 = 0.033$$

Therefore the Drag = $\frac{1}{2} \times 0.4105 \times (210)^2 \times 193.3 \times 0.033 = 58260 \text{ N}$

Net thrust in lbf = $58260 / (4 \times 4.448 \times 1.7) = 1926 \text{ lbf}$ and the corresponding fuel flow and SFC from the engine data is 0.127 kg/s/eng and $1.54\text{E-}05 \text{ kg/N/s}$ respectively.

Therefore the fuel consumed = $1.54\text{E-}05 \text{ kg/N/s} \times 58260 \text{ N} \times 3214 \text{ s} = 2884 \text{ kg}$.

Fuel Used during First Interval of Cruise Segment = 2 884 kg

The mass of the aircraft for the start of the second interval of the cruise segment = $100154 - 2884 = 97270 \text{ kg}$. By completing the calculations for the remaining intervals the aircraft mass at the end of the last interval is the also the mass of the aircraft at the

top of the descent = 79311 kg. For the design case the total fuel used during the climb is 1346 kg and the total fuel used during the cruise on a design mission = 20843 kg.

DESCENT PERFORMANCE

9. The descent performance is calculated in exactly the same way that the climb performance is calculated except that instead of a positive climb angle the angle is negative or is referred to as the descent angle and the rate of climb is also negative or called the rate of descent. However the major difference in calculating the descent performance is that the engines are no longer providing anywhere near maximum thrust. As there was no descent information available from Rolls Royce on the RB509-18 engine existing engine data from other aircraft such as the ATP and Airbus was scrutinised and it appeared that during descent the engines were often throttled back to provide only enough power for the secondary systems. In some cases this required as little as 2% of the available engine thrust. Therefore it is assumed that only 2% of the available engine thrust is used during the descent phase. It is also assumed that the descent speed is a constant 250 KEAS which is not an unrealistic figure for a military transport aircraft.

Descent Angle

10. The descent angle is calculated using Eqn (1), (2), (3) and (4):

Descent Angle for the first interval:

$$\text{Thrust} = 2628 \text{ (lbf at 10km)} \times 4 \times 4.448 \times 1.7 \times 0.02 = 1590 \text{ N}$$

$$\text{Weight} = 79311 \times 9.81 = 778041 \text{ N}$$

$$\rho = 0.4105$$

$$V_{TR} = 210 \text{ m/s at 10km altitude.}$$

Therefore:

$$C_L = \frac{778041}{\frac{1}{2} \times 0.4105 \times (210)^2 \times 193.3} = 0.44$$

$$C_D = 0.21 + 0.39(0.44)^2 = 0.029$$

and:

$$\text{TAN } \gamma_c = \frac{(52.56)}{(778041)} - \frac{(0.029)}{(0.44)} = 0.065$$

$$\text{Descent Angle during First Interval} = 3.8^\circ$$

Rate of Descent

11. The rate of descent is calculated using Eqn (5):

Rate of Descent during the first interval:

$$\text{Thrust} = 2628 \text{ (lbf at 10km)} \times 4 \times 4.448 \times 1.7 \times 0.02 = 1590 \text{ N}$$

$$\text{Weight} = 79311 \times 9.81 = 778041 \text{ N}$$

$$\text{Using Eqn (2) } V_{TR} = 210 \text{ m/s at 10km altitude.}$$

$$\text{Using Eqn (3) } C_L = 0.44$$

$$\text{Using Eqn (4) } C_D = 0.029$$

$$\text{Therefore Drag} = \frac{1}{2} \times 0.4105 \times (210)^2 \times 193.3 \times 0.029 = 50294 \text{ N}$$

and:

$$V_c = \frac{(1590 - 50294)210}{778041} = 13.15 \text{ m/s}$$

$$\text{Rate of Descent during the First Interval} = 13.15 \text{ m/s}$$

Table C-1 at para 15 shows the individual interval results.

Time to Height

12. The vertical distance in the first interval of the descent is only 856m. Therefore time to descend for the first interval is $856 / 13.15 = 65.1 \text{ s}$ or 1.1 min.

$$\text{Time to Descend for the First Interval} = 1.1 \text{ minute}$$

By adding up all the individual times to descend through each interval the F93B takes a total of 18.4 minutes to descend from an altitude of 10km.

$$\text{Total Time to Descend from 10km} = 18.4 \text{ minutes}$$

Ground Distance Covered

13. The ground distance covered is calculated using $V_{TR} \cos \gamma_C \times t$:

Ground Distance covered during the first interval:

$$V_{TR} = 210 \text{ m/s, } \gamma_C = 3.8^\circ \text{ and } t = 65.1 \text{ s then the ground distance covered} = 210 \times \cos 3.8^\circ \times 65.1 = 13.7 \text{ km}$$

$$\text{Ground Distance Covered during the First Interval} = 13.7 \text{ m}$$

Individual interval results are shown in para 15. By summing the individual results the total ground distance covered during the descent is established.

$$\text{Total Ground Distance Covered during the Descent} = 155.2 \text{ km}$$

Fuel Consumption

14. During the descent the fuel flow and consequently the SFC are also reduced due to the decrease in thrust required. Again other aircraft data was considered and in general the fuel flow for engines in the descent phase was usually 20% of the climb fuel flow. Therefore it is assumed that the fuel flow for the F93B is 20 % of the fuel flow required during the climb phase.

Fuel consumed during the first descent interval:

Total thrust = 1590 N, fuel flow = $0.157623 \times 0.2 = 0.031525$ kg/s/eng and SFC = 0.00014 kg/N/s at 10km and $t = 65.1$ s. Therefore $SFC \times T \times t = 14.5$ kg.

Fuel Consumed during the First Interval = 14.5kg

The individual interval results are shown in para 15. The total amount of fuel used during the descent is the sum of the individual results = 349kg.

Total Fuel Consumed during the Descent = 349 kg

RESULTS

15. Table C-1 contains the individual interval results for the design mission . The results will vary according to the take-off weight of the aircraft and the climb and cruise speed of the aircraft.

3048	79123	3.7	9.5	2.7	24	72
1524	79051	3.6	8.7	2.6	24.7	83
0	78962	-	-	-	-	-
10mi				18.4mi	155.9km	349kg
TOTAL				2.4hrs	55900m	2153kg

Table C-1

INDIVIDUAL INTERVAL RESULTS						
Alt (m)	Mass (kg)	Climb Angle (degrees)	Rate of Climb (m/s)	Time to Climb (mins)	Ground Distance (km)	Fuel Used (kg)
CLIMB						
0	101500	10.2	23.2	1.1	8.3	141
1524	101359	9	21.9	1.2	9.5	177
3048	101182	7.4	19.5	1.3	11.6	174
4572	101008	5.6	15.8	1.6	15.5	193
6096	100815	4.1	12.7	2	21	209
7620	100606	2.8	9.5	2.7	30.5	253
9144	100353	1.6	5.8	2.4	31	199
Total		-	-	12.3 mins	127.4km	1346kg
CRUISE						
10000	100154	-	-	6.9hrs	5217.4km	20843kg
	Mass (kg)	Descent Angle (degrees)	Rate of Descent (m/s)	Time to Descend (mins)	Ground Distance (km)	Fuel Used (kg)
DESCENT						
10000	79311	3.6	13.1	1.1	13.7	14
9144	79297	3.8	13.8	1.8	23	30
7620	79267	3.7	12.5	2	23.2	38
6096	79229	3.7	11.4	2.2	23.4	47
4572	79182	3.7	10.4	2.4	23.7	59
3048	79123	3.7	9.5	2.7	24	72
1524	79051	3.6	8.7	2.9	24.2	89
0	78962	-	-	-	-	
total		-	-	18.4mins	155.2km	349kg
TOTAL				7.4hrs	5500km	22538kg

Table C-1

The information which is required for the fatigue analysis programme must be split into several intervals covering the climb, cruise and descent stages of the aircraft flight profile. Following on from the intervals used in Appendix C to calculate the climb, cruise and descent statistics, the same intervals are used to derive the necessary information for the fatigue cycle information. Therefore, there are 15 flight stages throughout each aircraft profile which consist of 7 climb stages, 1 cruise stage and 7 descent stages. For each stage the altitude (Hp) and mass (M) at the start position of each interval is required plus the horizontal distance (D) covered along the ground. It is

APPENDIX D

FATIGUE CYCLE CALCULATIONS

Introduction

1. In order to assess structural fatigue certain performance characteristics of the F93B are required. These characteristics are used as input variables to a fatigue analysis method used by various structural groups. As such, this appendix only includes the performance analysis and for any further information on the fatigue analysis method the reader is recommended to those members of the F93B group project who designed structural components.

Flight Pattern Distribution

2. Fig D-1 from Ref (1) details the flight pattern distribution which would produce the fatigue cycle for the F93B.

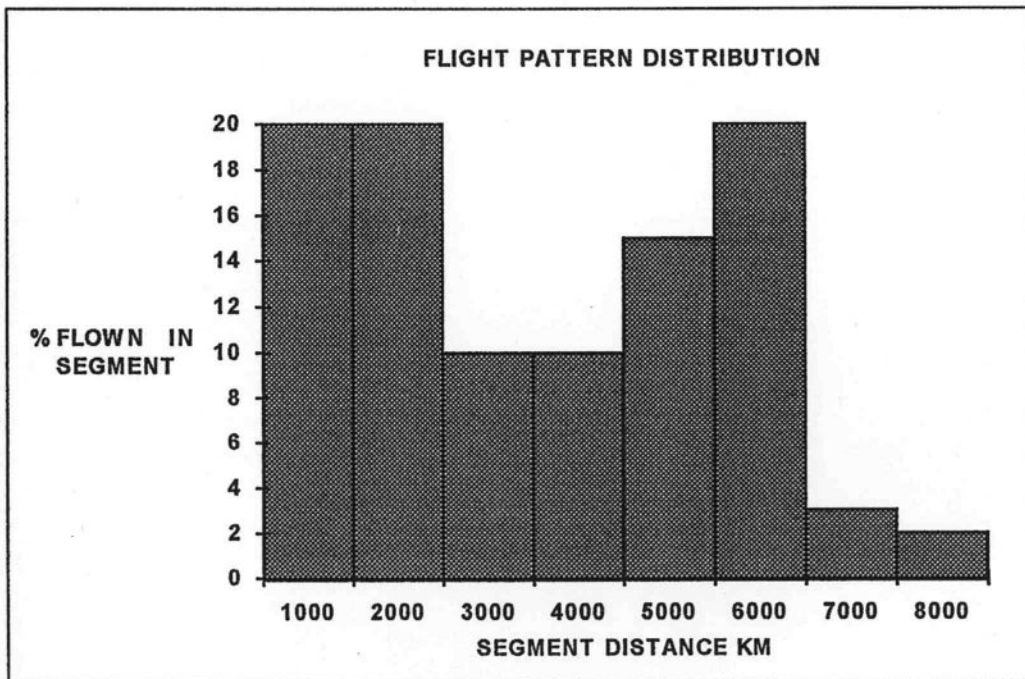


Figure D-1

The information which is required for the fatigue analysis programme must be split into several intervals covering the climb, cruise and descent phase of the aircraft flight profile. Following on from the intervals used in Appendix C to calculate the climb, cruise and descent statistics the same intervals are used to derive the necessary information for the fatigue cycle information. Therefore, there are 15 flight stages throughout each aircraft profile which consist of 7 climb stages, 1 cruise stage and 7 descent stages. For each stage the altitude (H_p) and mass (M) at the mid position of each interval is required plus the horizontal distance (D) covered along the ground. It is

assumed that all the aircraft profiles are flown at a 250 KEAS climb and descent speed and a cruising speed of Mach 0.7. The mass of the aircraft was calculated at the end of each climb, cruise and descent interval for the F93B in Appendix C. Therefore, knowing the mass of the aircraft at the beginning and end of each interval means that the average mass over the whole interval can be established. Similarly the average height is taken as the mid position over the whole interval. The ground distance covered during each interval is calculated using exactly the same method as specified in Appendix C.

The 8 flight profiles are detailed in Tables D-1 through to D-8.

5	15.31778	8572	0.54825	18112.72	128.5	181.8574	0.33
6	17.37572	10000	0.41204	61710.32	122	210.0533	0.07
7	22.84208	9572	0.45831	78128.06	128.5	210.0533	0.09
8	23.03573	8572	0.54825	18112.72	128.5	181.8574	0.02
9	23.2407	6838	0.85289	73089.83	128.5	178.0426	0.53
10	23.48702	5724	0.77052	73027.8	128.5	181.8574	0.5
11	23.7756	3710	0.94484	77975.4	128.5	149.6317	0.46
12	24.05134	2284	1.00595	77910.82	128.5	141.8304	0.43
13	24.25181	762	1.225	77841.97	128.5	128.5	0.38
TOTAL	190.479						

Table D-1

CALCULATIONS FOR FATIGUE CYCLE FOR 2000KM MISSION (PAYLOAD 1000kg)							
Take-off Mass		80370 kg					
Flight Sta	Alt (km)	Hgt (m)	ρ	Mass (kg)	Velocity	W (kg)	Mass (kg)
1	9.801038	732	1.225	80370.27	128.5	128.5	0.38
2	7.750522	2294	1.3275	85467.31	128.5	141.8304	0.43
3	5.457042	3810	0.95854	85467.31	128.5	178.0426	0.18
4	3.248063	5234	0.77052	85467.31	128.5	181.8574	0.5
5	16.5391	6838	0.85289	85467.31	128.5	178.0426	0.53
6	23.48702	8572	0.54825	85467.31	128.5	181.8574	0.02
7	21.84208	9572	0.45831	85467.31	128.5	210.0533	0.09
8	17.37572	10000	0.41204	61710.32	122	210	0.7
9	22.84208	9572	0.45831	78128.06	128.5	210.0533	0.06
10	23.03573	8572	0.54825	18112.72	128.5	181.8574	0.02
11	23.23800	6776	0.85289	73089.83	128.5	178.0426	0.53
12	23.48702	5724	0.77052	73027.8	128.5	181.8574	0.5
13	23.7756	3710	0.94484	77975.4	128.5	149.6317	0.46
14	24.05134	2284	1.00595	77910.82	128.5	141.8304	0.43
15	24.25181	762	1.225	77841.97	128.5	128.5	0.38
TOTAL	33.6524						

Table D-2

CALCULATIONS FOR FATIGUE CYCLE FOR 1000KM MISSION						PAYLOAD	20000KG
Take-off Mass		82250 kg					
Flight Sta	D (km)	Hp (m)	ρ	M(kg)	Ve(m/s)	Vt(m/s)	Mach No
1	6.421604	762	1.225	82194.97	128.5	128.5	0.38
2	7.323114	2284	1.00555	82071.24	128.5	141.8304	0.42
3	8.869251	3810	0.90464	81935.42	128.5	149.5317	0.46
4	11.73804	5334	0.77082	81795.28	128.5	161.9924	0.5
5	15.51778	6858	0.65269	81644.65	128.5	176.0426	0.56
6	21.83078	8382	0.54895	81476.77	128.5	191.9574	0.62
7	20.13423	9572	0.45831	81321.64	128.5	210.0833	0.69
8	743.5725	10000	0.41284	79699.85	122	210	0.7
9	22.84363	9572	0.45831	78135.67	128.5	210.0833	0.69
10	23.03733	8382	0.54895	78113.72	128.5	191.9574	0.62
11	23.2407	6858	0.65269	78079.94	128.5	176.0426	0.56
12	23.46953	5334	0.77082	78037.7	128.5	161.9924	0.5
13	23.77721	3810	0.90464	77985.5	128.5	149.5317	0.46
14	24.04296	2284	1.00555	77920.92	128.5	141.8304	0.42
15	24.25344	762	1.225	77841.07	128.5	128.5	0.38
TOTAL	1000.072						

