Cranfield University

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Performance Investigations and Systems Architectures for the More Electric Aircraft



School of Engineering

PhD Thesis

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Executive Summary

Implementation of all or more electric technologies has the potential of improving the performance of a given aircraft. Savings can manifest through improved engine performance, reduced equipment weight and improved secondary power and systems utilization. Despite the potential benefits and the amount of research invested the concept has not been implemented yet and its impact on the overall performance of the aircraft remains unclear.

Adoption of a single form of secondary power and implementation of more or all electric technologies is claimed to offer a number of distinct advantages over the conventional secondary power systems. These claims include the following:

1. Improved engine performance through the optimization of off-takes and elimination of bleed air. These issues is said to become critical as the bypass ratio of the engine increases and the size of the core reduces. However, despite concerns over the capability of high bypass ratio to provide secondary power requirements of future aircraft the claims remain unproven and it is generally believed that off-take penalties are not dependent on the configuration of the engine and can be kept to a minimum if the engine is designed to provide them.

2. Improved system utilization and implementation of more efficient power units. Conventional secondary power systems developed during the years that fuel was relatively fuel was relatively inexpensive and have not been optimized for maximum efficiency. Bleeding air from the engine is largely inefficient and a large amount of power is wasted for no useful purpose. In addition, hydraulics are often sized for loads that are small in duration thus increasing the mass and power requirements of the system.

3. Elimination of conventional hydraulic and bleed air systems could lead to a significant mass reduction. However, the mass and size of the electrical equipment would also increase and these changes must be accounted for.

4. Additional benefits include reduced single type maintenance, improved reliability and life cycle costs.

The present study is focused on assessing the potential benefits of adopting the all or more electric aircraft concepts in the case of high capacity long-range aircraft. Previous studies concentrated on low to medium range aircraft powered by medium bypass bass ratio engines. As a result, the moderate power requirements and the more important weight considerations, for the class of aircraft reviewed, reduced the potential benefits and led to retrofitted designs with low savings. It is thought that high capacity long range aircraft as those examined by the present study are more suited to the all electric concepts and could benefit more.

The study was aimed to address the first three claims made and also provide a better understanding of the issues involved. As a result the following topics have been considered.

The effects of off-takes on the performance of the engine were studied. Two methods for assessing these effects were developed. The analysis concluded that engine parameters such as pressure ratio and turbine entry temperature have an important effect on the performance of an engine operating with bleed air or shaft power extraction.

An aircraft performance model was developed and used to asses the impact of all or more electric technologies on aircraft in the classes of A330-200, A340-500 and A380-100. The models were validated against published payload range data. All electric derivatives were developed by incorporating changes in aircraft mass and engine off-takes. Sensitivity studies were also conducted.

Conceptual designs and electric system architectures were developed and studied. Adoption of all electric technologies and elimination of the hydraulic system was estimated to reduce the OEW by 0.4% to 0.1% for the A330-200 class aircraft, 0.37% to 0.25% for the A340-500 class aircraft and 0.3% to 0.1% for the A380-100 class aircraft. The total fuel reduction with implementation of all electric technologies and elimination of hydraulics and bleed air could be as high as 2.6% for the A330-200 class aircraft, 2.75% for the A340-500 class aircraft and 3.5% for the A380-100 class aircraft.

Adoption of all or more electric technologies could allow for the design of more efficient aircraft. Increasing the aspect ratio of the A80-100 class aircraft to 8.33 could improve aerodynamic efficiency leading to an overall fuel reduction of 4.4% and a MTOM reduction of 2.0%. In the case of twin engine aircraft reductions in MTOM and reduced off-take

penalties could allow an increase in engine bypass ratio leading to increased overall performance. For the A330-200 class aircraft elimination of bleed air and increase in bypass ratio resulted in a fuel reduction of 4.5%.

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Nomenclature

AB	Afterburner
ACEE	Aircraft Eneergy Efficient Program
AEA	All Electric Aircraft
AEE	All Electric Engine
AGB	Accessory Gearbox
APU	Auxiliary Power Unit
BRW	Break Release Weight
BWB	Blended Wing Body
CFD	Computational Fluid Dynamics
cg	Centre of Gravity
CoA	College of Aeronautics
CSCF	Constant Speed Constant Frequency
CSD	Constant Speed Drive
DOC	Direct Operating Costs
ECS	Environmental Control System
E^3	Energy Efficient Engine
EHA	Electro-Hydraulic Actuator
EMA	Electro Mechanical Ancuator
FAR	Fuel to Air Ratio
FBW	Fly by Wire
FOD	Foreign Object Damage
FRP	Fibre Reinforced Plastics
GRA	Geared Rotary Actuator
HBPR	High Bypass Ratio
HLFC	Hybrid Laminar Flow Control
HP	High Pressure
IDEA	Integrated Digital Electric Aircraft
IDG	Integrated Drive Generator
IGV	Inlet Guide Vanes
IP	Intermediate Pressure
ISA	International Standard Atmosphere
LFC	Laminar Flow Control
LP	Low Pressure
LVDT	Linear Variable Differential Transformer
MEA	More Electric Aircraft
MEE	More Electric Engine
MTOW	Maximum Take Off Weight
NDM	Non Dimensional Mass Flow
OEW	Operating Empty Weight
PR	Pressure Ratio
RAT	Ram Air Turbine
ROC	Rate o Climb (m/s)
ROD	Rate of Descent (m/s)
SCR	Silicon Control Rectifiers

sfc	Specific Fuel Consumption (mg/Ns)
SPS	Secondary Power Systems
SRM	Switched Reluctance Machine
ST	Specific Thrust (N/kg/s)
TET	Turbine Entry Temperature (Kelvin)
TOC	Top of Climb
TOD	Top of Descent
TOW	Take Off Weight
TRU	Transformer Rectifier Unit
VFCV	Variable Frequency Constant Voltage
VSCF	Variable Speed Constant Frequency
VVVF	Variable Voltage Variable Frequency
ZFW	Zero Fuel Weight

Chapter 1: Introduction & Project Specifications

The chapter defines the context of the study along with its specifications and deliverables. A definition of secondary power systems is given along with that of All Electric Aircraft/Engine concepts. The findings of previous studies are presented together with a discussion on the novelty and contribution of the present study.