Table D-1

CALCULATIONS FOR FATIGUE CYCLE FOR 2000KM MISSION						PAYLOAD	20000kg
Take-off Mass		86370 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	6.801988	762	1.225	86225.87	128.5	128.5	0.38
2	7.759922	2284	1.00555	86095.09	128.5	141.8304	0.42
3	9.407662	3810	0.90464	85951.36	128.5	149.5317	0.46
4	12.48063	5334	0.77082	85802.7	128.5	161.9924	0.5
5	16.5591	6858	0.65269	85642.36	128.5	176.0426	0.56
6	23.44565	8382	0.54895	85462.67	128.5	191.9574	0.62
7	21.94603	9572	0.45831	85295.08	128.5	210.0833	0.69
8	1737.572	10000	0.41284	81710.32	122	210	0.7
9	22.84203	9572	0.45831	78125.56	128.5	210.0833	0.69
10	23.03573	8382	0.54895	78103.61	128.5	191.9574	0.62
11	23.23909	6858	0.65269	78069.83	128.5	176.0426	0.56
12	23.46792	5334	0.77082	78027.6	128.5	161.9924	0.5
13	23.7756	3810	0.90464	77975.4	128.5	149.5317	0.46
14	24.04134	2284	1.00555	77910.82	128.5	141.8304	0.42
15	24.25181	762	1.225	77830.99	128.5	128.5	0.38
TOTAL	2000.626						

Table D-2

CALCULATIONS FOR FATIGUE CYCLE FOR 3000KM MISSION						PAYLOAD	20000 kg
Take-off Mass		90290kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	7.186814	762	1.225	90228.72	128.5	128.5	0.38
2	8.203126	2284	1.00555	90090.78	128.5	141.8304	0.42
3	9.956612	3810	0.90464	89938.99	128.5	149.5317	0.46
4	13.24432	5334	0.77082	89781.61	128.5	161.9924	0.5
5	17.64167	6858	0.65269	89611.22	128.5	176.0426	0.56
6	25.15399	8382	0.54895	89419.13	128.5	191.9574	0.62
7	23.93539	9572	0.45831	89238.1	128.5	210.0833	0.69
8	2730.567	10000	0.41284	83649.85	122	210	0.7
9	22.83186	9572	0.45831	78061.6	128.5	210.0833	0.69
10	23.02554	8382	0.54895	78039.66	128.5	191.9574	0.62
11	23.22889	6858	0.65269	78005.89	128.5	176.0426	0.56
12	23.4577	5334	0.77082	77963.68	128.5	161.9924	0.5
13	23.76535	3810	0.90464	77911.5	128.5	149.5317	0.46
14	24.03108	2284	1.00555	77846.95	128.5	141.8304	0.42
15	24.24153	762	1.225	77767.15	128.5	128.5	0.38
TOTAL	3000.47						

Figure D-3

CALCULATIONS FOR FATIGUE CYCLE FOR 4000KM MISSION						PAYLOAD	20000kg
Take-off Mass		94948 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	7.643969	762	1.225	94882.97	128.5	128.5	0.38
2	8.731364	2284	1.00555	94736.5	128.5	141.8304	0.42
3	10.61447	3810	0.90464	94575.07	128.5	149.5317	0.46
4	14.16854	5334	0.77082	94407.15	128.5	161.9924	0.5
5	18.96821	6858	0.65269	94224.52	128.5	176.0426	0.56
6	27.29018	8382	0.54895	94017.05	128.5	191.9574	0.62
7	26.53608	9572	0.45831	93818.86	128.5	210.0833	0.69
8	3721.613	10000	0.41284	86191.69	122	210	0.7
9	22.92367	9572	0.45831	78642.71	128.5	210.0833	0.69
10	23.1175	8382	0.54895	78620.68	128.5	191.9574	0.62
11	23.321	6858	0.65269	78586.78	128.5	176.0426	0.56
12	23.54998	5334	0.77082	78544.4	128.5	161.9924	0.5
13	23.85783	3810	0.90464	78492.02	128.5	149.5317	0.46
14	24.12374	2284	1.00555	78427.22	128.5	141.8304	0.42
15	24.33439	762	1.225	78347.11	128.5	128.5	0.38
TOTAL	4000.794						

Figure D-4

CALCULATIONS FOR FATIGUE CYCLE FOR 5000KM MISSION						PAYLOAD	20000kg
Take-off Mass		98000 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	7.949618	762	1.225	97780.27	128.5	128.5	0.38
2	9.085609	2284	1.00555	97780.27	128.5	141.8304	0.42
3	11.05788	3810	0.90464	97612.35	128.5	149.5317	0.46
4	14.7972	5334	0.77082	97437.29	128.5	161.9924	0.5
5	19.88113	6858	0.65269	97246.27	128.5	176.0426	0.56
6	28.78887	8382	0.54895	97028.1	128.5	191.9574	0.62
7	28.44114	9572	0.45831	96817.61	128.5	210.0833	0.69
8	4716.545	10000	0.41284	87265.69	122	210	7
9	22.78984	9572	0.45831	77798.13	128.5	210.0833	0.69
10	22.98345	8382	0.54895	77776.22	128.5	191.9574	0.62
11	23.18672	6858	0.65269	77742.52	128.5	176.0426	0.56
12	23.41544	5334	0.77082	77700.38	128.5	161.9924	0.5
13	23.723	3810	0.90464	77648.3	128.5	149.5317	0.46
14	23.98863	2284	1.00555	77583.87	128.5	141.8304	0.42
15	24.19899	762	1.225	77504.2	128.5	128.5	0.38
TOTAL	5000.833						

Figure D-5

CALCULATIONS FOR FATIGUE CYCLE FOR 6000KM MISSION						PAYLOAD	19772 kg
Take-off Mass		101500 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	8.30654	762	1.225	101429.5	128.5	128.5	0.38
2	9.500386	2284	1.00555	101270.6	128.5	141.8304	0.42
3	11.57939	3810	0.90464	101095.1	128.5	149.5317	0.46
4	15.54265	5334	0.77082	100911.6	128.5	161.9924	0.5
5	20.97514	6858	0.65269	100710.6	128.5	176.0426	0.56
6	30.61671	8382	0.54895	100479.4	128.5	191.9574	0.62
7	30.86115	9572	0.45831	100253.6	128.5	210.0833	0.69
8	5709.514	10000	0.41284	88784.57	122	210	0.7
9	22.7271	9572	0.45831	77407.66	128.5	210.0833	0.69
10	22.92059	8382	0.54895	77385.82	128.5	191.9574	0.62
11	23.12375	6858	0.65269	77352.2	128.5	176.0426	0.56
12	23.35234	5334	0.77082	77310.18	128.5	161.9924	0.5
13	23.65974	3810	0.90464	77258.24	128.5	149.5317	0.46
14	23.92522	2284	1.00555	77193.98	128.5	141.8304	0.42
15	24.13544	762	1.225	77114.52	128.5	128.5	0.38
TOTAL	6000.74						

Figure D-6

CALCULATIONS FOR FATIGUE CYCLE FOR 7000KM MISSION							PAYLOAD 16136 kg
Take-off Mass		101500 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	8.30654	762	1.225	101429.5	128.5	128.5	0.38
2	9.500386	2284	1.00555	101270.6	128.5	141.8304	0.42
3	11.57939	3810	0.90464	101095.1	128.5	149.5317	0.46
4	15.54265	5334	0.77082	100911.6	128.5	161.9924	0.5
5	20.97514	6858	0.65269	100710.6	128.5	176.0426	0.56
6	30.61671	8382	0.54895	100479.4	128.5	191.9574	0.62
7	30.86115	9572	0.45831	100253.6	128.5	210.0833	0.69
8	6713.195	10000	0.41284	86924.93	122	210	0.7
9	22.10163	9572	0.45831	73688.54	128.5	210.0833	0.69
10	22.29367	8382	0.54895	73667.27	128.5	191.9574	0.62
11	22.4953	6858	0.65269	73634.57	128.5	176.0426	0.56
12	22.72221	5334	0.77082	73593.69	128.5	161.9924	0.5
13	23.02754	3810	0.90464	73543.15	128.5	149.5317	0.46
14	23.29113	2284	1.00555	73480.6	128.5	141.8304	0.42
15	23.49956	762	1.225	73403.26	128.5	128.5	0.38
TOTAL	7000.008						

Figure D-7

CALCULATIONS FOR FATIGUE CYCLE FOR 8000KM MISSION							PAYLOAD 12548 kg
Take-off Mass		101500 kg					
Flight Sta	D (km)	Hp (m)	ρ	Mass (kg)	Ve(m/s)	Vt(m/s)	Mach No
1	8.30654	762	1.225	101429.5	128.5	128.5	0.38
2	9.500386	2284	1.00555	101270.6	128.5	141.8304	0.42
3	11.57939	3810	0.90464	101095.1	128.5	149.5317	0.46
4	15.54265	5334	0.77082	100911.6	128.5	161.9924	0.5
5	20.97514	6858	0.65269	100710.6	128.5	176.0426	0.56
6	30.61671	8382	0.54895	100479.4	128.5	191.9574	0.62
7	30.86115	9572	0.45831	100253.6	128.5	210.0833	0.69
8	7718.842	10000	0.41284	85064.33	122	210	0.7
9	21.42447	9572	0.45831	69967.53	128.5	210.0833	0.69
10	21.61436	8382	0.54895	69946.88	128.5	191.9574	0.62
11	21.81372	6858	0.65269	69915.18	128.5	176.0426	0.56
12	22.03812	5334	0.77082	69875.53	128.5	161.9924	0.5
13	22.34023	3810	0.90464	69826.51	128.5	149.5317	0.46
14	22.60095	2284	1.00555	69765.83	128.5	141.8304	0.42
15	22.80684	762	1.225	69690.77	128.5	128.5	0.38
TOTAL	8000.862						

Figure D-8

APPENDIX E

SPARV RESULTS

Introduction

1. SPARV is an aerodynamic panel program which uses an input model consisting of a multitude of quadrilateral panels to compute pressure distributions in both viscous and non-viscous modes. For the F93 group design project, most frequent use was made of the force/moment output file (SPARV.LD) obtained from integrating the pressure distribution. For a complete briefing on the programs capabilities refer to the user manual.

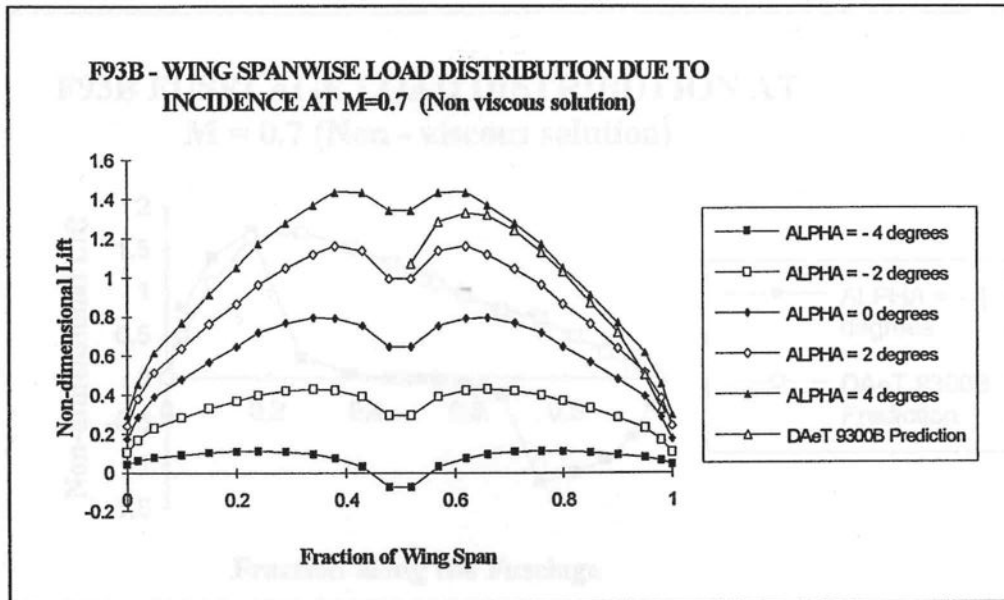
Spary Input Models

2. Wing Component An input model was created for the F93B wing component at different angles of attack in the inviscid mode at Mach 0.7. The results were used to determine aerodynamic loading information. The load distribution is non-dimensionalized by dividing the local load by the mean load $c*CL/\bar{c}*\bar{CL}$. Numerical data extracted from the SPARV.CPX file is listed in Table E-1 for the wing where ETA is the fraction of the wing span. Figure E-1 contains the information in graph format showing the different load distributions at the various angle of attacks. The SPARV data appears to follow the DAet 9300B prediction fairly closely although as these results only refer to the inviscid solution where the viscous effects of air are ignored it is likely that the airload distribution would show inaccuracies at the wing-fuselage junction where the interference effects cannot be accurately predicted.

3. Fuselage Component An input model was also created for the F93B fuselage component in the inviscid mode at Mach 0.7. The numerical data extracted from the SPARV.CPX file is listed in Table E-2 where ETA is the fraction of the total fuselage length. Figure E-2 contains the information in graph format. The input programme for the fuselage dimensions contains limited information which reduces the accuracy of the result. In this case a simple cigar shaped cylinder was tested and because the flow was inviscid the results were as expected in that there was no account taken of the tapering of the rear fuselage. Despite the above conditions the SPARV results were significantly different to the DAet 9300B predicted results which caused some consternation as the information was to be used to verify the aerodynamic loading data. After discussions with the DAet staff it was felt that the DAet predicted results were at fault.

WING SPANWISE LOAD DISTRIBUTION						
ETA	LOAD at AOA = -4°	LOAD at AOA = -2°	LOAD at AOA = 0°	LOAD at AOA = +2°	LOAD at AOA = +4°	DAet LOAD
0	0.0404	0.10494	0.17371	0.24114	0.29736	-
0.02	0.06099	0.16864	0.28637	0.38006	0.45735	-
0.05	0.07823	0.2293	0.39078	0.51455	0.61691	-
0.1	0.09225	0.28289	0.48114	0.6368	0.76684	-
0.15	0.10298	0.3304	0.56981	0.76307	0.91303	-
0.2	0.11026	0.36908	0.64876	0.86618	1.05248	-
0.24	0.11252	0.40022	0.71962	0.96331	1.17192	-
0.29	0.10844	0.42174	0.767	1.04852	1.28023	-
0.34	0.09647	0.43108	0.79612	1.11766	1.37078	-
0.38	0.07436	0.42438	0.79221	1.16239	1.43884	-
0.43	0.03313	0.39098	0.75482	1.13672	1.43752	-
0.48	-0.07445	0.29426	0.64719	0.99476	1.34537	-
0.52	-0.07445	0.29427	0.64719	0.99478	1.34539	1.0722
0.57	0.03312	0.39099	0.75484	1.13675	1.43755	1.2851
0.62	0.07437	0.42442	0.79229	1.16251	1.43897	1.3325
0.66	0.09647	0.43111	0.79618	1.11776	1.3709	1.319
0.71	0.010844	0.42177	0.76705	1.04589	1.2803	1.2432
0.76	0.11252	0.40023	0.71964	0.96333	1.17194	1.1312
0.8	0.11026	0.36907	0.64873	0.86613	1.0524	1.0301
0.85	0.10299	0.33041	0.5698	0.76304	0.91299	0.8711
0.9	0.09231	0.283	0.4813	0.63792	0.76718	0.72
0.95	0.07803	0.22943	0.39096	0.51478	0.61718	0.5
0.98	0.06103	0.16871	0.28645	0.38015	0.45744	0.32995
1	0.04043	0.10497	0.17375	0.24118	0.29739	-

Table E-1

**Figure E-1**

FUSELAGE LOAD DISTRIBUTION		
ETA	SPARV LOAD	DAet LOAD
0	0	0
0.031944	0.80934	0.5
0.090893	1.40718	1.1
0.167944	1.7061	1.63
0.268867	0.24252	1.7
0.369713	0.04935	1.53
0.4492	0.00732	1.34
0.4922	-0.00254	1.2
0.511226	-0.00367	1.17
0.541431	-0.01636	1.1
0.597252	-0.05739	0.98
0.670175	-0.18894	0.8
0.746175	-1.18863	0.7
0.816687	-1.09557	0.5
0.879893	-0.92778	0.3
0.931508	-0.66552	0.15
0.968021	-0.32007	0.1
0.985303	0	0

Table E-2

F93B FUSELAGE LOAD DISTRIBUTION AT M = 0.7 (Non - viscous solution)

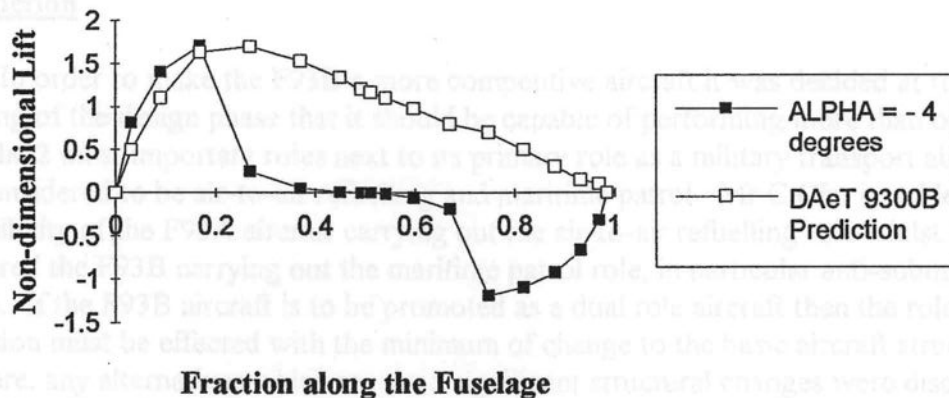


Figure E-2

INTERNAL EQUIPMENT

Electronic Surveillance Equipment

2. As there was insufficient time to design the tactical operator positions from scratch several existing maritime aircraft layouts were considered, such as the Mk2 and the PCO Orion, as well as the Future Long Range Maritime Patrol Aircraft (FLRMPA) proposed by SH Chicken (Ref (4)). There are typically 6 tactical operator positions situated as either a bank of 2 facing forward and 4 facing sideways, or all 6 facing sideways. For ease of access to the forward facilities in the F93B it was decided to place all the consoles in a row facing aft. The dimensions for the consoles are detailed in para (19).

Additional Avionics Equipment

3. For most aircraft designs it is difficult to distinguish between the location of the navigational avionics equipment and the maritime surveillance equipment. Therefore it is assumed that the dimensions and weight of the rear avionics bay specified in the FLRMPA should cater for the additional electronic equipment required in the F93B. The dimensions and weights are detailed in para (19). Traditionally extra avionics are situated at the rear of an aircraft where any heat and noise causes minimal effect to the

APPENDIX F

MARITIME PATROL AIRCRAFT CONFIGURATION CHANGES

Introduction

1. In order to make the F93B a more competitive aircraft it was decided at the beginning of the design phase that it should be capable of performing more than one role. The 2 most important roles next to its primary role as a military transport aircraft were considered to be air-to-air refuelling and maritime patrol. Mr C Chu considered the feasibility of the F93A aircraft carrying out the air-to-air refuelling role whilst I considered the F93B carrying out the maritime patrol role, in particular anti-submarine warfare. If the F93B aircraft is to be promoted as a dual role aircraft then the role conversion must be effected with the minimum of change to the basic aircraft structure. Therefore, any alternatives which required significant structural changes were discarded at an early stage of the role re-design. This appendix deals with the configuration changes, both internal and external, required to convert the basic transport aircraft into a maritime patrol aircraft. This included the installation of electronic surveillance equipment, anti-submarine weapons, ESM pods, radar, sonobouys, chaff and flare, magnetic anomaly detector, observation windows and all the necessary accommodation for the extra personnel. The role configuration drawings are at the end of the appendix. Drawing No F93B P-01 shows the internal role conversion required whilst F93B P-02 shows the location of the external stores required to carry out maritime surveillance and anti-submarine warfare.

INTERNAL EQUIPMENT

Electronic Surveillance Equipment

2. As there was insufficient time to design the tactical operator positions from scratch several existing maritime aircraft layouts were considered, such as the Nimrod Mk2 and the PC3 Orion, as well as the Future Long Range Maritime Patrol Aircraft (FLRMPA) proposed by SH Chicken (Ref (4)). There are typically 6 tactical operator positions situated as either a bank of 2 facing forward and 4 facing sideways or all 6 facing sideways. For ease of access to the forward facilities in the F93B it was decided to place all the consoles in a row facing stbd. The dimensions for the consoles are detailed in para (19).

Additional Avionics Equipment

3. For most aircraft designs it is difficult to distinguish between the location of the navigational avionics equipment and the maritime surveillance equipment. Therefore it is assumed that the dimensions and weight of the rear avionics bay specified in the FLRMPA should cater for the additional electronic equipment required in the F93B. The dimensions and weights are detailed in para (19). Traditionally extra avionics are situated at the rear of an aircraft where any heat and noise causes minimal effect to the

crew; however, in this case the F93B has a ramp door which must not be obstructed. Therefore, it was decided to situate the additional avionics bay on the stbd side of the aircraft close to the consoles and the ECS pack, which is located in the stbd undercarriage blister. This reduces the amount of cabling required and any consequent loss in efficiency and improves access for maintenance. The dimensions for the avionics bay are detailed in para (19)

Sonobuoy Equipment

4. Due to the very short undercarriage on the F93B it is not practical to locate any sonobuoys externally underneath the fuselage. Whilst releasing the sonobuoys in flight would not present any problems there is insufficient room under the fuselage to access the loading bay whilst the aircraft is on the ground. Therefore all the sonobuoys must be carried internally plus the equipment needed to eject them. The RAF Nimrod maritime aircraft typically carries 180 Barra sonobuoys which must be housed on racks, preferably next to the ejection equipment. Therefore, the ejection equipment and racks are situated along the port side of the aircraft opposite the consoles. All dimensions are detailed in para (19) and the internal aircraft arrangement is shown in Drawing No F93B P-01.

Observation Windows

5. Observation windows are a desirable feature which can be incorporated into the F93B Maritime Patrol Aircraft (MPA) as a role fit providing there are suitable windows in the right location specified in the basic F93B transport aircraft. In order to view the ejection of the sonobuoys and life rafts (see para 7) 2 observation windows would be required, one on the port side forward in the cabin between the sonobuoy racks and the ejection equipment and the other on the stbd side at the rear of the cabin close to the ramp. Drawing F93B P-01 shows the exact location of the observation windows.

Crew Rest Areas

6. The extended operating duration of maritime patrol aircraft means that there is a requirement for additional crew facilities. The area below the upstairs galley and bunks is capable of holding an additional galley and toilet despite the restriction of the nose undercarriage bay. There is also room for a locker area on the other side of the nose undercarriage bay opposite the forward passenger door. The additional galley, toilet and locker are a permanent fit and, therefore, there is no requirement for any extra cooking or toilet facilities when the basic F93B is re-roled to the maritime patrol fit. Apart from those facilities already mentioned there is a requirement for enough sleeping and rest accommodation for 2 complete crews. There are already 2 bunks on the F93B aircraft situated upstairs behind the cockpit next to the galley area and there is room for 6 additional bunks along the port side of the freight bay opposite the avionics bay, which means there would be enough bunks on the aircraft for a complete crew of 8 personnel. Apart from the sleeping areas there is a need for a rest area which comprises 4 seats and a table and is located on the stbd side of the aircraft next to the avionics bay. Drawing F93B P-01 details the accommodation facilities on the F93BMPA.

Life Rafts

7. Part of the remit of a maritime surveillance patrol aircraft is to be able to drop life rafts which would normally be dropped from a bomb bay or from the a passenger door at low level. The F93B aircraft does not have a bomb bay or a rear passenger door so the life rafts will have to be dropped by lowering the rear ramp and pushing them out the back, similar to dropping resupplies. Therefore, 2 x 26 man life rafts are situated on the rear ramp ready for ejection.

Internal Fuel

8. Any additional fuel that can be carried is a great asset to a maritime patrol aircraft therefore the remaining area at the rear of the freight bay can be useful utilised for an internal fuel tank. Most internal fuel tanks used by the RAF as role fits, to achieve an extended ferry range, are circular in nature and rest on a rectangular frame which helps to spread the floor loading. The additional weight of fuel that can be carried is calculated as follows:

The maximum tank diameter = 3m and the maximum tank length = 3m also.
Therefore:

$$\text{Tank volume} = \pi r^2 \times l = \pi \times (1.5)^2 \times (3) = 21.206 \text{ m}^3 \times 1000 = 21206 \text{ litres}$$

$$\text{The Fuel Density for A1} = 6.74 \text{ lbs/gallon} = (6.74 \times 0.22) / 2.2 = 0.674 \text{ kgs/litre}$$

$$\text{Therefore weight of additional fuel} = 21206 \times 0.674 = 14\,293 \text{ kg}$$

Additional Fuel in Internal Fuel Tank = 14293kg

This is a significant amount of fuel, a 36% increase in fuel load, and detailed stressing calculations would be required to ensure that the freight bay floor was capable of withstanding the maximum fuel load in the internal tank.

Radar

9. A surveillance radar is required for maritime operations and is typically either a single unit placed below the aircraft or in a drooped nose, or 2 radars placed at the front and rear of the aircraft to give 360° coverage. Because of the rear ramp on the F93B there is no suitable location for a radar to be placed at the rear of the aircraft. Therefore a single radar placed in the nose of the aircraft is the preferred solution. The area available inside the nose cone for a radar is from Stn 0 to Stn 1 and measures approximately 1m x 2.1m x 2.5m. There is enough room in the nose cone area for most radars currently in production and having looked at the merits of several different radars mentioned in Ref (5) the APS - 134 (LW) radar was selected primarily because of its light weight. It is a parabolic radar measuring 1.14m x 0.4m and is sited as close to the bottom skin of the nose cone as possible without restricting its radius of scan. There is a possibility that the field of view of the radar may be limited by the underside of the aircraft when scanning to the rear, in which case the nose cone of the aircraft may have

to be lowered in order to provide the radar with an unimpeded view. Only flying trials of the F93BMPA will be able to assess the radar coverage available.

Additional Equipment

10. There are several items which can be installed to satisfy the maritime commitment to a greater or lesser degree. Which equipment a buyer wishes to install, if any, is very much a personal decision based on what type of maritime commitment is required. The location of this additional equipment is discussed below:

11. **Search Light** A searchlight is required for maritime patrol observation and identification of targets such as survivors in the water at night. The ideal position for a searchlight is close to the centreline of the aircraft preferable under the forward fuselage. In this case it was decided to place a retractable searchlight behind the radar between Stn 1 and 2 on the stbd side of the centreline to allow for other equipment to be located on the port side of the aircraft (See para 12). By making the searchlight retractable it will not interfere with the radar for the majority of the time but is readily available in a useful position when needed. A searchlight which is carried internally for the majority of the time also removes the drag penalty associated with an external searchlight. Drawing No F93B P-01 shows the location of the searchlight.

12. **Low Light Television (LLTV)** The LLTV is also a very useful aid to maritime surveillance by being able to record and thus enabling subsequent identification of targets in poor visibility conditions. The LLTV is also made retractable to avoid any drag penalty and is situated next to the searchlight on the port side of the centreline. Drawing No F93B P-01 shows the location of the LLTV.

13. **Forward Looking Infra Red (FLIR)** The ability to see well at night particularly when flying at low level is satisfied by an infra red system. A forward looking infra red system is more practicable in this case because of the lack of accessible space below the aircraft. However a FLIR placed above the windscreen in the cockpit roof is not acceptable because of the possibility of installing an air-to-air refuelling probe in that area. Therefore the most suitable place for a FLIR system is in the space above the radar between Stn 0 and 1. This will require a slight reshape of the aircraft skin around the FLIR lens to allow an unrestricted view and aircraft trials would need to be carried out to establish the best configuration.

EXTERNAL EQUIPMENT

14. The carriage of external equipment on the aircraft is very important in determining the drag characteristics of the aircraft. A large increase in drag will significantly reduce the range and thus the effectiveness of the aircraft. Therefore every consideration must be given to limiting the increase in drag created by external stores.

Electronic Equipment

15. Magnetic Anomaly Detectors (MAD) An important part of the anti-submarine role is being able to detect the submarines. One way this can be achieved is by using a magnetic anomaly detector which detects a change in the earth's magnetic field due to the presence of a large metallic object. Because this equipment is sensitive to large metallic objects it must be placed away from the aircraft and is usually situated on a boom behind the aircraft. A typical MAD such as the AN/ASQ - 504(V) is 0.813m long and 0.178m in diameter. As there is enough room at the rear of F93B for a MAD to be attached the end of the tailboom is the preferred location as shown on F93B P-01.

16. Electronic Support Measures (ESM) In addition to the MAD, electronic detectors required to carry out maritime surveillance are housed in ESM pods which are usually located one on each wing tip and one on the fin. Although the F93B has a twin fin arrangement it is only necessary to place an ESM pod on one of the fins. The exact location of the pods is shown on Drawing No F93B P-01.

Weapons

17. Offensive Measures In order to carry out effective anti-submarine and anti-ship warfare an aircraft must have a credible means of attacking its enemy once located. The preferred weapons currently used by the RAF on their Nimrod aircraft are the Stingray torpedo and the AGM-84 Harpoon air-to-surface missile. If the F93BMPA is to be sold to other European nations it should also be capable of carrying other popular weapons such as the AM 39 Exocet air-to-surface missile. The dimensions of the weapons mentioned above are detailed in para (19). Initially, the creation of a bomb bay in the underside of the aircraft was considered as the most drag efficient method of transporting a significant number of weapons. However this would require the raising of the freight bay floor by approximately 1.5m which, although this would still leave enough room for all the internal equipment, was considered beyond the scope of a role fit and there was still the problem of how to access the weapons bay whilst on the ground. Additionally, the cut-outs required for the bomb bay doors would probably require increased aircraft structure around that area. Therefore there is no option but to carry the weapons externally thus incurring the penalty of increased drag and reducing the number of weapons that can be carried. The method of carrying external stores is on underwing pylons situated on hard points between the engines similar to the position of the external fuel tanks carried by the RAF C-130 aircraft. In this position the F93BMPA can carry a Stingray torpedo, an Exocet missile or a Harpoon missile on each wing. The drag produced by the underwing pylons depends on the height of the pylon and the interference of the weapon with the wing. Although there is an ideal pylon height for each individual weapon, a fixed height which is the best fit is used and due consideration must also be given to whether the flaps can still be fully extended when weapons are fitted. The location of the weapons is shown in Drawing No F93B P-02 and the drag penalty is considered in Appendix G.

18. Defensive Measures In order to fulfil its primary role a maritime patrol aircraft usually carries some form of self defence, be it active or passive systems:

a. Active Defence Typically active defence takes the form of air-to-air missiles which for RAF maritime aircraft is usually AIM-9 short range IR missiles. The only possible locations for these missiles is attached to the side of the fuselage below the wing root or on underwing pylons between the outer engine and the wing tip. In this instance the latter solution is chosen because the location of the hard points for the AIM-9 missiles could also be utilised for other weapons, such as additional torpedoes, unlike the fuselage mounting which could not be used for any form of free fall weapon due to the extension of the undercarriage blister below the attachment point. Table F-1 details the dimensions of the AIM-9 missiles and Drawing No F93B P-02 show their location.

b. Passive Defence One passive defence measure involves the ejection of flare cartridges to try and distract incoming infra red seeking missiles whilst the aircraft manoeuvres out of its line of fire. Because they are trying to distract attention from the main heat sources on the aircraft the flares need to be ejected close to the heat sources so that the aircraft can manoeuvre away from the blanket of flares without the missiles detecting the change in direction. Therefore as the largest heat source on the F93B are the engines then the flares are situated on the rear of the fuselage behind the engines facing outwards and rearwards at an angle of 45° . A further passive defence measure designed to distract radar seeking missiles involves the ejection of chaff cartridges which are usually located with the flare cartridges as they use the same ejection system. The system location is shown in Drawing No F93B P-01.