1.0 Introduction

The scope of this research work was to investigate the performance and systems architectures of aircraft that may in future utilise an increasingly More/All Electric approach for their secondary power systems (SPS). Secondary power is the power distributed around the engine and airframe systems and not used for propulsion, this being primary power. On most aircraft secondary power is distributed in three forms:

- electric

- hydraulic

- bleed air or pneumatic

Hydraulic power is used for flight controls and other actuation functions whilst electrical power is required by the avionics, instruments, utilities etc. Bleed air is mainly used for the supply of the environmental control system (ECS), cabin pressurisation, ice protection system and finally engine starting - though in some cases actuation functions are also performed.

The most obvious source of secondary power on an aircraft is the main propulsion unit. Since, the engines must be present on all powered aircraft it is sensible to use them as a source of secondary as well as primary power. Past and current turbofan and turbo jet engines are capable of supplying secondary power in two forms:

i) Shaft power is extracted from the engine's high pressure shaft to drive a gearbox on which secondary power system components are mounted in addition to the engine's fuel and oil pumps

ii) Hot, pressurised air is extracted from the engine's compressors

Alternative secondary power sources while on ground and/or emergencies may include an auxiliary power unit, ground connections and the use of a ram air turbine (RAT). Although extraction and distribution of secondary power is necessary and inevitable, it comes with its problems and constrains. Airframe and engine systems constitute a considerable proportion of the aircraft's initial cost. In addition, they increase operating costs due to their mass, power consumption and maintenance requirements. Therefore, great emphasis should be placed in reducing both capital and operating costs by minimizing the detrimental effects of supplying and distributing secondary power, while increasing the efficiency and performance of a given aircraft as a whole system. The desired cost reduction can manifest itself as a reduction in operating costs due to fuel savings and improved maintenance, or capital cost due to commonality and reduced complexity. It appears that a change in the pattern of secondary power could offer savings in both areas.

1.1 Advanced Secondary Power Systems

The configuration of present (conventional) secondary power systems –mentioned above and shown in Figure 1.1- was not fully determined until the post war period and essentially the introduction of the jet engine. The development of the gas turbine engine together with the continuing increase of aircraft speed and size meant that environmental control, engine/airframe ice protection and power assisted controls were required. The advances in hydraulic technology, which enabled high power density and relatively simple solutions, coupled with the ability of the gas turbine engine to supply hot pressurised air in a simple manner, led to the adoption and development of the present configuration of SPS. Despite their maturity and relatively simple design present SPS have a number of major drawbacks including: power consumption, maintenance requirements, size and overall mass.

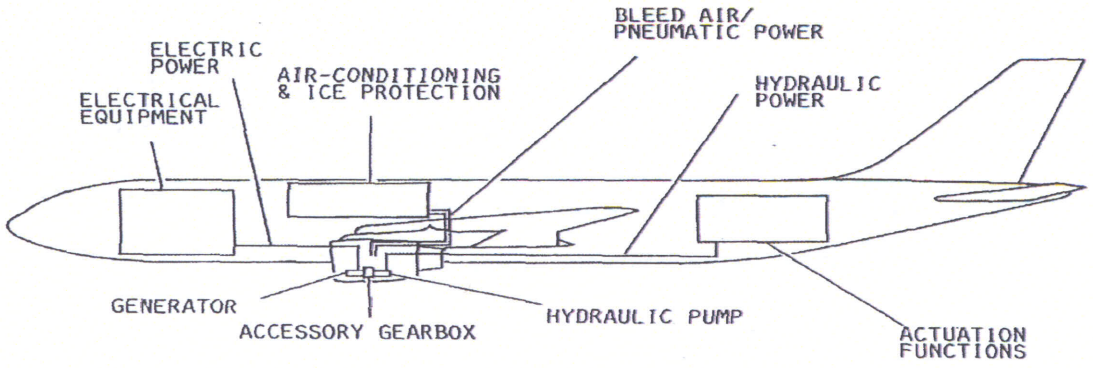


Figure 1.1 Conventional Secondary Power Systems [1]

Advanced SPS studies, often referred to as All/More Electric Aircraft (M/AEA) studies, proposed that the traditional form of secondary power should be replaced by electrically powered systems, Figure 1.2. This is on the basis that the amount of electrically powered equipment on aircraft has increased rapidly over the years. Most of this electrically powered equipment can only be powered in this way whilst hydraulic and pneumatic functions can also be electrically powered. Thus, adoption of the AEA concept could reduce complexity and also improve reliability and maintenance.

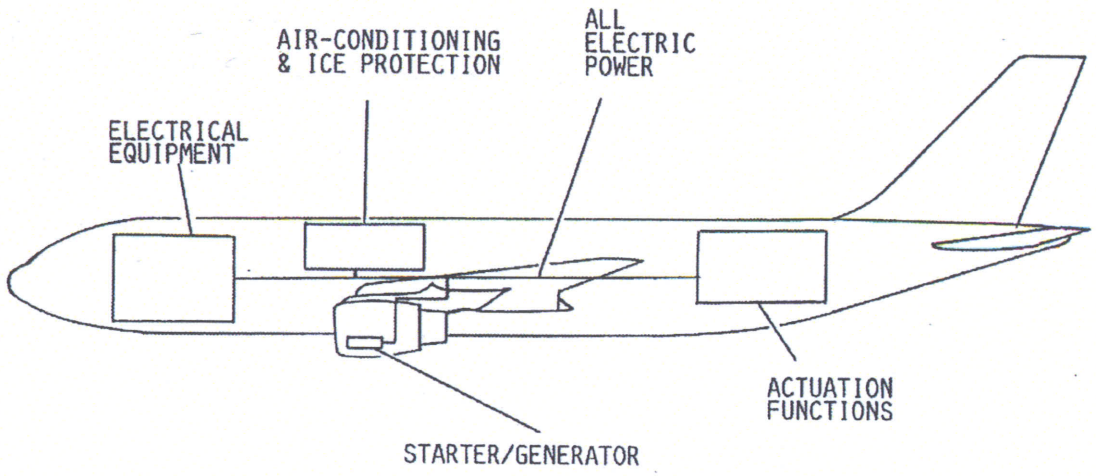


Figure 1.2 Advanced All Electric Secondary Power Systems (Reference 1)

The concept is not new and has been proposed by Woodford [2] in 1945. Cronin [3], who also supported the AEA concept for many years, claims that adoption of a single source of secondary power will improve system utilisation, thus enabling a smaller and more efficient power system to be used. The elimination of the subsystems in favour of a single system is also claimed to offer a weight reduction. Recent developments in rare earth metals for generators and motors together with electronic power control have improved the capability of the electrical power system making the AEA concept more feasible.