MARITIME PATROL EQUIPMENT DIMENSIONS

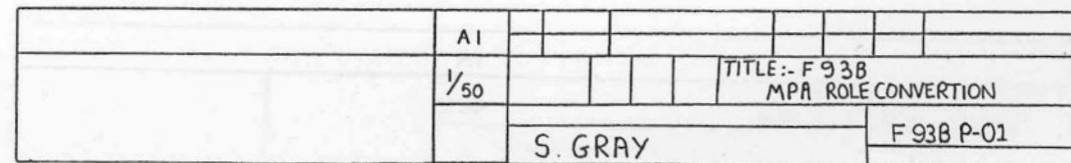
19. The dimensions for all the additional equipment required for the maritime patrol aircraft are shown in Table F-1 and the drawings showing the location of the equipment are at the end of the appendix. The scale of the drawings is 50:1.

EQUIPMENT DIMENSIONS				
EQUIPMENT (QTY)	LENGTH (m)	WIDTH (m)	HEIGHT (m)	WEIGHT (kg)
INTERNAL				
Operators Consoles	6.04	1.6	2.1	400
Electronic Equipment	2.9	2.6	-	1140
Sonobuoy Racks	3.3	1.6	1.5	2286
Sonobuoy Chute	1.3	0.8	1	50
Rest Area	3.4	1.8	-	-
Sleeping Bunks	6	1	2	-
Radar	-	1.14	0.4	237
Searchlight	0.7	0.5	-	10
LLTV	0.25	0.25	0.5	6
FLIR	0.5	0.5	0.25	10
SAR Rafts (2)	1	0.5	0.8	90
Extra Internal Fuel	3	1.5 radius	-	14293
Additional Crew (6)	-	-	-	540
Observation Windows (2)	0.6	0.3 dia	-	-
EXTERNAL				
MAD	0.84	0.18 dia	-	16
ESM pods (3)	2	0.46 dia	-	60
AIM-9 missiles (2)	2.87	0.127 dia	-	174
Exocet missiles (2)	4.7	0.35 dia	-	670
Harpoon missiles (2)	3.9	0.343 dia	-	1060
Stingray Torpedoes (2)	2.6	0.324 dia	-	530

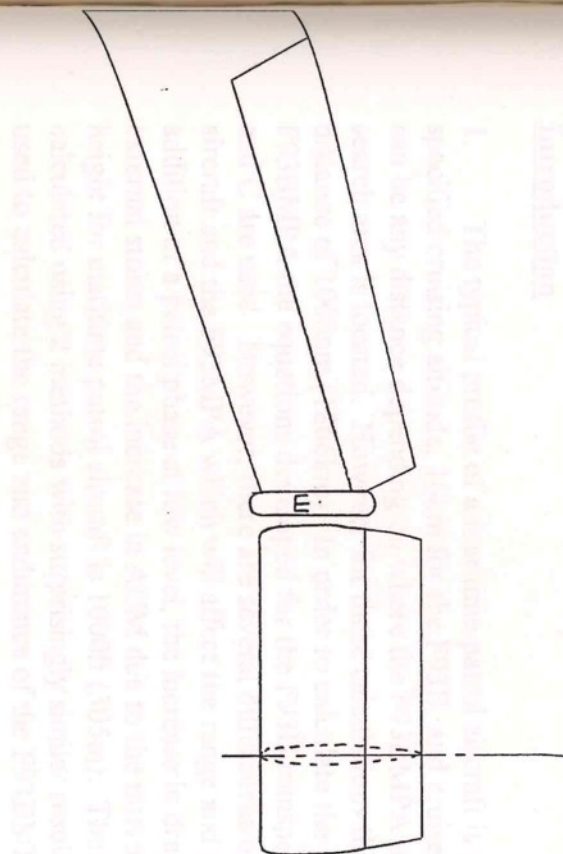
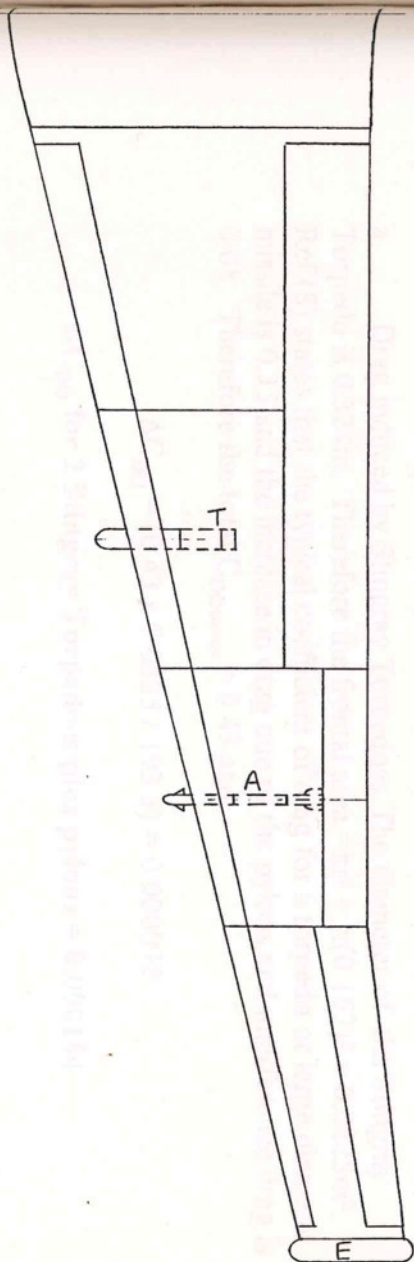
Table F-1

A hand-drawn floor plan of a boat interior, oriented horizontally with the bow on the left and the stern on the right. The plan is divided into several functional areas:

- Forward Section (Bow):** Includes a small cabin area with a toilet and a storage compartment labeled 'S'. A 'GALLEY' is located near the bow.
- Mid-Section:** Features a '6 TACTICAL OPERATORS' station, 'AVIONICS EQUIP', and a 'REST AREA'.
- Stowage and Access:** A 'CLOSET' and a 'CHUTE' are located near the bow.
- Lower Deck/Storage:** Below the main deck, there are 'SONOBUOYS' and three sets of '2 BUNKS'.
- Engine and Fuel:** A 'FUEL TANK' is located near the stern.
- Life Rafts:** A large area labeled 'LIFE RAFTS' is situated towards the stern, enclosed by dashed lines.
- Compass:** A compass rose is located at the very bow.
- Labels:** Various labels are present, including 'C' at the stern, 'M' at the bow, and 'S' in the forward cabin.



T	TORPEDO
A	AIM-9 MISSILE
E	ESM POD



DRAG ESTIMATION

The first method of calculating the drag increase was to add up the calculated drag contributions provided by the stores using that (5). While the second method was to use the DPLTADT drag reduction program in the Design of Armament.

Drag Increase using Manual Calculation

The increase in drag (delta C_D) of the F-105B is calculated manually by adding the coefficient of drag for the individual stores (C_D). Using a table of the frontal area of the stores and the air wing area (115.3 m²) the potential increase in the zero lift drag for the whole aircraft is calculated. By adding together all the calculated C_D values, a coefficient of drag value for C_D is known and the new drag point for the F-105B is calculated.

	AI	F93B MPA EXTERNAL STORES	
	Y30		
		S. GRAY	F93B P-02

APPENDIX G

MARITIME PATROL AIRCRAFT DRAG ESTIMATION

Introduction

1. The typical profile of a maritime patrol aircraft is to take-off and climb to a specified cruising altitude, 10km for the F93B, and cruise out to the patrol area. This can be any distance depending on where the F93BMPA is based and where the target search area is located. However for these calculations the patrol area is taken to be at a distance of 1000nm (1866km). In order to calculate the patrol endurance of the F93BMPA the equations developed for the F93B transport aircraft in Appendix A, B and C are used. However, there are several differences between the F93B transport aircraft and the F93MPA which will affect the range and endurance calculations: the addition of a patrol phase at low level, the increase in drag due to the carriage of external stores and the increase in AUM due to the internal fuel tank. The usually patrol height for maritime patrol aircraft is 1000ft (305m). The increase in external drag is calculated using 2 methods with surprisingly similar results and the revised drag polar is used to calculate the range and endurance of the F93BMPA.

DRAG ESTIMATION

2. The first method of calculating the drag increase was to add up all the individual drag contributions produced by the stores using Ref (8). Whilst the second method was to use the DELTADD drag estimate programme at the College of Aeronautics.

Drag Increase using Manual Calculations

3. The increase in zero lift drag (ΔC_{DO}) of the F93BMPA is calculated manually by establishing the coefficient of drag for the individual stores ($C_{DOstores}$). Using the ratio of the frontal area of the store over the total wing area (193.3 m²) the overall increase in the zero lift drag for the whole aircraft is established. By adding together all the individual $C_{DOstores}$ contributions a new value for C_{DO} is known and the new drag polar for the F93BMPA is established.

- a. Drag induced by Stingray Torpedoes The diameter of the Stingray Torpedo is 0.324m. Therefore the frontal area $= \pi r^2 = \pi(0.162)^2 = 0.0825\text{m}^2$. Ref (8) states that the typical coefficient of drag for a torpedo or large diameter missile is 0.35 and the increase in drag due to the pylons and interference drag is 0.08. Therefore the total $C_{DOstores} = 0.43$ and:

$$\Delta C_{DO} = (0.43 \times 0.0825 / 193.3) = 0.000092$$

$$\Delta C_{DO} \text{ for 2 Stingray Torpedoes plus pylons} = 0.000184$$

- b. Drag induced by Harpoon Missiles The diameter of a Harpoon missile is 0.343m therefore the frontal area = $\pi(0.1715)^2 = 0.0924 \text{ m}^2$ and the total $C_{DO\text{stores}}$ figure for the missile and pylon from Ref (8) is the same as for the Stingray Torpedo = 0.43. Therefore:

$$\Delta C_{DO} = 0.43 \times 0.0924 / 193.3 = 0.0002055$$

$$\Delta C_{DO} \text{ for 2 Harpoon missiles plus pylons} = 0.000411$$

- c. Drag induced by Exocet Missiles The diameter of an Exocet missile is 0.35m therefore the frontal area = $\pi(0.175)^2 = 0.0962 \text{ m}^2$ and the total $C_{DO\text{stores}} = 0.43$ also. Therefore:

$$\Delta C_{DO} = 0.43 \times 0.0962 / 193.3 = 0.000214$$

$$\Delta C_{DO} \text{ for 2 Exocet missiles plus pylons} = 0.000428$$

- d. Drag induced by AIM-9 Missiles The drag induced by 2 AIM-9 missiles was calculated using Fig 14 in Ref (9) which shows typical store drag increments. At Mach Nos between 0.4 and 0.8 the coefficient of zero lift drag ($C_{DO\text{stores}}$) for 2 AIM-9 missiles plus pylons = 1.1 based on a reference wing area of 354 ft². The F93B has a wing area of 2079.5 ft² therefore:

$$\Delta C_{DO} \text{ for an AIM-9 missile plus pylon} = (1.1/2079.5) / 2 = 0.000264$$

$$\Delta C_{DO} \text{ for 2 AIM-9 missiles plus pylons} = 0.000529$$

- e. Drag induced by Wing Tip and Fin ESM Pods The diameter of the ESM pods = 0.46m therefore the frontal area = $\pi(0.23)^2 = 0.166 \text{ m}^2$ and $C_{DO\text{stores}} = 0.055$ from Ref (8). Therefore:

$$\Delta C_{DO} = 0.055 \times 0.166 / 193.3 = 0.0000472$$

$$\Delta C_{DO} \text{ for 2 Wing Tip Pods plus 1 Fin pod} = 0.0001416$$

- f. Drag induced by Observation Windows The radius of curvature of the observation windows = 0.15m therefore the because the window is a hemisphere the frontal area = $\pi(0.15)^2 / 2 = 0.035 \text{ m}^2$. $C_{DO\text{store}} = 0.42$ for the window from Ref(4).

$$\Delta C_{DO} = 0.42 \times 0.035 / 193.3 = 0.000076$$

$$\Delta C_{DO} \text{ for 2 Observation Windows} = 0.000152$$

Therefore in the worst case when 2 Harpoon missiles are fitted with 2 wing tip pods and one fin pod plus 2 AIM-9 missiles and 2 observation windows:

$$\text{Total } \Delta C_{D0} = 0.000411 + 0.001416 + 0.000529 + 0.000152 = 0.00123$$

Therefore the zero lift component in the F93B drag polar is increased by 0.00123 due to the carriage of external stores whilst the coefficient of lift is assumed to be unaffected. The drag polar for the F93B transport aircraft = $C_D = 0.021 + 0.039 C_L^2$ from Ref (1) when cruising at an altitude of 10km and Mach 0.7. Therefore the drag polar for the F93BMPA under the same conditions is:

$$C_D = 0.0222 + 0.39 C_L^2$$

4. Disadvantages of using Manual Calculations There is insufficient information available in Ref (4).(8) or (9) to establish the change in the zero lift component when the aircraft is being operated at its patrol height of 1000ft and at slow speeds. Therefore, in order to establish a drag polar for these conditions the DELTADD programme was utilised.

Drag Increase using DELTADD Programme

5. Input Data The College of Aeronautics operates a drag estimation programme called DELTADD which provides basic data such as the overall drag polar, C_D and C_L values for each specific case as well as an indication of the C_L value above which buffet may occur. The programme input data consists of aircraft dimensions and specific case requirements. Table G-1 details the input dimensions required. In particular the F93BMPA requires extra data to be input into the programme specifying the external stores that are carried in the worst case scenario. This additional data is input under the section called additional zero lift drag coefficient and is accessed by placing a -1 in the column. This accesses a further section which specifies extra drag data and the external stores are added to the programme at this stage. Bearing in mind that the programme asks for the qty, frontal area and the drag coefficient of each type of external store the information generated in para 3 is not wasted. Indeed the DELTADD programme allows further manipulation of the manual calculations to produce more specific information.

DELTADD DRAG ESTIMATE INPUT DATA	
GENERAL	
Input data units (0= imperial 1 = metric)	1
Output type (1 = report 2 = detailed)	1
Temp diff from ISA Conditions(c)	0
Additional zero lift drag coefficient	-1
WING	
Aerofoil type (1 = conventional 2 = advanced)	2
Span (m of ft)	46.1

Area (m ² or ft ²)	193.3
Wetted area (m ² or ft ²)	390
Mean aerodynamic chord (m or ft)	4.726
Aspect ratio	11
C/4 sweep	12
LE sweep	13.75
Taper ratio	0.284
t/c ratio	0.13
Camber (@ 70% semi-span)	0.01
FUSELAGE	
Length (m or ft)	39
Height (m or ft)	5.5
Width (m or ft)	5.8
Base area (m ² or ft ²)	0
Max xsect area excluding inlets (m ² or ft ²)	30
Wetted area (m ² or ft ²)	570
TAILPLANE	
MAC (m or ft)	3
Wetted area (m ² or ft ²)	100
t/c ratio	0.1
FIN	
MAC (m or ft)	5.78
Wetted area (m ² or ft ²)	68.6
t/c ratio	0.1
NASCELLES AND PYLONS	
Number	4
Length (m or ft)	5.32
Wetted area - single nacelle (m ² or ft ²)	30
Wetted area - single pylon (m ² or ft ²)	18
EXTRA DRAG DATA	
Flap type (0=split 1=slotted or fowler)	1
Flap chord/wing chord ratio - 1	0.3
Flap chord/wing chord ratio - 2	0.12
Flap chord/wing chord ratio - 3	0
Flap outer end semi-span ratio	0.75
Flap inner end semi-span ratio	0.121
Number of piston engines	0
Max design AUM	101500
Max design diving speed (Mach No)	0.77
Max lift coefficient	1.4
Miscellaneous additional coefficient	0.002
STORES CLASS 1 - Torpedo/Missiles	
No of stores	2
Frontal area (each)	0.092
Drag coefficient	0.43
STORES CLASS 2 - AIM-9 Missiles	
No of stores	2

Frontal area (each)	0.013
Drag coefficient	0.055
STORES CLASS 3 - ESM Pods	
No of stores	3
Frontal area (each)	0.166
Drag coefficient	0.055
STORES CLASS 4 - Observation Windows	
No of stores	2
Frontal area (each)	0.35
Drag coefficient	0.42

Table G-1

6. Output Data A typical patrol speed for a maritime aircraft is 200 knots; however, the lift coefficient required to maintain 200 knots at low level may not be achievable or desirable. Therefore case data was requested at different speeds in order to establish the preferred patrol speed. Also a reference case was specified at 10km altitude and Mach 0.7 to cross refer with the manual calculations. Table G-2 shows the case data that was requested and Table G-3 highlights the relevant information that DELTADD produced for each case.