A second, and probably more important, reason to consider implementation of the AEA concept is the desire to improve engine efficiency and reduce fuel usage. SPS evolved during the years that fuel was relatively inexpensive and as result their design was not optimised for minimum fuel consumption. Bleeding air from engines is largely inefficient and has detrimental effects on the performance of the propulsion unit. In addition, concerns over the capability of high bypass ratio (HBPR) engines to provide the bleed air requirements of future aircraft make the AEA concept an attractive alternative that may enable the introduction of such engines and maximise their potential benefits.

1.1.1 Definition of Advance Secondary Power Systems Concepts

The move towards the AEA has fundamental effects on the design of the engines, their associated systems and accessory drive gearboxes, environmental control, ice protection and the various actuation and control systems (primary and secondary flight controls and utility systems, which include landing gear, doors, brakes, steering, etc.). Several technological challenges and difficulties must be resolved and therefore these changes cannot be achieved in a single step. For this reason, the original term and concept of the AEA has 'evolved' over the years and now extends to additional definitions such as the More Electric Aircraft and More/All Electric Engine. Although there is no strict definition of any differences between these terms and while all of them are frequently used interchangeably, for the purposes of this study more detailed definitions, covering their use and meaning, are given below.

All Electric Aircraft (AEA): The concept assumes the integration of all SPS into a single electric source. Present hydraulic and pneumatic actuation functions would be eliminated together with extraction of bleed air for the supply of the environmental control and ice protection systems, (Figure 1.2). More information regarding possible system architectures for an AEA are given in Chapters 6 and 7.

More Electric Aircraft (MEA): Some MEA assume that the majority of SPS would be electrically powered. However, present forms of hydraulic, mechanical or pneumatic power would be retained. Dedicated or localised hydraulic systems may be incorporated together with the extraction of bleed air.

All Electric Engine (AEE): The AEE concept would use large generators, mounted on the core of the engine, to supply power to both engine and airframe accessories. Hydraulic systems could be retained, though the pumps would be electrically driven as opposed to the present system that utilises mechanically driven pumps. In addition, the engine's oil system and accessory gearbox would be eliminated. Finally, electromagnetic bearings would replace conventional bearings. In accordance with the AEA concept engine bleed air would be eliminated.

More Electric Engine (MEE): The MEE concept would also use a large generator to supply power to the majority of engine and airframe accessories. However, the Accessories Gearbox (AGB) could be retained or modified to accept the generator itself and/or some mechanically driven accessories. In addition, the oil systems could be reserved along with conventional bearings.

1.2 Previous Studies

The AEA concept received much attention during the 1980s in the US. Three studies, by Boeing[4], Lockheed[5] and NASA[6], investigated the potential savings on short to medium range aircraft of using a single electric form of secondary power with digital technology. Although these Integrated Digital Electric Aircraft (IDEA) studies showed significant fuel savings, the findings were inconsistent and fuel savings for a range of 2000 nautical miles varied from 12% to 3%. A closer examination challenged the accuracy of the gains and revealed that the performance benefits of the AEA concept alone would not be significant. This has been confirmed by studies at Cranfield College of Aeronautics that claimed a 0.82% fuel reduction over a similar range. Despite the fact that at that time the gains of the AEA concept were not large enough to justify a change in the pattern of secondary power, there are other benefits of the AEA systems -such as reduced single type maintenance- and therefore an active interest on the AEA systems has been maintained with subsequent studies concentrating on the development of individual and enabling technologies such as advanced generation systems, power distribution and control, electric actuation, etc. More information regarding the approach and individual findings of the above-mentioned studies is given in Chapter 2.

1.3 Study Specifications and Research Method

The overall aim of the study is to assess the benefits and implications of adopting the All Electric Aircraft/Engine (AEA/E) concepts relative to conventional secondary power systems on a series of well-defined baseline aircraft/engines. The figures of merit are engine sfc, overall fuel usage and aircraft/engine mass. Changes in Turbine Entry Temperature (TET) as well as reliability and maintainability issues are also considered and studied. In addition, the study determines whether individual or combined parts of the AEA/E concepts permit and/or favor new transport configuration designs. Finally, it is the purpose of this work to identify any risks and implications associated with the implementation of the AEA/E concepts and to propose appropriate actions required to reduce these risks. In order to quantify the benefits of the AEA/E concepts application of the following subsystems and technologies on the baseline aircraft have been examined:

- Elimination of Bleed Air Systems

- Electric Anti-Icing or De-icing Systems

- Electric Environmental Control Systems

- Electric Engine Starting Systems

- Electric Power Distribution and Management Systems

- Electric Actuation

The following sets of objectives were also established:

- Develop an aircraft/engine-level simulation model that is capable of accurately calculating the performance of the aircraft and engine system for given mission specifications

- Quantify fuel and mass changes on aircraft level

- Develop, examine and optimise a number of feasible electric architectures for use with the AEA/E concepts

- Develop a method for assessing and optimising individual and collective benefits associated with the application of the AEA/E concepts on a range of civil aircraft configurations

- Identify areas of risk requiring further investigation

1.3.1 Selection of Baseline Aircraft and Cases Examined

The present study focuses on high capacity, long-range aircraft that served as baselines and were compared to identical aircraft with the same mission specifications incorporating advanced sub-systems defined by the present study. Where possible and applicable, the designs comply with airworthiness requirements of JAR 25 and FAR 25. The cases examined include:

1. A conventional aircraft similar to A340-500 powered by four engines similar to Rolls Royce Trent 500 (Baseline)

2. Conventional aircraft similar to A330-200 and A380 powered by two engines similar to Trent 700 and four engines similar Trent 900 respectively.

To assess the benefits of the AEA/E concepts a baseline aircraft with conventional systems had to be chosen. Although several options for a baseline aircraft do exist, the final choice refers to a long-range high-capacity aircraft similar to Airbus A340-500 with Rolls Royce Trent 535 engines. The long cruise segment of such a design allows the aircraft, both in terms of aerodynamic and engine performance, to be optimised for cruise conditions and implies that even modest sfc improvements may lead to considerable fuel savings. Moreover, the large number of passengers imposes increased bleed air requirements for the operation of the ECS and therefore there is plenty of scope in eliminating conventional bleed air and pneumatic secondary power systems. As far as changes in mass are concerned, it is anticipated that potential savings, due to the replacement of hydraulics with electrical power for actuation, will be more noticeable in the case of a large aircraft.

Although there have been a number of studies on eliminating bleed air while adopting a more or all electric approach, none of them considered the case of a long-range, high-capacity aircraft. As a result, the moderate power requirements and the more important weight considerations, for the class of aircraft reviewed, led to moderate sfc savings and retrofitted designs, which in turn were considered to be inadequate to justify a change in the methods of extracting secondary power. This work shows that for a new large aircraft these metrics will change and implementation of new SPS technologies could lead to more favorable results.

With respect to the second case examined, implementation of high bypass ratio engines has been selected on the basis that this type of powerplant is more sensitive to the extraction of secondary power and in particular bleed air off-takes.

Finally, to be able and study the effects of All/More Electric concepts on the design and performance of various aircraft configurations it was decided to include two more baseline aircraft each one presenting its own characteristics and requirements with respect to mission specifications, mass, number of engines and power requirements. As it will be demonstrated, the approach allows for the individual findings to be interrelated and validated with each other and those of previous studies.

1.3.2 Overall Approach and Thesis Structure

Relevant Literature Review: The findings are presented in Chapter 2 and cover the following areas:

- Future trends and requirements in aerospace design: Since previous studies on the subject of the AEA concept concluded that the performance benefits alone would not justify a change in the pattern of SPS it was decided to review the general trends and requirements of future aircraft and determine how the All/More Electric systems fit with the overall design approach.

- Engine, aircraft and systems integration: It will be demonstrated that one of the main advantages of adopting the AEA concepts is the optimisation of off-takes and the improved engine efficiency that can be achieved.

- Systems architecture for the AEA: Previous studies and current technology levels were reviewed to identify suitable and enabling system architectures for use with the AEA/E concepts.

Development and application of non-dimensional analysis for estimating effects of off-takes on engine performance: Off-takes impose a significant penalty on the performance of aero-engines. A detailed study covering the development and application of a non-dimensional analysis for estimating the effects of secondary power on the performance of various engine configurations is also presented in Chapter 3. The analysis is combined with a study that evaluates the effects on the performance of a typical military engine of driving the afterburner pump, Chapter 4. The selection of a military engine reflects its ability to operate within a wide range of altitudes and Mach No. The study also covers particular aspects of adopting the AEA/E concepts in the case of a military aircraft.

Development of an aircraft-engine level performance model: To be able and evaluate the performance of the aircraft as a complete system an aircraft performance model was developed. Turbomatch, an existing engine performance program, was integrated as the engine module. This tool was then used to assess the performance of the baseline aircraft together with the impact of the AEA concept on the trip fuel for simplified inputs, corresponding to alternative designs. The work is presented in Chapter 5. Chapter 5 also includes the definition and analysis of the baseline aircraft and their secondary power systems.

Elimination of bleed air and hydraulic systems (review of AEE/A concepts): Inefficiencies and weaknesses of present secondary power systems are presented and discussed in Chapter 6. A review of the AEA/E concepts and the extent to which they can be implemented is also included. Alternative secondary power systems and architectures are reviewed.

Power generation and distribution systems: Electric power generation and distribution systems are reviewed in chapter 7. Different approaches and system architectures are presented and reviewed. Special consideration is given on the type of aircraft and number of engines.

Sensitivity Analysis - Discussion: Final results are presented in Chapter 8. These are followed by sensitivity analyses into the impact of sfc, aircraft mass and power off-take changes on trip fuel usage. The integration of secondary power systems is a complex multi-disciplinary task depending heavily on the type and configuration of the aircraft and engine in question. Estimates on the specifications of advance secondary system technologies are also presented for the various aircraft reviewed.

Chapter 2: Literature Review

The literature survey concentrates on three main topics: Future Trends in Aerospace Design, Engine Aircraft Integration Issues and All Electric Aircraft Studies and Technologies. Over the years concerns have been expressed over the capability of High Bypass Ratio engines and alternative advance propulsion systems to provide conventional forms of secondary power. For this reason, future trends in aerospace design and engine aircraft integration issues have also been included in the literature review and the findings are presented in this chapter. The relevance of these topics to the present study becomes evident considering the results and conclusions drawn from the literature on the subject of the AEA, which suggest that the overall effects of advanced SPS will heavily depend on the type and size of aircraft as well as number and configuration of engines. In addition, the conclusions derived from the literature review imply that although implementation of advanced secondary power systems could improve overall efficiency and reduce trip fuel it should be treated as enabling technology and not as a single solution which will revolutionize the design and performance of the aircraft/engine system. The findings of the literature survey are discussed and summarised at the end of the chapter.

2.0 Future Trends in Aerospace Design

Over the past 50 years, the aerospace industry has seen remarkable progress in the performance and fuel efficiency of civil transport aircraft. Following the introduction of the jet engine in the 1950s, significant developments in engine and airframe designs established aviation as the primary mover in the long haul passenger traffic. Implementation of technical improvements and advances in propulsion technology, aerodynamics, structures and materials, manufacturing methods, airframe systems and the adoption of avionics and computer based technologies have been the key elements for the design of more efficient and safe aircraft. Finally, with the introduction of wide body aircraft and the subsequent increase in passenger capacity, aircraft fuel burn per passenger has been reduced even more. The above technical aspects coupled with the development of advanced theoretical analysis tools, such as Computational Fluid Dynamics (CFD), and a better understanding of flow physics have enabled the design of advanced long range transport aircraft that are 70% more fuel efficient than the early jet powered aircraft [6, 10].