CASE DATA			
CASE NO	MACH NO	ALTITUDE (m)	AIRCRAFT MASS (kg)
1	0.7	10000	100085.8
2	0.3	305	93043.3
3	0.35	305	93043.3
4	0.4	305	93043.3

Table G-2

DELTADD DRAG ESTIMATE RESULTS				
CASE NO	DRAG POLAR	CASE C_L	CASE C_D	BUFFET ONSET C_L
1	$C_D = 0.0223 + 0.039 C_L^2$	0.5592	0.0329	0.857
2	$C_D = 0.0233 + 0.039 C_L^2$	0.7658	0.0451	1.415
3	$C_D = 0.0229 + 0.039 C_L^2$	0.5626	0.035	1.345
4	$C_D = 0.0225 + 0.039 C_L^2$	0.4307	0.0285	1.287

Table G-3

7. Analysis of Results Case 1 confirmed that the drag polar produced by manual calculations ($C_D = 0.0222 + 0.039 C_L^2$) under the same conditions was very close to the

drag polar produced by DELTADD ($C_D = 0.0223 + 0.039C_L^2$). Therefore the drag polars produced by DELTADD for the low speed patrols at 1000ft were assumed to be credible. At Mach 0.3 (200 knots) the coefficient of lift = 0.77 and is considered to be too high to maintain comfortably especially when the surveillance equipment requires a steady flat platform. At Mach 0.35 (230 knots) the coefficient of lift = 0.56 which is a more realistic value to maintain; whilst at Mach 0.4 (264 knots) the coefficient of lift = 0.43, but it is doubtful whether the RB509-18 engine can achieve this speed at low level. Therefore a patrol speed of Mach 0.35 (230 knots) is considered to be a satisfactory balance between a low patrol speed and a low C_L value.

$$\text{Drag Polar during Cruise at 10km Altitude} = C_D = 0.0223 + 0.039C_L^2$$

$$\text{Drag Polar during Patrol with External Stores} = C_D = 0.0229 + 0.039C_L^2$$

Appendix F para 3 as 14293kg. This means the AIM of the aircraft is raised to a maximum value of 113162kg when carrying the maximum weight of external stores and internal fuel.

b. The revised drag polars for the F93MPA calculated in Appendix G are used throughout the calculations to take account of its external stores.

$$\text{Drag Polar at cruising altitude of 10km} = C_D = 0.0223 + 0.039C_L^2$$

$$\text{Drag Polar at patrol altitude of 300m} = C_D = 0.0229 + 0.039C_L^2$$

c. An assessment of the most fuel efficient method of carrying out the patrol phase is required given the limited amount of engine data available particularly at low altitudes.

Table H-1 shows some typical weapon and fuel configurations with the corresponding range and endurance parameters.

TAKE-OFF PERFORMANCE

2. Using the sub-continua as stated in Appendix A the take-off distance for the increased aircraft mass can be calculated. AIM = 113162kg then:

$$\text{Ground run distance} = 1084m$$

$$\text{Distance from rotate to lift-off} = 193m$$

$$\text{Distance from lift-off to screen height} = 350m$$

$$\text{Maximal Take-off Distance required for F93MPA} = 1627m$$

This take-off distance is outside the original design specification for the F93 transport aircraft which means that the F93MPA could not take-off at its maximum AIM when

APPENDIX H

MARITIME PATROL AIRCRAFT PERFORMANCE EVALUATION

Introduction

1. To evaluate the performance of the F93BMPA the same equations generated in Appendices A, B and C are utilised with a few variations:

a. The 3 g manoeuvre requirement placed on the F93B transport aircraft is relaxed on the F93BMPA because it is unlikely that a maritime patrol aircraft would ever be required to carry out any violent manoeuvres. Therefore the AUM of the aircraft can be increased to allow for the carriage of additional fuel in an internal fuel tank. The maximum amount of additional fuel was calculated in Appendix F para 8 as 14293kg. This means the AUM of the aircraft is raised to a maximum value of 113162kg when carrying the maximum weight of external stores and internal fuel.

b. The revised drag polars for the F93MPA calculated in Appendix G are used throughout the calculations to take account of the external stores:

$$\text{Drag Polar at cruising altitude of 10km} = C_D = 0.0223 + 0.039C_L^2$$

$$\text{Drag Polar at patrol altitude of 305m} = C_D = 0.0229 + 0.039C_L^2$$

c. An assessment of the most fuel efficient method of carrying out the patrol phase is required given the limited amount of engine data available particularly at low altitudes.

Table H-1 shows some typical weapons and fuel configurations with the corresponding range and endurance parameters.

TAKE-OFF PERFORMANCE

2. Using the sub-routines generated in Appendix A the take-off distance for the increased aircraft mass can be calculated. AUM = 113162kg then:

$$\text{Ground run distance} = 1084\text{m}$$

$$\text{Distance from rotate to lift-off} = 198\text{m}$$

$$\text{Distance from lift-off to screen height} = 350\text{m}$$

$$\text{Maximum Take-off Distance required for F93BMPA} = 1632\text{m}$$

This take-off distance is outside the original design specification for the F93B transport aircraft which means that the F93BMPA could not take-off at its maximum AUM when

operating from a 1500m runway. Figure H-1 shows that the maximum take-off AUM from a 1500m runway is 107200kg which allows for a maximum internal fuel load of 9141kg.

(Max AUM - Basic aircraft weight - Extra mass due to MP equipment - existing fuel load) = $(107200 - 52370 - 5689 - 40000 = 9141\text{kg})$.

Maximum AUM for F93BMPA to Operate from a 1500m Runway = 107200kg

The additional requirement that the F93B should be able to take-off from a semi-prepared runway 900m does not apply to the F93BMPA as all maritime aircraft operate from conventional bases with standard runway lengths.

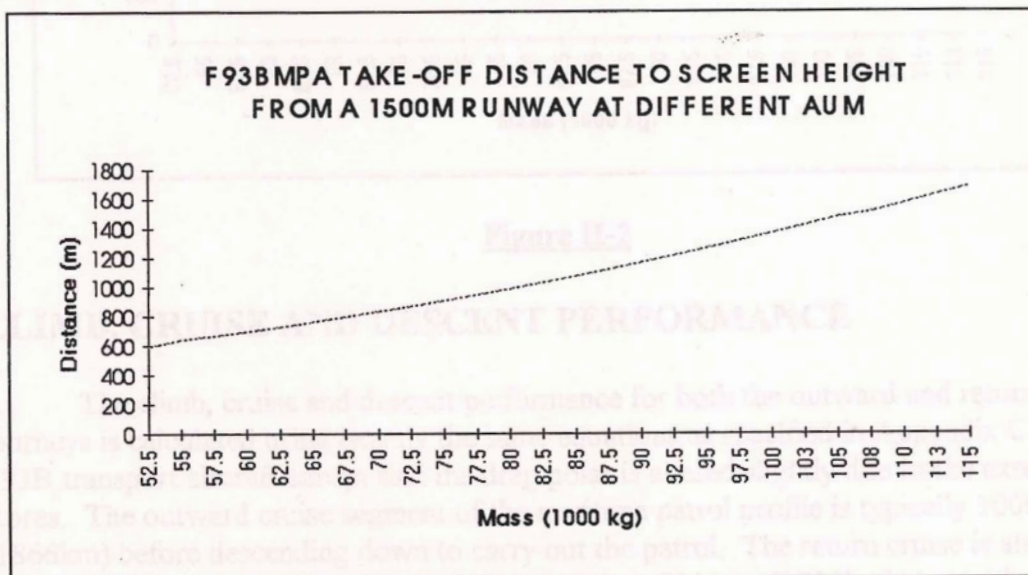
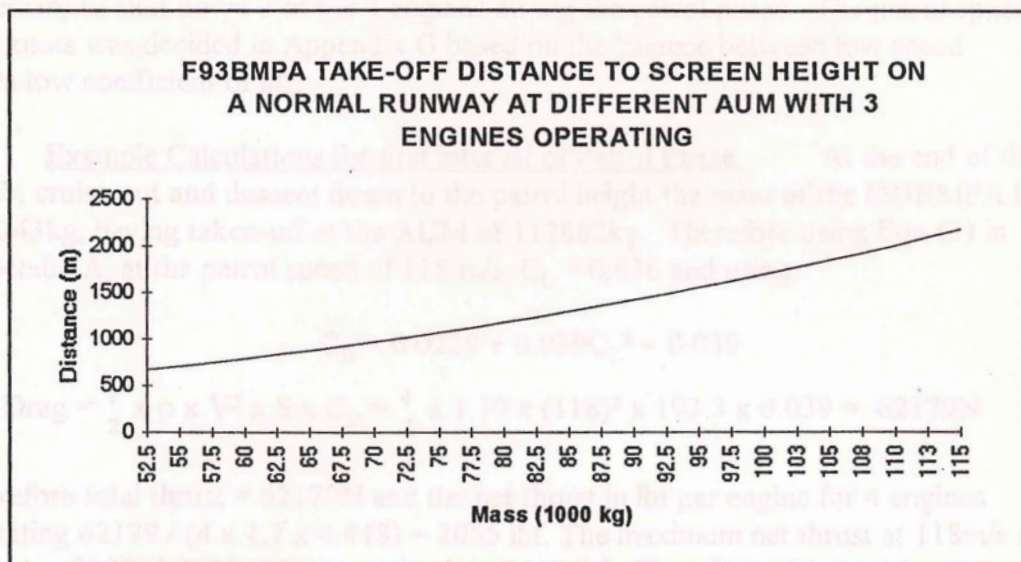


Figure H-1

However the capability of the F93BMPA to take-off with only 3 engines operating from a normal runway is a useful parameter, although it is unlikely to be practised very often. Fig H-2 shows the take-off distances required for a 3 engine take-off from a normal runway at different AUM.

**Figure H-2**

CLIMB, CRUISE AND DESCENT PERFORMANCE

3. The climb, cruise and descent performance for both the outward and return journeys is calculated using exactly the same equations as specified in Appendix C for the F93B transport aircraft except that the drag polar is altered slightly due to the external stores. The outward cruise segment of the maritime patrol profile is typically 1000nm (1866km) before descending down to carry out the patrol. The return cruise is also assumed to be 1000nm (1866km) which means that, 2000nm (3732km), a significant distance is covered without achieving any maritime surveillance. Therefore, it is important that as much fuel is carried as possible or air-to-air refuelling is considered.

PATROL PERFORMANCE

4. The patrol performance is calculated in exactly the same way as the cruise performance. The drag is equated to the thrust and a corresponding fuel flow and SFC is established for the net thrust per engine. However, the engine data available for at low altitudes and low thrust levels is fairly limited and Ref (10) was used to establish a relationship between fuel flow and reducing thrust. It was shown that in general as the thrust decreases from its maximum value the fuel flow drops initially and remains steady at approximately 3/4 of the fuel flow required at maximum thrust, until a point where the thrust requirement has decreased to around 1/3 of the maximum available when the fuel flow required rises almost exponentially. Therefore it is important not to reduce the thrust requirement for each engine below 1/3 of the maximum thrust available or the increased fuel flow and SFC will reduce the duration of the patrol due to a lack of fuel. In order to maintain the thrust required for the operating engines at a satisfactory level it may be

maintain the thrust required for the operating engines at a satisfactory level it may be necessary to shut down 2 of the 4 engines during the patrol phase. The patrol speed of 230 knots was decided in Appendix G based on the balance between low speed and a low coefficient of lift.

5. Example Calculations for first interval of Patrol Phase: At the end of the climb, cruise out and descent down to the patrol height the mass of the F93BMPA is 104243kg, having taken-off at the AUM of 112882kg. Therefore using Eqn (3) in Appendix A, at the patrol speed of 118 m/s, $C_L = 0.636$ and using:

$$C_D = 0.0229 + 0.039C_L^2 = 0.039$$

$$\text{and Drag} = \frac{1}{2} \times \rho \times V^2 \times S \times C_D = \frac{1}{2} \times 1.19 \times (118)^2 \times 193.3 \times 0.039 = 62179\text{N}$$

Therefore total thrust = 62179N and the net thrust in lbf per engine for 4 engines operating $62179 / (4 \times 1.7 \times 4.448) = 2055$ lbf. The maximum net thrust at 118m/s and an altitude of 1000ft (305m) is approximately 8460 lbf. Therefore with 4 engines operating the thrust required from each engine would be approximately 25% of the maximum thrust available which would incur heavy fuel flow penalties. However with only 2 engines operating the thrust required thrust per engine = 4110 lbf which is approximately 49% of the maximum thrust available per engine. Therefore for the best fuel economy 2 engines are shut down during the patrol phase and re-ignited prior to the climb up for the cruise home. The fuel flow required to provide the maximum thrust of 8460 lbf at 1000ft altitude = 0.395 kg/s. Therefore, as there is no specific manufacturers data available, the fuel flow required to satisfy a thrust requirement of 4110 lbf is assumed to be $0.75 \times 0.395 = 0.296$ kg/s from Ref (10). The SFC = Fuel flow / (Net thrust $\times 4.448$) = $0.296 / (4111 \times 4.448) = 0.000016$ kg/N/s. Throughout the patrol phase the fuel flow is assumed to 3/4 of the maximum fuel flow at an altitude of 1000ft.

LANDING PERFORMANCE

6. The landing performance is calculated using the equations derived in Appendix B. The landing mass depends not only on the take-off mass but the duration of the patrol completed and whether any of the stores, internal or external were released. The maximum landing distances of 1085m, with 4 engines operating, and 1152m, with 3 engines operating, are required when the aircraft returns shortly after take-off. Because the F93BMPA is a land based aircraft which would operate from a conventional airfield this requirement should not create any problems. However, fuel is often vented before such an aircraft re-lands so as to reduce the amount of braking required to stop the aircraft within the distance and to reduce the risk of fire should anything go wrong.

* 2 x Harrier missiles + 2 x ADM-9

** 2 x Stingray missiles + 2 x ADM-9

*** 2 x Hawk missiles + 2 x ADM-9

Table H-1

MISSION PROFILES

7. Table H-1 shows several aircraft configurations and their respective range and duration statistics. A reserve fuel factor of 5% is assumed for all missions which should allow for any diversions on returning to base. The basic aircraft mass = 52370kg.

MISSION PROFILES					
EQUIPMENT	1	2	3	4	5
Total internal surveillance Equipment + extra crew	2343kg	2343kg	2343kg	2343kg	2343kg
ESM pods + MAD	76kg	76kg	76kg	76kg	76kg
SAR Fit	90kg	90kg	90kg	90kg	90kg
Sonobuoys + Equipment	2336kg	NIL	2336kg	2336kg	2336kg
Weapons Load + Pylons	1334kg*	NIL	804kg**	1614kg***	804kg**
Chaff and Flare	40kg	NIL	40kg	40kg	40kg
TOTAL MPA EQUIP	6219kg	2509kg	5689kg	6499kg	5689kg
Extra Internal Fuel	NIL	12321kg	9141kg	8331kg	14293kg
TOTAL FUEL LOAD	40000kg	52321kg	49141kg	48331kg	54293kg
TOTAL TAKE-OFF AUM	98589kg	107200kg	107200kg	107200kg	112352
PATROL DURATION	6.2hrs	9.2hrs	8.4hrs	8.2hrs	9.7hrs
PATROL RANGE	2620km	3915km	3575km	3485km	4100km
TOTAL DURATION	11.1hrs	14.2hrs	13.4hrs	13.2hrs	14.6hrs
TOTAL RANGE	6233km	7546km	7215km	7127km	7758km

* 2 x Harpoon missiles + 2 x AIM-9

** 2 x Stingray missiles + 2 x AIM-9

*** 2 x Exocet missiles + 2 x AIM-9

Table H-1

APPENDIX I

UNIT COST ESTIMATION

Introduction

1. The F93B aircraft was envisaged to be in competition with the C-130-J, which is already available on the market, and the Future Large Aircraft (FLA); therefore a competitive selling price is required. One approach to finding a quick estimate of a new aircrafts selling price, the cost to the buyer, advocated by Ref (11) is called the top-down approach where existing aircraft are used to estimate the expected cost of a new aircraft. Having some idea of the selling price required to be competitive Ref (12) was used to calculate the actual unit cost of the F93B. This required the detailed estimation of the research and development costs and the acquisition costs.

TOP-DOWN APPROACH

2. The top-down approach to cost estimation is based on the premise that there is a correlation between existing aircraft prices and another attribute of the aircraft. This could be aircraft weight, range, maximum load or any other statistic which is readily available for comparison. In this case, due to the limited amount of accurate data available on military aircraft, the most appropriate statistic to compare aircraft prices against is maximum take-off weight.

AIRCRAFT COST COMPARISONS		
TYPE	MAX T/O WT in kg	COST IN \$ MILLIONS
Civilian		
747-400F	394 625	145
767-300ER	185 065	72
757-200PF	113 395	49
MD-11F	273 300	100
Military		
C-130J	70 310	45
C17	259 090	326
C-119G	33 778	4.4
C124	79 450	11.2
C141	143 600	28.7
C5	346 770	205
VC10	146 000	42
Tristar	195 045	74

Table I-1

Table I-1 shows the approximate cost and maximum take-off weight of a range of civilian and military aircraft. Figure I-1 shows the correlation between the cost of civilian freighters compared with their maximum take-off weight. The maximum take-off mass for the F93B is specified in Ref (1) as 101500 kg, therefore the F93B could be expected to cost just over US\$ 50 million based on Figure I-1.

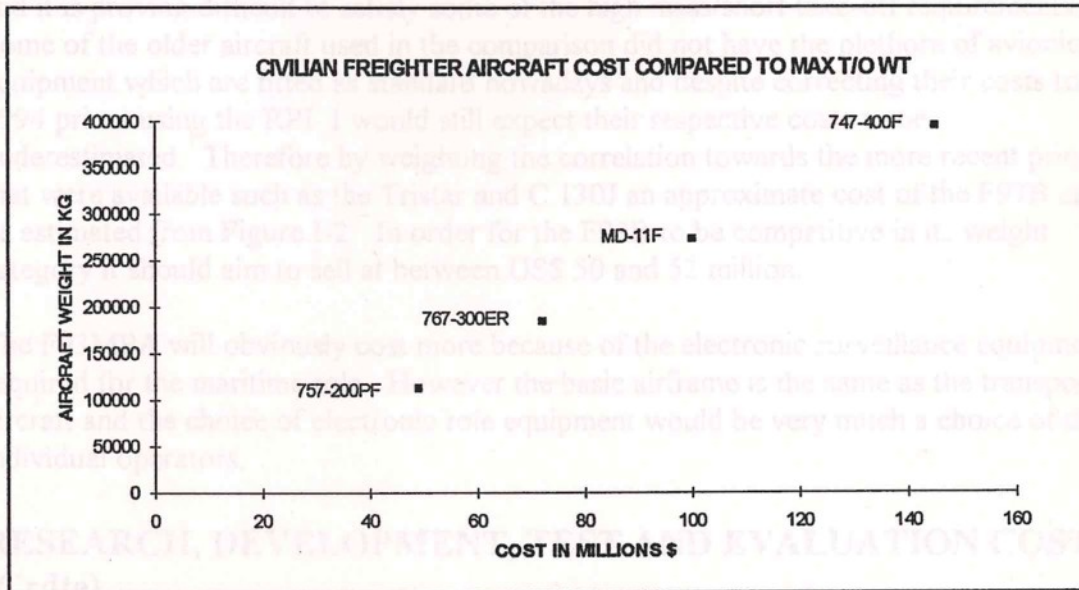


Figure I-1

The cost of military transport aircraft are compared in Figure I-2.

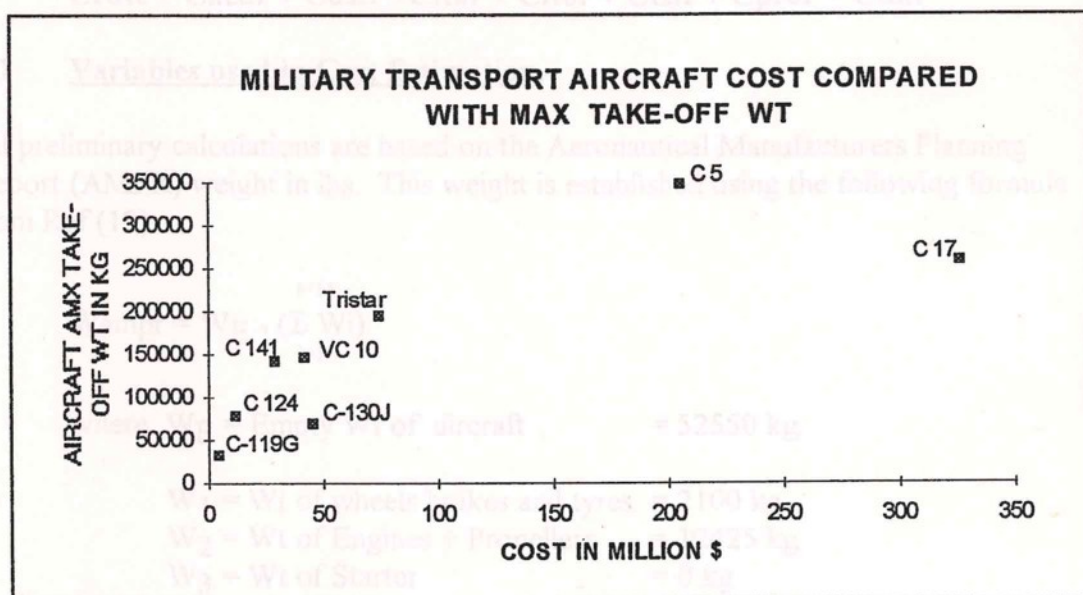


Figure I-2

Although the correlation between maximum take-off weight and cost is not as well defined for military aircraft, as civilian aircraft, by excluding the unrealistic cost of the C17 aircraft, an approximate correlation can be drawn between the remaining aircraft. It is thought that the C17 has already cost far in excess of predicted levels of expenditure, and it is still under development, because it is at the forefront of technology and it is proving difficult to satisfy some of the high mass/short take-off requirements. Some of the older aircraft used in the comparison did not have the plethora of avionics equipment which are fitted as standard nowadays and despite correcting their costs to 1994 prices using the RPI I would still expect their respective costs to be underestimated. Therefore by weighting the correlation towards the more recent prices that were available such as the Tristar and C 130J an approximate cost of the F93B can be estimated from Figure I-2. In order for the F93B to be competitive in its weight category it should aim to sell at between US\$ 50 and 52 million.

The F93MPA will obviously cost more because of the electronic surveillance equipment required for the maritime role. However the basic airframe is the same as the transport aircraft and the choice of electronic role equipment would be very much a choice of the individual operators.

RESEARCH, DEVELOPMENT, TEST AND EVALUATION COSTS (Crdte)

3. Research, development, test and evaluation cost.

This cost is divided into the following 7 components:

$$\text{Crdte} = \text{Caedr} + \text{Cdstr} + \text{Cftar} + \text{Cftor} + \text{Ctsfr} + \text{Cpror} + \text{Cfinr}$$

3.1 Variables used in Cost Estimation.

All preliminary calculations are based on the Aeronautical Manufacturers Planning Report (AMPR) weight in lbs. This weight is established using the following formula from Ref (12):

$$\text{Wampr} = \text{W}_E - \left(\sum_{i=1}^{i=11} \text{W}_i \right)$$

where W_E = Empty Wt of aircraft = 52550 kg

W_1 = Wt of wheels brakes and tyres = 2100 kg

W_2 = Wt of Engines + Propellers = 12425 kg

W_3 = Wt of Starter = 0 kg

W_4 = Wt of Cooling fluid = 10 kg

W_5 = Fuel cells = 0 kg

W_6 = Batteries and power supply = 2000 kg

W_7 = Avionics instruments	= 850 kg
W_8 = Electronic equipment	= 0 kg
W_9 = Air conditioning unit	= 800 kg
W_{10} = APU	= 300 kg
W_{11} = Trapped fuel and oil	= 260 kg

$$\sum_{i=1}^{i=11} W_i = 18745 \text{ kg}$$

$$\begin{aligned} \text{Therefore Wampr in kg} &= 33805 \text{ kg} \\ \text{Wampr in lbs} &= 74371 \text{ lbs} \end{aligned}$$

For military aircraft the maximum speed at sea level ($V_{\max}$) in knots is an important parameter used to calculate costs.

$$V_{\max} = 250 \text{ knots}$$

The number of prototype aircraft in this case is assumed to be 4, 1 for static fatigue testing and 3 for flight testing including 1 aircraft which is dedicated to the MPA role.

$$\begin{aligned} N_{rdte} &= 4 \\ N_{st} &= 1 \end{aligned}$$

The design and manufacture technology which would be used to produce the aircraft is fairly well established for transport aircraft. Therefore the difficulty factor would be:

$$F_{diff} = 1$$

Additionally, the use of CAD packages for design and development is assumed to be standard.

$$F_{cad} = 1$$

For a primary structure made of conventional aluminium alloy material :

$$F_{mat} = 1$$

Since no low observability technology is employed on the F93b:

$$F_{obs} = 1$$

3.2 Airframe Engineering and Design Cost during Research (Caedr).

$$Caedr = MHRaedr \times Rer$$

where MHRaedr are the manhours required to complete the airframe engineering and design and Rer is the labour rate for a design engineer.

$$\text{MHRaedr} = 0.0396(\text{Wampr})^{0.791}(\text{Vmax})^{1.524}(\text{Nrde})^{0.183}(\text{Fdiff})(\text{Fcad})$$

$$\text{MHRaedr} = 0.0396(74371)^{0.791}(250)^{1.524}(4)^{0.183}(1)(1)$$

$$\text{MHRaedr} = 1642907.1 \text{ Hours}$$

From Ref (13) the hourly wage of a design engineer (Rer) in 1994 is forecast to be US\$ 64.39.

$$\text{Therefore Caedr} = 1642907.1 \times 64.39$$

$$\text{Caedr} = \text{US\$ } 105\,786\,787$$

3.3 Development and Support and Testing Cost during Research (Cdstr)

$$\text{Cdstr} = 0.008325(\text{Wampr})^{0.873}(\text{Vmax})^{1.89}(\text{Nrde})^{0.346}(\text{CEF})(\text{Fdiff})$$

where CEF is a cost escalation factor which links prices in the past to current prices based on a particular price index.

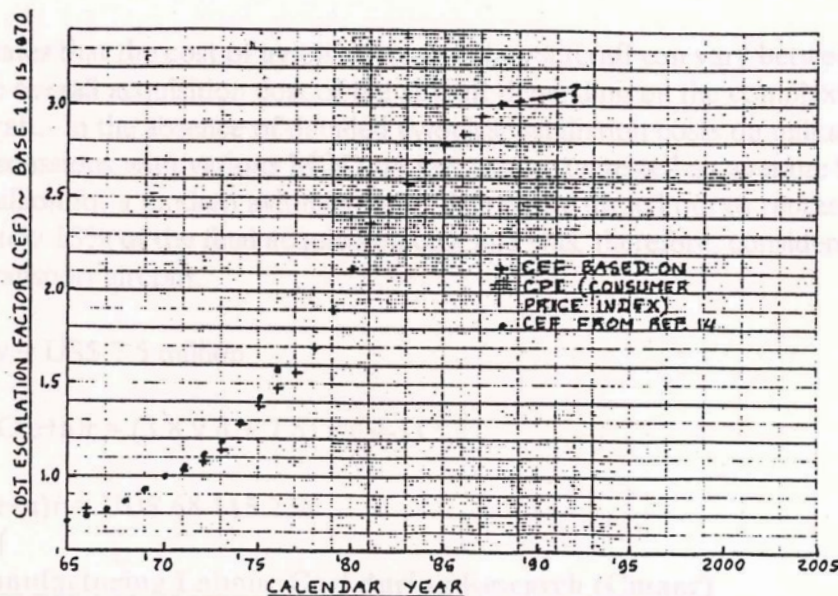


Fig I-3

Fig I-3 taken from Ref. (12) uses the US Consumer Price Index as a basis for cost escalation. $\text{CEF}_{1994} = 3.1$ Therefore:

$$\text{Cdstr} = 0.008325(74371)^{0.873}(250)^{1.89}(4)^{0.346}(3.1)(1)$$

$$\underline{Cdstr = US\$ 25\,403\,916}$$

3.4 Flight Test Costs for Research Aircraft (Cftar)

This cost is broken down as follows:

$$Cftar = C(e+a)r + Cmanr + Cmatr + Ctoolr + Cqcr$$

3.4.1 Cost of Engines and Avionics for Research C(e+a)r

$$C(e+a)r = (C_{er}N_e + C_{av}) \times (N_{rdte} - N_{st})$$

where N_e is the number of engines on the F93b = 4.