Despite these improvements the civil aerospace industry is frequently regarded as "mature" and evolutionary rather than revolutionary [7,11,12]. These views are supported by the fact that most of the improvements in fuel efficiency came in the first 20 years

and as result of the advances in propulsion technology [6]. Nearly two-thirds of the improvement in aircraft fuel efficiency is engine related and mainly due to higher bypass ratios, overall pressure ratios and turbine inlet temperatures [6].

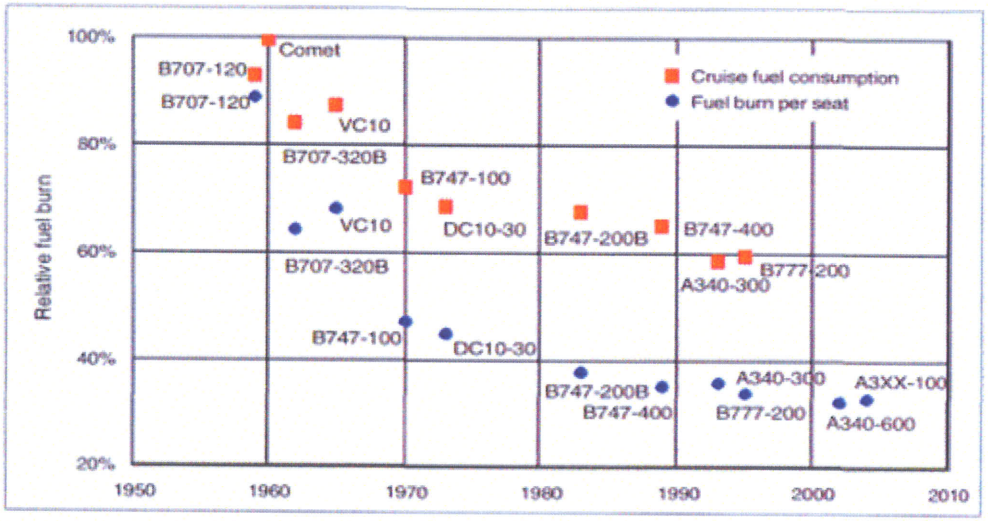


Figure 2.1: Aircraft Fuel Burn Reduction [6]

Further evidence for this perceived maturity is the fact that aircraft design has evolved incrementally over the last decades [7,12]. The general arrangement of current aircraft differ little from that of the original Comet design, with the most noticeable difference being the engine installation, which has changed from the original buried layout to the under wing podded configuration used in today's long range aircraft. Even this change has its origins in the Boeing 707 layout, which entered service only six years later than the Comet. Most of past and current design approaches have been based upon the design by disciplines, such as propulsion, structures and materials, aerodynamics, etc, rather than revolutionary and integrated design concepts [11]. Until recently, this technique provided effective solutions and therefore there was no or little scope in changing a well-known and proven method with a new and uncertain design and its associated risks.

Although the above holds true, especially considering the nature and status of civil aviation and more importantly the fuel prices, in order to be more precise one has to consider what will be the future trends in aerospace design and more importantly, the requirements of the future global aerospace market. As the improvements in both engine and airframe designs approach their limits - implementation of retrofitted designs and evolutionary technologies approach limits set by the laws of physics or practical constrains (i.e. maximum attainable TET, 80m box, etc - the prospects of entering a new era in aerospace design, which involves radical changes and a higher

level of integration, become more attractive and may be inevitable. These considerations led a number of international experts to express their concern of whether the civil aerospace industry, based upon the evolutionary and incremental development of technologies, will be in a position to meet the even more demanding requirements for air travelling during the 21st century. Many believe that this will only be achievable with the introduction of revolutionary concepts and the implementation of extensive synergies between the airframe and propulsion unit [7, 8, 9].

As many believe, today's aircraft are designed with affordability in mind. While the design process became even more complicated and time consuming, the interest has moved towards increased productivity and efforts are made to reduce costs through improved processes instead of new technology. Although fuel efficiency has still a significant economic effect, especially in the long-range sector, with the uncertainty in fuel prices and the reduction in airline yields, defined as the revenue generated per passenger mile, there is an increased pressure on improving cost of ownership as well as reliability and durability [(4)]. These elements are critical in reducing both direct maintenance costs and the indirect costs of delays and cancellations.

Given the current philosophy new technologies and innovative designs must ensure sufficient improvements in fuel efficiency along with, at least, the same standards of reliability with the existing designs if they are to be implemented. The extent of these metrics, however, is subject to the requirements of the future aviation market and what the aircraft and engine manufacturers will be expected to deliver. It seems that in the first quarter of the 21st century the aeronautics industry may see significant developments and innovations, which may result in a renaissance in aeronautics.

2.1 Future Requirements

2.1.1 Traffic Growth

Despite the recent crisis in the aerospace industry and air travelling, international experts forecast that a 5-8% annual growth in worldwide traffic demand will occur over the next 20 years [(9, 10, 11, 13, 14)]. Similar growth is also forecasted by the two main aircraft manufacturers. Much of this growth will come as a result of the continuous economic development of Latin America, Asia and Africa as well as the increase of tourism and the future, global market environment. The expansion of internet and e-commerce may also stress this trend. Although hard to predict, the ongoing evolution in telecommunication systems may lead to a business travel reduction over the next 20 years, leaving recreational and leisure travelling the dominant market sector, which in

turn is particularly price sensitive. The airline responses to this increased demand may include adding new routes as required, increase frequencies on current routes, replace smaller aircraft with larger current aircraft and/or replace current large aircraft with new large aircraft. Although adding new routes and increasing aircraft departures will be a definite response to the increased demand, the fact that traffic growth is forecasted to be greater than the departure growth, implies that the introduction of larger aircraft may be necessary.

2.1.2 System Capacity

Airport congestion and the overloaded air traffic control system, especially in Europe and Japan, limit the potential of carrying more traffic by increasing frequency. Pusch[14] states that a quarter of all flights in Europe in 1999 were delayed. Improvements in air traffic control capabilities could lead to fewer delays, reduced holding times and improved system capacity. It is also estimated that advances in electronics, navigation and telecommunication systems could improve air traffic management leading to an 8%-18% reduction in aviation fuel burn.

One issue of particular concern, however, is the separation required between aircraft. Existence of wake vortices reduce system capacity and constrain the capabilities of airports. Although the introduction of a new large aircraft, such as Airbus A380, may result in greater separations (as a consequence of the increased weight and thus vortex strength) it is anticipated that the wider separations may be offset by the increased passenger capacity of such an aircraft.

In terms of aircraft/engine design and performance, system capacity can be improved through increased take off thrust, top of climb thrust requirements, reduced aircraft weight, wing loading -defined as aircraft mass per square meter of wing area- and increased wing aspect ratio. Increased take off thrust and improved top of climb performance allows the aircraft to reach the required cruise altitude in a shorter time and distance leading to reduced block fuel and time. Such requirements, however, are likely to have a detrimental effect on the off design performance of the engine (i.e. cruise fuel consumption), maintenance requirements (i.e. increased power setting and TET) as well as the installed weight of the propulsion unit. The configuration of the engine in terms of bypass ratio and specific thrust could also be affected.

As far as aircraft design is concerned, a balance must be struck between the conflicting factors of wing loading and aspect ratio on climb performance, vortex strength, aerodynamic efficiency at cruise, physical constrains and finally aircraft mass. For

example, although reduced wing loading could reduce thrust requirements at take off and vortex strength at climb, it requires a larger wing area which has a detrimental effect on aircraft mass and drag. Increased aspect ratio, on the other hand, could improve both aerodynamic efficiency and vortex strength but would be limited by mass consideration and the 80 meters box requirement.

2.1.3 Environmental Issues

Over the past 30 years environmental concerns have led to the introduction of many regulations on aircraft noise and emissions. With the continuous growth in air traffic it is expected that the introduction of more strict regulations will be inevitable. Whilst current regulations are concerned with the landing and take-off cycle only, it is estimated that future regulations will also include emissions produced through the cruise segment.

As far as aircraft noise is concerned, even current regulations constrain the design of very large aircraft and engines. This is partly due to the fixed limit above a certain aircraft weight. In addition as aircraft size increases, airframe noise becomes critical and at approach conditions airframe noise is as important as engine noise.

2.1.4 Affordability Versus Fuel Efficiency

Affordability is one of the leading factors in today's aircraft design process. Birch [4] states that over the past forty years, yields from civil aviation have been fallen faster than unit costs. As a result there has been continuous pressure on both engine and airframe manufacturers for lower ownership and maintenance costs. The manufacturers in turn have responded with the development of product families in order to allow a high degree of commonality between the different products and to reduce initial costs.

Although, fuel efficiency is still a driving factor in aircraft economics, the relative low fuel prices coupled with the fact that ownership costs account for approximately fifty percent of the direct operating costs impose additional constrains in the development and implementation of new technologies [10]. However, since it is quite possible that a sudden increase in fuel prices may occur over the next twenty years and given the strong influence of fuel costs on airline economics, the need of increased fuel efficiency, even in the expense of ownership costs, is extremely important and may be inevitable. To ensure affordability, a balance must be struck between the conflicting factors of improved fuel efficiency and the increased research and development costs.

2.1.5 Passenger Comfort

With the increased competition between airlines passenger comfort, in the long-range sector, attracts even more attention. More space and payload, less noise and vibrations as well as better internal air quality are some of the issues that become more and more important when it comes to long air travels. As far as passenger comfort is concerned, a number of constrains do exist and must be carefully weighted in order to ensure that the trade off between aircraft weight and measures to reduce passenger fatigue are well balanced. For example, every attempt to increase the minimum cabin pressure results in a corresponding increase of fuselage weight. Moreover, as the requirements for in-flight entertainment increase along with the demand for better internal air quality, more power off-takes from the propulsion system are required, which in turn affect the performance of the engine. The latter becomes critical as the specific thrust, and therefore the core of the engine, reduces.

2.2 Requirements and Limitations of a New Aircraft

The subsonic transport market generates the highest profits for both airframe and engine manufacturers. This sector is further divided into the short, medium and long-range aircraft markets. Each one of these sectors has its own requirements, mostly related to the speed and size of the aircraft in question. Depending on the size and range of the aircraft a relationship between sfc, mass and initial costs can be determined.

It can be argued that for a short to medium range aircraft, speed and fuel efficiency are not always the dominant factors. In such a case initial costs, especially with the current fuel prices, seem to be the driving factor. As the range increases, however, the requirements tend towards better fuel efficiency, lower weight and finally acquisition costs. With fuel efficiency having a snowball effect in the design and performance of the aircraft, high acquisition costs can be, and most of the times are, balanced by reduced fuel costs, engine and airframe weights and the ability to carry more passengers for a given maximum take-off weight. As an example, a significant improvement in fuel efficiency not only has an immediate impact in fuel costs, which may count for 20-30% of the DOC, but also reduces airframe weight and size, which in turn reduces the weight and the size of the engines and so on.

Historically, aircraft prices are based on cost per pound of empty operating weight, while the price of the propulsion system is determined in large extend by its weight and thrust produced. The impact in acquisition costs due to the implementation of

advanced technologies can be balanced by reducing aircraft weight and thrust required. Therefore, as the range and the size of the aircraft increases, fuel efficiency becomes the dominant factor and may lead to a significant reduction of the DOC even if initial costs are to increase.

Taking into account the above considerations it is the high capacity long-range subsonic aircraft that will be discussed in the following sections of this work. One more factor that supports this decision, is the fact that the long cruise segment of such a design allows the aircraft, both in terms of aerodynamics and engine performance, to be optimised for the cruise conditions. This is evident in the current designs where actual cruise thrust is close to the minimum specific fuel consumption the engine can deliver, and aircraft cruise at lift to drag ratios approximately 92-95% of their maximum.

The requirements of a new large aircraft –such as Airbus A380- include a passenger capacity of more than 550 and range around 7500-8000 nm. In addition as a successor of the jumbo jet the cruise speed can be set at 0.80-0.85 M. Field performance is targeted to be as good as the current four engine aircraft.