The engine chosen for the F93B in the design specification is a scaled up version of the Rolls Royce SR Turboprop (RB509-18). The scaling factor is 1.7 and it is assumed that the price quoted for the RB509-18 of US\$2.2 million in 1991 should also be scaled by 1.7. Therefore the price of an engine for the F93B in 1991 is US\$3.74 million. To obtain 1994 prices the cost is multiplied by (CEF 94/CEF 91). The cost escalation factor for 1991, from Fig I-3, is 3.05. Therefore each engine costs US\$ 3.74 x (3.1/3.05) in 1994:

$$C_e = \text{US\$ } 3.8 \text{ million}$$

Ref (12) states that the cost of avionics on a military aircraft can vary between 5% and 40% of the overall acquisition cost of the aircraft depending on the complexity and role of the aircraft. In the absence of detailed avionics installation costs on military transport aircraft, discussions with various MOD project teams suggested an average figure of US\$ 7.5 million for a modern avionics suite. This figure turned out to represent approximately 15% of the final acquisition cost and was, therefore, considered a suitable cost of a transport aircraft.

$$C_{av} = \text{US\$ } 7.5 \text{ million}$$

$$\text{Therefore } C(e+a)r = (3.8 \times 4 + 7.5) \times (4-1)$$

$$C(e+a)r = \text{UD\$ } 68\,115\,738$$

3.4.2 Manufacturing Labour Cost during Research (Cmanr)

$$Cmanr = MHR_{manr} \times Rmr$$

where Rmr is the hourly rate for manufacturing engineers/technicians = US\$ 54.58, from Ref (13) at 1994 prices.

$$MHR_{manr} = 28.984(W_{ampr})^{0.74}(V_{max})^{0.543}(N_{rdte})^{0.524}(F_{diff})$$

$$\text{MHR}_{\text{manr}} = 4\,836\,797 \text{ hrs}$$

$$\text{C}_{\text{manr}} = (4836797) \times 54.58$$

$$\text{C}_{\text{manr}} = \text{US\$ } 263\,992\,380$$

3.4.3 Research Material Costs (C_{matr})

$$\text{C}_{\text{matr}} = 37.632(\text{F}_{\text{mat}})(\text{W}_{\text{ampr}})^{0.689}(\text{V}_{\text{max}})^{0.624}(\text{N}_{\text{rdte}})^{0.792}(\text{CEF})$$

where $\text{F}_{\text{mat}} = 1$ for a primary structure made of conventional aluminium alloy material.

$$\text{C}_{\text{matr}} = \text{US\$ } 24\,915\,073$$

3.4.4 Research tooling Costs (C_{toolr})

$$\text{C}_{\text{toolr}} = \text{MHR}_{\text{toolr}} \times \text{R}_{\text{tr}}$$

where R_{tr} is the hourly rate for tooling engineers = US\$ 66.13

$$\text{MHR}_{\text{toolr}} = 4.0127(\text{W}_{\text{ampr}})^{0.764}(\text{V}_{\text{max}})^{0.899}(\text{N}_{\text{rdte}})^{0.178}(\text{N}_{\text{rr}})^{0.066}(\text{F}_{\text{diff}})$$

N_{rr} is the production rate during the RDTE phase which assumed to be a typical rate of 0.33 units/month.

$$\text{MHR} = 3\,600\,193 \text{ hrs}$$

$$\text{C}_{\text{toolr}} = 3600193 \times 66.13$$

$$\text{C}_{\text{toolr}} = \text{US\$ } 172\,809\,241$$

3.4.5 Quality Control Cost during Research (C_{qcr})

$$\text{C}_{\text{qcr}} = 0.13(\text{C}_{\text{manr}})$$

$$\text{C}_{\text{qcr}} = 0.13 \times 263992380$$

$$\text{C}_{\text{qcr}} = \text{US\$ } 34\,319\,009$$

Total cost of the flight test aircraft :

$$\text{C}_{\text{ftar}} = 68115738 + 263992380 + 24915073 + 172809241 + 34319009$$

$$\text{C}_{\text{ftar}} = \text{US\$ } 564\,151\,442$$

3.5 Cost of Research Flight Test Operations (Cftor)

$$C_{ftor} = 0.001244(W_{ampr})^{1.160}(V_{max})^{1.371}(N_{rdte}-N_{st})^{1.281} (CEF) (F_{diff}) (F_{obs})$$

$$C_{ftor} = \text{US\$ } 13\,670\,310$$

3.6 Test and Simulations Facility Cost (Ctsfr)

To determine the fatigue life of the aircraft structure the building of a dedicated fatigue test rig is required. The cost associated with this building is expressed as a cost adjustment factor (Ftsf) which is related to the overall RDTE cost.

$$\begin{aligned} C_{tsfr} &= F_{tsf} C_{rdte} \\ &= 0.03 C_{rdte} \end{aligned}$$

3.7 RDTE Profit (Cpror)

If a 10% profit is aimed for on the RDTE phase then the cost adjustment factor $F_{pror} = 0.1$. Therefore:

$$C_{pror} = 0.1 C_{rdte}$$

3.8 Cost to Finance the RDTE phase (Cfinr)

Assuming a compound interest rate of 10% over the time allocated to complete the RDTE phase then $F_{finr} = 0.1$. Therefore:

$$C_{finr} = 0.1 C_{rdte}$$

3.9 Final Expression for RDTE Cost (Crdte)

$$\begin{aligned} C_{rdte} &= C_{aodr} + C_{dstr} + C_{ftor} + C_{tsfr} + C_{pror} + C_{finr} \\ &= 105\,786\,787 + 25\,403\,916 + 564\,151\,442 + 13\,670\,310 + 0.03C_{rdte} + \\ &\quad 0.1C_{rdte} + 0.1C_{rdte} \end{aligned}$$

$$= 709\,012\,455 + 0.23C_{rdte}$$

$$C_{rdte} = \text{US\$ } 920\,795\,396$$

The costs incurred in the RDTE phase specified as a factor of the RDTE cost are as follows:

$$\text{Test and Simulation Facilities Cost (Ctsfr)} = \text{US\$ } 27\,623\,862$$

$$\text{RDTE Profit (Cpror)} = \text{US\$ } 92\,079\,539$$

$$\text{Cost to finance the RDTE phase (Cfinr)} = \text{US\$ } 92\,079\,539$$

ACQUISITION COSTS (Cacq)

4. The acquisition costs comprise two basic elements: total manufacturing costs (Cman) and profit (Cpro).

$$Cacq = Cman + Cpro$$

where $Cman = Caedm + Capcm + Cftom + Cfinm$

In order to provide an aircraft which can compete with rival transport aircraft the price must be competitive, which usually means as big a production run as possible. However it is rare for European military aircraft to be built in great numbers. Therefore it is assumed that a production run (Nm) of 200 aircraft is realistic for the European market. Including the research aircraft a total of 204 aircraft would be built for the whole program (Nprogram).

4.1 Airframe Engineering and Design Cost during Manufacture (Caedm)

$$Caedm = (MHRaedrprogram)(Rem) - Caedr$$

where Rem is the labour rate for manufacturing design engineers = Rer = US\$ 64.39
 $MHRaedrprogram = 0.0396(Wampr)^{0.791}(Vmax)^{1.524}(Nprogram)^{0.183}(Fdiff)(Fcad)$

$$= 0.0396(74371)^{0.791}(250)^{1.524}(204)^{0.183}(1)(1)$$

$$MHRaedrprogram = 3\,373\,637 \text{ hrs}$$

$$Caedm = (3\,373\,637)(64.39) - 105\,786\,787$$

$$Caedm = \text{US\$ } 111\,441\,699$$

4.2 Airframe Program Production Costs (Capcm)

$$C(apcm) = C(e+a)m + Cint + Cmanm + Cmatm + Ctoolm + Cqcm$$

4.2.1 Cost of Engines and Avionics for the Program (C(e+a)m)

$$C(e+a)m = [(Ce \times (CEF\,94/CEF\,91))(Ne) + Cav] \times [Nprogram]$$

$$= [(3\,740\,000 \times (3.1/3.05))(4) + 7\,500\,000] \times [204]$$

$$C(e+a)m = \text{US\$ } 4\,541\,049\,180$$

4.2.2 Cost of the Interior (Cintm)

The cost of the interior on this military transport is assumed to be negligible.

4.2.3 Manufacturing Labour Cost (Cmanm)

$$Cmanm = (MHRmanprogram \times Rmm) - Cmanr$$

where Rmm is the hourly labour rate for manufacturing engineers/technicians = Rmr = US\$ 54.58.

$$MHRmanprogram = 28.984(Wampr)^{0.74}(Vmax)^{0.543}(Nprogram)^{0.524}(Fdiff)$$

$$= 37\,959\,863 \text{ hrs}$$

$$Cmanm = (37\,959\,863 \times 54.58) - 263\,992\,380$$

$$Cmanm = \text{US\$ } 1\,807\,856\,926$$

4.2.4 Manufacturing Material Cost (Cmatm)

$$Cmatm = Cmatprogram - Cmatr$$

$$Cmatprogram = 37.632(Fmat)(Wampr)^{0.689}(Vmax)^{0.624}(Nprogram)^{0.792}(CEF\,94)$$

$$Cmatm = 560\,864\,235 - 24\,915\,073$$

$$Cmatm = \text{US\$ } 535\,949\,162$$

4.2.5 Manufacturing Tooling Cost (Ctoolm)

$$Ctoolm = (MHRtoolprogram \times Rtm) - Ctoolr$$

where Rtm is the hourly tooling manufacture rate = Rtr = US\$ 66.13.

$$MHRtoolprogram = 4.0127(Wampr)^{0.764}(Vmax)^{0.899}(Nprogram)^{0.178}(Nrm)^{0.066}(Fdiff)$$

where Nrm is the monthly airframe production rate during manufacture = 3

$$Ctoolm = (8\,385\,738 \times 66.13) - 172\,809\,241$$

$$Ctoolm = \text{US\$ } 381\,739\,625$$

4.2.6 Cost of Quality Control during Manufacture (Cqcm)

$$Cqcm = 0.13 \, Cmanm$$

$$Cqcm = \text{US\$ } 235\,021\,400$$

Total cost of airframe production program is therefore:

$$\begin{aligned} \text{Capcm} &= C(e+a)m + C_{int} + C_{manm} + C_{matm} + C_{toolm} + C_{cqm} \\ &= 4\,541\,049\,180 + 0 + 1\,807\,856\,926 + 535\,949\,162 + 381\,739\,625 + \\ &\quad 235\,021\,400 \end{aligned}$$

$$\text{Capcm} = \text{US\$ } 7\,501\,616\,294$$

4.3 Cost of Flight Test Operations (Cftom)

The cost of flight test operations is assumed to be negligible for this aircraft.

4.4 Cost of Financing the Manufacturing Phase (Cfin)

$$C_{finm} = 0.1 C_{man}$$

Total Cman costs are therefore:

$$\begin{aligned} C_{man} &= C_{aedm} + C_{acpm} + C_{ftom} + 0.1 C_{man} \\ &= 111\,441\,696 + 7\,501\,616\,294 + 0 + 0.1 C_{man} \end{aligned}$$

$$\text{Cman} = \text{US\$ } 8\,458\,953\,322$$

Therefore Cfin = US\$ 845 895 332

4.5 Profit Margin (Cpro)

$$C_{pro} = 0.1 C_{man}$$

$$\text{Cpro} = \text{US\$ } 845\,895\,332$$

4.6 Total Acquisition Cost

$$\begin{aligned} C_{acq} &= C_{man} + C_{pro} \\ &= 8\,458\,953\,322 + 845\,895\,332 \end{aligned}$$

$$\text{Cacq} = \text{US\$ } 9\,304\,848\,654$$

UNIT COST OF F93B

5. The unit cost of an aircraft is calculated using the RDTE costs plus acquisition costs divided by the number of aircraft manufactured for sale. i.e 200 in this case.

$$\text{Unit Cost} = (C_{acq} + C_{rdte})/N_m$$

$$= (9\,304\,848\,654 + 920\,795\,395)/200$$

Unit Cost of the F93B = US\$ 51 128 220

Figure I-4 shows the variation in unit cost of the F93B with the number of aircraft produced in the manufacturing run. It highlights the fact that it is not economical to produce less than 200 aircraft as the unit price would make the aircraft too expensive compared to rival aircraft such as the C-130J which costs approximately US\$45 million.

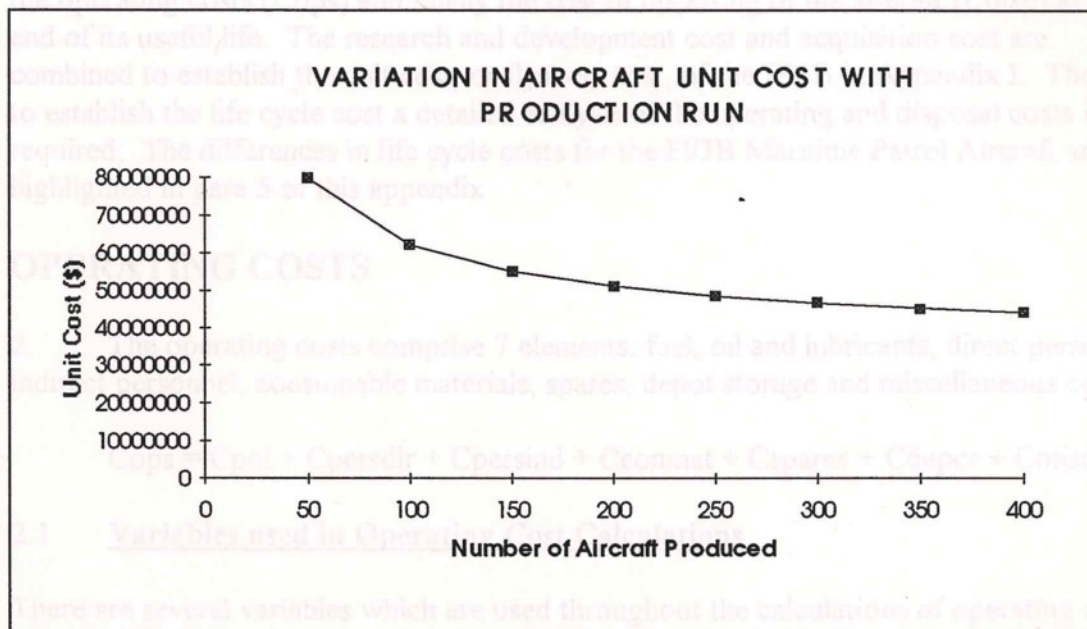


Figure I-4

UNIT COST OF F93MPA

6. The unit cost of the F93MPA is the basic cost of the F93B aircraft plus the cost for the additional equipment specified in Appendix F required to convert the aircraft to a maritime patrol role. Ref (4) specifies a total cost of US\$11.8 million to provide all the necessary equipment to carry out maritime surveillance. Therefore the unit cost of the F93BMPA = US\$ 51.1 + 11.8 million.

Unit Cost of F93BMPA = US\$ 63 million

APPENDIX J

LIFE CYCLE COST ESTIMATION

Introduction

1. For a military aircraft it is more important to establish the life cycle costs than the direct operating costs as military aircraft are not utilized to the same extent as civilian aircraft but they still require the same, if not more, maintenance. The total life cycle costs (Clcc) comprise the research and development costs (Crdte), the acquisition costs (Cacq), the operating costs (Cops) and finally the cost of disposing of the aircraft (Cdisp) at the end of its useful life. The research and development cost and acquisition cost are combined to establish the unit cost, or flyaway cost, of the F93B in Appendix I. Therefore to establish the life cycle cost a detailed analysis of the operating and disposal costs is required. The differences in life cycle costs for the F93B Maritime Patrol Aircraft are highlighted in para 5 of this appendix

OPERATING COSTS

2. The operating costs comprise 7 elements: fuel, oil and lubricants, direct personnel, indirect personnel, consumable materials, spares, depot storage and miscellaneous costs.

$$\text{Cops} = \text{Cpol} + \text{Cpersdir} + \text{Cpersind} + \text{Cconmat} + \text{Cspares} + \text{Cdepot} + \text{Cmisc}$$

2.1 Variables used in Operating Cost Calculations

There are several variables which are used throughout the calculations of operating cost. These calculations are based on the F93B transport aircraft and any differences in the variables for the F93BMPA are detailed in para 5.

- a. Annual utilization (Uann) in flt hrs for the F93B = 800 hrs
- b. Average mission time (tm) = 8.75 hrs
- c. Number of missions flown per year (Nmission) = $\text{Uann}/\text{tm} = 91.4$
- d. Number of years in active service (Nyr) = 30
- e. Loss rate per 100 000 flying hrs (Lr) from Ref (12) = 0.5
- f. A fleet size of 60 aircraft would be required to replace the current RAF transport fleet. Therefore it is assumed that the RAF would acquire the same number of replacement aircraft (Nacq) = 60

g. No of reserve aircraft (N_{res}) = $0.1 N_{acq}$ = 6

h. No of aircraft in service (N_{serv}) = (N_{acq}) - (N_{res}) - $0.5(N_{loss})$

where (N_{loss}) is the number of aircraft lost during the service life:

$$\begin{aligned} N_{loss} &= (L_r)(U_{ann})(N_{yr})(N_{serv})/100\ 000 \\ &= (0.5)(800)(30)(N_{serv})/100 \\ &= 0.12 N_{serv} \end{aligned}$$

$$\begin{aligned} \text{Therefore } N_{serv} &= 60 - 6 - 0.5(0.12 N_{serv}) \\ &= 54 - 0.06 N_{serv} \end{aligned}$$

$$N_{serv} = 50.94$$

$$\text{Therefore } N_{loss} = 0.12 \times 50.94 = 6.1$$

2.2 Program Cost of Fuel, Oil and Lubricants (C_{pol})

$$C_{pol} = [(F_{ol})(W_{fused})\{(F_{P}/F_{D})\}(N_{mission})(N_{serv})(N_{yr})]$$

where:

F_{ol} is the cost factor for oil and lubricants = 1.005.

W_F is the mission fuel used in lbs per mission. For the F93B the design mission was a distance of 5500 km which required 22577 kg of fuel; Therefore $W_F = 49670$ lbs.

F_P is the price of fuel in US\$ per gallon. Ref (12) quoted a price of US\$ 0.8 per gallon.

F_D is the fuel density in lbs per gallon. Jet A = 6.74 lbs/gallon.

$$\text{Therefore } C_{pol} = [(1.005)(49670)(0.8/6.74)(91.43)(49.27)(30)]$$

$$C_{pol} = \text{US\$ } 800\ 724\ 006$$

2.3 Cost of Direct Personnel ($C_{persdir}$)

$$C_{persdir} = C_{crew} + C_{mpersdir}$$

2.3.1 Cost of aircrew (C_{crew})

$$C_{crew} = (N_{serv})(N_{crew})(R_{cr})(P_{aycr})(OHR_{crew})(N_{yr})$$

where:

Ncrew is the number of crew on the aircraft. Ref (12) initial specified a crew of only 2. Although this would be suitable for simple transportation and ferry tasks, an additional crewman would be required during paratroop or resupply tasks. Therefore the operating costs have been calculated using 3 crew members as the maximum operating costs. Details of the additional crew required for the maritime patrol task are specified in para 5.

Rcr is the crew ratio per aircraft required to keep the aircraft airborne. Ref (12) specifies a value of 1.5 for military transport aircraft which fly less than 1200 hours per year. Crew pay is based on a senior Flight Lieutenants wage of US\$ 45 000 per year.

The OHRcrew is an overhead rate factor which is associated with actual crew pay and from Ref (12) is assumed to be a value of 3.

$$\text{Therefore } C_{\text{crew}} = (49.27)(3)(1.5)(45\,000)(3)(30)$$

$$C_{\text{crew}} = \text{US\$ } 897\,945\,750$$

2.3.2 Cost of Direct Maintenance Personnel (Cmpersdir)

$$C_{\text{mpersdir}} = (N_{\text{serv}})(N_{\text{yr}})(U_{\text{annflt}})(M_{\text{HRflthr}})(R_{\text{Mml}})$$

where:

MHRflthr is the number of maintenance manhours required per flying hour. The RAF C-130 Hercules has a very low MHRflthr of around 7 manhours per flying hour. Whilst the F93B may initially require a greater number of maintenance manhours per flying hour it is assumed that it would reduce to a similar rate to that of the C-130. Therefore an average MHRflthr rate of 8 maintenance manhours per flying hour is assumed for the F93B.

RMml is the military maintenance labour rate which was US\$ 45 per hour in 1989. Therefore, applying the cost escalation factor for 1994:

$$R_{\text{Mml}} = 45(3.1/3.02) = \text{US\$ } 46.2 \text{ per hour}$$

$$\text{Therefore } C_{\text{mpersdir}} = (49.27)(30)(800)(8)(46.2)$$

$$C_{\text{mpersdir}} = \text{US\$ } 437\,044\,608$$

$$\text{Total } C_{\text{persdir}} = C_{\text{crew}} + C_{\text{mpersdir}} = 897\,945\,750 + 437\,044\,608$$

$$\underline{C_{\text{persdir}} = \text{US\$ } 1\,334\,990\,358}$$

2.4 Program Cost of Indirect Personnel (Cpersind)

This cost component covers the cost of all squadron personnel which are not directly involved in flight operations or aircraft maintenance.

$$Cpersind = (f persind)(Cops)$$

where f persind is the operating cost fraction for each type of aircraft. Ref (12) specifies a fraction of 0.13 for the C-141 which is the nearest aircraft type to the F93B. Therefore:

$$Cpersind = 0.13 (Cops)$$

2.5 Program Cost of Consumable Spares (C conmat)

This cost component covers the cost of consumable materials used in conjunction with military aircraft maintenance functions.

$$C conmat = (Nserv)(Nyr)(Uannflt)(MHRflthr)(R conmat)$$

where R conmat is the average cost per maintenance hour for consumable materials. Ref (12) specifies a figure of US\$ 6.5 for 1989. Therefore:

$$R conmat = 6.5 (3.1/3.02) = US\$ 6.67$$

$$C conmat = (49.27)(30)(800)(8)(6.67)$$

$$C conmat = US\$ 63 117 803$$

2.6 Program Cost of Spares (C spares)

$$C spares = (f spares)(Cops)$$

where f spares from Ref (12) is 0.16 for the C-141. Therefore the same figure is assumed for the F93B.

$$C spares = 0.16 (Cops)$$

2.7 Program Cost Depot Maintenance (C depot)

Many overhaul and maintenance actions on military aircraft are not carried out at the squadron or wing level anymore. Instead, they are carried out at third and fourth line levels often referred to as depots.

$$C depot = (f depot)(Cops)$$

where f_{depot} is assumed to be = 0.15 from Ref (12)

$$\underline{C_{\text{depot}} = 0.15 (C_{\text{ops}})}$$

2.8 Program Cost of Miscellaneous Items (C_{misc})

Miscellaneous cost elements include: the requirement for technical data to support maintenance functions, the requirement for training, training data and training equipment and the requirement for support equipment.

$$C_{\text{misc}} = (f_{\text{misc}})(C_{\text{ops}})$$

where f_{misc} is assumed to be = 0.01 from Ref (12). Therefore:

$$\underline{C_{\text{misc}} = 0.01(C_{\text{ops}})}$$

2.9 Total Operating Costs (C_{ops})

This is the overall cost of operating a fleet of aircraft over a specified time period.

$$C_{\text{ops}} = C_{\text{pol}} + C_{\text{persdir}} + C_{\text{persind}} + C_{\text{conmat}} + C_{\text{spares}} + C_{\text{depot}} + C_{\text{misc}}$$

Therefore:

$$C_{\text{ops}} = 800\,724\,006 + 1\,334\,990\,358 + 0.13(C_{\text{ops}}) + 63\,117\,803 + 0.16(C_{\text{ops}}) + 0.15(C_{\text{ops}}) + 0.01(C_{\text{ops}})$$

$$C_{\text{ops}} = 2\,198\,832\,227 / [1 - (0.13 + 0.16 + 0.15 + 0.01)]$$

$$\underline{C_{\text{ops}} = \text{US\$ } 3\,997\,876\,776}$$

Therefore:

$$C_{\text{persind}} = \text{US\$ } 519\,723\,981$$

$$C_{\text{spares}} = \text{US\$ } 639\,660\,284$$

$$C_{\text{depot}} = \text{US\$ } 599\,681\,516$$

$$C_{\text{misc}} = \text{US\$ } 39\,978\,768$$

2.10 Operating Costs per Flight Hour

In a commercial market the cost of operating the aircraft per flight hour is an important evaluation figure.

$$\text{Operating cost per flt hr} = C_{\text{ops}} / [(N_{\text{serv}})(N_{\text{yr}})(U_{\text{annflt}})]$$

$$= 3\,997\,876\,776 / [(49.27)(30)(800)]$$

Operating Cost per Flight Hour = US\$ 3 381

DISPOSAL COSTS

3 There comes a time when any aircraft no longer has military value. That point is normally reached when:

- a. The aircraft has reached the end of its safe structural life and structural repairs are judged to be uneconomical.
- b. The aircraft has reached the end of its economic life; it can no longer compete effectively in the face of more modern aircraft.
- c. The aircraft ceases to have any military value. For example, its mission is no longer a viable one, or improvements in technology have rendered the aircraft obsolete.
- d. The aircraft has been damaged beyond repair.

Whenever an aircraft ceases to have any military value it needs to be disposed of. The cost of disposing of aircraft depends on the types of materials used in the construction and operation of the aircraft. Disposal usually consists of:

- a. Temporary storage
- b. Draining of liquids and disposal thereof.
- c. Disassembly of engines and certain systems, such as computers and instruments.
- d. Cutting up of the airframe and disposal of the resulting materials.

There are costs associated with each of these actions. However, it is also possible that there is a salvage value for some items, such as engines, instruments and scrap metals. Therefore, any disposal costs are partially offset by these resale values. Unfortunately there is very little data available on the calculation of disposal costs. Therefore the disposal cost is assumed to be 0.01 of the life cycle cost from Ref (12).

LIFE CYCLE COST

4. Total Life Cycle Cost

$$\text{Total Clcc} = \text{Crdte} + \text{Caqc} + \text{Cops} + \text{Cdisp}$$

$$= \text{Crdte} + \text{Cacq} + \text{Cops} + 0.01\text{Clcc}$$

If there is a production run of 204 aircraft then the unit cost for each aircraft is US\$51 128 220 (See Appendix I). Assuming the RAF purchase 60 airframes to replace their transport fleet then the total Crdte and Cacq cost to the RAF is the unit cost x no of aircraft bought.

$$\text{Life cycle cost for RAF fleet} = [(51\ 128\ 220) \times (60) + 3\ 997\ 876\ 776] / (1 - 0.01)$$

$$\text{Total life cycle cost for RAF fleet} = \text{US\$ } 7\ 136\ 939\ 370$$

$$\text{Therefore Cdisp} = \text{US\$ } 71\ 369\ 393$$

4.1 Unit Life Cycle Cost

The total life cycle cost for the fleet is an important figure for the government budget holders who may have to spread the costs over several years. However an equally important figure is the unit life cycle cost which shows whether the aircraft is competitive and may decide how many aircraft the government can actually afford to buy.

$$\text{Unit Clcc} = \text{Clcc} / \text{Nacq}$$

$$= 7\ 136\ 939\ 370 / 60$$

$$\text{Unit life cycle cost} = \text{US\$ } 118\ 948\ 989$$

F93B MARITIME PATROL VARIANT COST ESTIMATION

5. Differences in Operating Cost

In order to operate a continuous maritime surveillance commitment a fleet of approximately 24 aircraft would be needed to replace the RAF Nimrod fleet. Although the F93BMPA variant is only a role conversion from the transport aircraft to the maritime patrol role the MPA aircraft will cost more to operate due to its increased mission duration and the number of crew members required. The following variables are changed for maritime operations:

- a. The annual utilization (Uann) is assumed to be 1000 flying hours.
- b. Although the maximum endurance mission is of 14 hours duration, the typical peace time mission will be of 6 to 8 hours duration without a relief crew on board. Therefore for cost purposes the average mission time (tm) is assumed to be 10 hours which allows the crew to remain within their peacetime crew duty limit of 14 hours.

c. The number of missions (N_{mission}) flown per year = $U_{\text{ann}}/t_m = 100$

d. The number of aircraft required to fill the MPA role depends on whether a full time commitment is envisaged or a limited response is required. It is assumed for cost purposes that a fleet of 24 aircraft are acquired for the RAF maritime role to take over from the Nimrod in the short or long term. Therefore $N_{\text{acq}} = 24$ and the number of reserve aircraft (N_{res}) = 2.4.

e. Using the equation stated in Para 2.1.h, and reducing the loss rate (L_r) to 0.3 per 100 000 flying hours, the number of serving aircraft (N_{serv}) = 20.67 and the number of aircraft lost throughout 30 years of service = 1.86

f. The number of crew members has increased to 8 without a relief crew.

g. The number of maintenance manhours per flight hour (MHR_{flthr}) is assumed to have increased to at least 12 manhours due to the addition of the maritime electronic equipment.

h. The fuel used during an average mission (W_{Fused}) = 76826 lbs.

All remaining variables and multiplying factors remain unchanged for the maritime variant of the F93B.

5.1 MPA Operating Cost

The operating cost is calculated using exactly the same formula as for the transport aircraft:

$$C_{\text{ops}} = [C_{\text{pol}} + C_{\text{persdir}} + C_{\text{conmat}}]/[1 - 0.13 - 0.16 - 0.15 - 0.01]$$

5.1.1 Program Cost of Fuel, Oil and Lubricants (C_{pol})

$$C_{\text{pol}} = [(F_{\text{ol}})(W_{\text{Fused}})(F_{\text{P}}/F_{\text{D}})(N_{\text{mission}})(N_{\text{serv}})] \times N_{\text{yr}}$$

$$C_{\text{pol}} = (1.005)(76826)(0.8/6.74)(100)(20.67)(30)$$

$$C_{\text{pol}} = \text{US\$ } 568\,284\,886$$

5.1.2 Program Cost of Direct Personnel (C_{persdir})

$$C_{\text{persdir}} = C_{\text{crew}} + C_{\text{mpersdir}}$$

where $C_{\text{crew}} = (N_{\text{serv}})(N_{\text{crew}})(R_{\text{cr}})(\text{Paycr})(\text{OHR}_{\text{crew}})(N_{\text{yr}})$

$$= (20.67)(8)(1.5)(45000)(3)(30)$$

$$C_{\text{crew}} = \text{US\$ } 1\,004\,562\,000$$

$$\begin{aligned} \text{and } C_{\text{mpersdir}} &= (N_{\text{serv}})(N_{\text{yr}})(U_{\text{annflt}})(M_{\text{HRflthr}})(R_{\text{mml}}) \\ &= (20.67)(30)(1000)(12)(46.19) \end{aligned}$$

$$C_{\text{mpersdir}} = \text{US\$ } 343\,709\,028$$

$$\text{Therefore } C_{\text{persdir}} = 1\,004\,562\,000 + 343\,709\,028$$

$$C_{\text{persdir}} = \text{US\$ } 1\,348\,271\,028$$

5.1.3 Program Cost of Consumable Materials (C_{conmat})

$$\begin{aligned} C_{\text{conmat}} &= (N_{\text{serv}})(N_{\text{yr}})(U_{\text{annflt}})(M_{\text{HRflthr}})(R_{\text{conmat}}) \\ &= (20.67)(30)(1000)(12)(6.67) \end{aligned}$$

$$C_{\text{conmat}} = \text{US\$ } 49\,648\,721$$

5.2 Total MPA Operating Cost (C_{ops})

$$C_{\text{ops}} = [568\,284\,886 + 1\,348\,271\,028 + 49\,648\,721]/[1 - 0.45]$$

$$\text{Total MPA Operating Cost} = \text{US\$ } 3\,574\,917\,518$$

Therefore:

$$C_{\text{persind}} = \text{US\$ } 464\,739\,277$$

$$C_{\text{spares}} = \text{US\$ } 571\,986\,803$$

$$C_{\text{depot}} = \text{US\$ } 536\,237\,628$$

$$C_{\text{misc}} = \text{US\$ } 35\,749\,175$$

5.3 Operating Costs per Flight Hour

$$= C_{\text{ops}}/[(N_{\text{serv}})(N_{\text{yr}})(U_{\text{annflt}})]$$

$$= 3\,574\,917\,518/[(20.67)(30)(1000)]$$

$$\text{Operating Cost per Flight Hour} = \text{US\$ } 5\,765$$

6 Life Cycle Cost (Clcc)

Using the same equation as that used for the F93B transport aircraft:

$$\text{Total Clcc} = \text{Total (Crdte + Cacq + Cops)} / (1 - \text{Cdisp})$$

The unit cost (Crdte + Cacq) for the F93BMPA airframe is identical to the F93B transport aircraft = US\$51 128 220. The additional cost of the maritime surveillance equipment and stores is approximately US\$ 11.8 million from Ref (4). Therefore the total unit cost for the F93BMPA is US\$62 928 220. The total Crdte and Cacq cost for the RAF MPA fleet is the unit cost x the number of aircraft purchased.

$$\text{Total Crdte + Cacq} = 62\,928\,220 \times 24 = \text{US\$ } 1\,510\,277\,280$$

$$\text{Total Clcc} = (1\,510\,277\,280 + 3\,574\,917\,518) / (1 - 0.01)$$

$$\text{Total F93BMPA Life Cycle Cost} = \text{US\$ } 5\,136\,560\,402$$

$$\text{Therefore Cdisp} = \text{US\$ } 51\,365\,604$$

7. Unit Life Cycle Cost

$$\text{Unit life cycle cost} = \text{Clcc} / \text{Nacq} = 5\,136\,560\,402 / 24$$

$$\text{Unit Life Cycle Cost} = \text{US\$ } 214\,023\,350$$