Although, technically speaking the aircraft is feasible, there is a significant number of challenges that must be resolved in order to achieve the desired performance. As far as affordability is concerned, the airlines response is that for such an aircraft in order to replace the 747-400 a seat-mile cost advantage of 15% must be reached [16]. However according to Acton [17], British Airways set a target of 20%. Although, increasing the capacity of 747-400 to 600 passengers and applying latest state of the art to the aerodynamic design can yield a 9% reduction in DOC per seat-mile, this is 6-11% short of the airliners target, which will therefore require application of new technology and may require drag reduction targets in excess of 15% besides the application of advanced materials and high bypass ratio engines [20,21].

Another issue that needs to be addressed is the type of advanced technology that offers the largest savings. In the field of aerodynamics, for example, although a hybrid laminar flow control system appears to have the largest potential for drag reduction levels, and therefore fuel usage, required to meet the target set by the airlines, it is considered as a new and high-risk technology. Therefore, even if the DOC target is to be reduced, given the strong influence of engine sfc on fuel usage it is anticipated that improvements in propulsion technology would be more likely, and in some cases required, to occur during the following years.

In addition, considering that aero-engines take longer to develop than airframes and are often in a direct competition for one such airframe, it becomes even more important for engine manufacturers to place great emphasis on forecasting customer needs and future market developments.

2.3 Previous Advanced Engine Development Programs

The rise in fuel prices during the mid and late seventies and the predicted further escalation in price led NASA to promote and encourage research in a number of aircraft technologies aimed at improving fuel efficiency. The Aircraft Energy Efficient Program (ACEE)[22] was developed and directed part of its effort to energy conservative propulsive systems for subsonic transport aircraft. The ACEE programme was directed at modest but significant improvements in the then near term, considerable greater savings for the mid to late eighties and more ambitious advances for the nineties. The propulsion programme included research into three main areas: engine component improvements, the Energy Efficient Engine programme and the Advanced Turboprop project. The first two programmes attempted to identify improvements that could be adopted on the then turbofan engines to improve fuel efficiency.

The engine component improvements project concentrated in near term technologies that would reduce fuel consumption of the then turbofans engines, specifically the CF6 engine manufactured by General Electric and the JT8D and JT9D engines manufactured by Pratt & Whitney. The work was contracted to General Electric and Pratt & Whitney with support from transport aircraft manufacturers and airlines. The goal of the project was to achieve a 5% reduction in fuel consumption over the life of the engine. Typical improvements studied included:

- Improved fan blade design to minimise vibrations, aerodynamic losses and tip clearances

- New front mount to minimise radial force vectors and improve compressor efficiency and tip clearance

- New high pressure turbine to improve cooling, tip clearance and component efficiency

- Cabin air recirculation to reduce extraction of bleed air for the supply and operation of environmental control system

The Energy Efficient Engine programme aimed for an over than 12% improvement in sfc accompanied by at least 5% reduction in DOC along with improved performance retention over the life of the engine. Results and engine definitions studies were presented and compared against typical and advanced turbofans engines. The study indicated that both the sfc and DOC targets could be achieved with increase in bypass ratio, engine pressure ratio, turbine inlet temperature and implementation of better materials, improved use of cooling flows, increased component efficiencies, tighter clearances and use of active clearance controls, exhaust gas mixing, etc.

The programme improved the understanding of engine technologies and the majority of the findings are incorporated on the present turbofan engines. The findings of the Energy Efficient Engine programme were also used in the early all electric studies and therefore the work undertaken in the programme is particularly relevant to this study and its significance is further considered in the following paragraphs of this chapter.

The third ACEE propulsion project was the advanced turboprop project that sought a 30% to 40% improvement in fuel usage. The project recognised the potential for significant fuel savings if the propulsive efficiency of turbofan engines could be improved to a similar level achieved by the propeller and therefore proposed a revolutionary change in the configuration of the engine. Propellers have powered aircraft extensively up to the mid fifties but in most cases have been replaced by turbojet and latter by turbofan engines. Although, turbojet and/or turbofan engines use more fuel than an equivalent propeller they offer a significant increase of flying speed that is particularly attractive to both passengers and aircraft operators.

The main disadvantage of propellers is the loss of efficiency due to the existence of sonic speeds at the tip of the blades. NASA Lewis Laboratories demonstrated that implementation of highly swept short broad blades could delay the occurrence of sonic speeds, thus enabling their use at high flying speeds and altitudes that are comparable with those of turbofan engines. The newly designed blades could improve propulsive efficiency from around 70-75% for turbofans to 90-95%. Such designs became known as propfans.

For the blades to operate efficiently it is necessary to rotate at speeds lower than the equivalent tip speeds of turbofans. If the blades are driven directly by the turbine, as in the case of a conventional turbofan engine, the lower operating speed required would compromise the performance and efficiency of the turbine. For this reason, use of reduction gearboxes has been proposed.

Reduction gearboxes have also been proposed for turbofans to allow increased fan diameter and a reduction in fan blade loading so that bypass ratio could increase to values above 15. Geared fan designs were given the designation of ultra high bypass ratio engines. Such engines would retain the duct that surrounds the fan improving protection in the case of a blade separation and allowing under-wing installations. The existence of the bypass dust would also improve noise attenuation. However, the growth of bypass ratio and engine mass flow results in large cowl diameter with the subsequent mass and drag penalties. Therefore, at the time it was argued that the weight of the bypass duct and engine cowl would limit the development of ultra high bypass ratio engines and the open rotor propfan configuration will predominate.

Alternative designs for matching blade and turbine rotational speed involve the use of contra-rotating, variable pitch blades driven directly by a pair of contra-rotating turbines. In such configuration the hot section does not have stators and as the turbines are contra-rotating their relative speed increases allowing efficient operation of both the turbine and the propfan blades. This concept avoids the complexity and the mass penalties of the gearbox.

Regardless of the final engine configuration, ducted or open rotor, single or contra rotating, the improvements in engine efficiency will result in a greater proportion of the gas generator work being converted to useful thrust. These improvements will therefore reduce the mass flow through the gas generator reducing specific thrust and the power recirculated within the core of the engine.

2.3.1 The Energy Efficient Engine

The E^3 engine was used by Boeing, Lockhead and NASA in their studies on advanced secondary power systems. For this reason it is appropriate to review the findings of the study. Both General Electric and Pratt and Whitney considered four turbofan configurations: direct drive and gear drive with mixed and separate exhausts. Each of the engine configurations used in the study offered similar fuel savings within 2.5%. Each engine used a 10 stage high pressure compressor with a nominal pressure ratio of 23:1. The low pressure compressor pressure ratio was adjusted depending on the engine bypass ratio and exhaust form to maintain an overall engine pressure ratio of 39:1.

The direct drive design incorporated a 7:1 bypass ratio with a fan pressure ratio of 1.74 or 1.66, depending on the exhaust form. For the geared fan the bypass ratio was increased to 9.1 with fan pressure ratio varying from 1.58 to 1.52, depending on the

type of the exhaust. Implementation of a gear drive reduced the tip speed from 470m/s on the direct drive to 370m/s and increased sfc and fuel savings by 2.5% and 1.5%, respectively. However, due to the complicated design and arrangement of gear drives Pratt and Whitney suggested a 10% weight penalty along with a 10% percent increase in first cost and 4% increase in maintenance costs for their use.

All four configurations studied in E^3 offered significant fuel savings of over 12% compared with the then current turbofans. The slight advantage offered by the geared fan, however, did not justify the extra effort required to develop and implement it.

Significant improvements in component and engine thermal efficiency were achieved by minimising losses associated with the poor aerodynamic design of the baseline engines. The engine's mount structure was also modified to give a stiffer case assembly, which when combined with the use of active clearance, achieved improvements in compressor and turbine efficiency. These improvements resulted in 5% reduction in sfc. Increase in engine pressure ratio improved thermal efficiency and led to a 6% reduction in sfc. Finally, the use of mixed exhaust contributed another 2.7% improvement. The combination of these improvements along with other minor refinements achieved a more than 15% improvement in sfc. In addition to sfc and DOC targets NASA provided targets in relation to noise, performance retention and exhaust emissions. All configurations studied by both companies met the noise, CO and deterioration targets but exceeded those set for NOx. NASA also specified the maximum cruise thrust, for sfc comparisons, at M= 0.8 and 10668m with zero bleed and power extraction. These operating conditions established the design point of the engine and consequently the running line of the compressor.

Apparently, the running line of the compressor with no off-takes was set below the regions of maximum efficiency. As a result, use of the E^3 configurations for subsequent all electric aircraft studies reported that the sensitivity to bleed off-takes was considerably more severe than in the case of shaft power extraction. It is shown in Chapter 3 that extraction of shaft power alters the position of the running line which moves upwards towards the surge line. Extraction of bleed air, on the other hand, has the opposite effect and the running line moves away from the surge line. It is for this reason that placement of the running line with no off-takes below the maximum efficiency favoured extraction of shaft power rather than bleed air. The project reports, however, do not discuss the reasons for placing the running line.

Although it is hard to comment on this decision, especially without knowing the exact conditions and reasons that led to this action, it is reasonable to assume that the

positioning of the running line was determined to large extent by the required surge margin. The compressor of the General Electric E^3 engine had a pressure rise of 23:1 in ten stages at a time when most high pressure compressors achieved around half this pressure rise. To minimise mass penalties and manufacturing costs, stage loading could increase through increased rotational speed and diffusion factor. This approach was adopted by Pratt and Whitney. Increased stage loading concentrates the efficiency islands and results in steeper non-dimensional speed lines. Both these, affect the sensitivity of the compressor and the effect of bleed and shaft power extraction.

In addition, the high pressure ratio of the compressor will give a flat operating line that may be susceptible to surge at low power settings and non-dimensional speeds. The General Electric design incorporated variable inlet guide vanes. The four stator stages were also of variable geometry and moved in accordance with the inlet guide vanes. Although this configuration could improve the surge characteristics of the compressor and extent the operating region, the running line would still need to be placed for additional surge margin. The combination of high stage loading and flat working line might have resulted in the working line being fixed below the maximum efficiency regions.

The compressor had customer bleed ports at the fifth stage and the outlet. Staerns[22] , describes a bleed air/fuel heater, implemented in the General Electric demonstrator, that improves engine efficiency by transferring heat from the bleed air system to the fuel. The system eliminates the loss of extracting fan air to pre-cool the bleed air before transferring it to the airframe. In addition, it recovers some of the energy of the bleed air, which is otherwise lost. Whilst at take-off, climb and cruise the fuel flow is enough to act a heat sink, at lower power settings (i.e. idle, descent) a proportion of the bleed air was bypasses to an air pre-cooler drawing air form the fan. The report* claims that the sfc improvement over the General Electric CF6-50C was 14.5% with zero bleed and shaft power extraction. With customer bleed and shaft power off-takes the benefit of recovering energy from the bleed air improved the installed sfc to 14.95%.

The development of the fuel heater was not identified in the original report or the preliminary E^3 design report. Therefore, it is quite possible that the use of the heater was not proposed until the development stage, towards the mid eighties, and has not been included in the subsequent all electric aircraft studies.

2.4 Future Engine Developments

Engine specific fuel consumption is a function of flight speed, thermal and propulsive efficiency and finally fuel calorific value (F_{HV}) as shown below:-

$$sfc = \frac{V_o}{\eta_{th}\eta_{prop}F_{HV}}$$

For a given flight speed and fuel calorific value, sfc can be improved as thermal efficiency and/or propulsive efficiency increase. The product of propulsive and thermal efficiencies is the overall efficiency of the engine and is a measure of how effectively the heat input "q" is converted into available thrust power Vo(Vj-Vo), hence:

$$\eta_{overall} = \frac{V_0(V_j - V_0)}{q}$$

where :
V_0 = Flight velocity (m/s)
V_j = Jet velocity (m/s)

Since the velocity and the temperature within the jet are higher than those in the free stream, energy is left in the jet stream in the form of kinetic energy and heat. While the heat left behind is a measure of thermal efficiency the kinetic energy is accounted for by the propulsive efficiency.

2.4.1 Thermal Efficiency

Thermal efficiency η_{th} measures the kinetic energy in the jet in terms of the heat input:

$$\eta_{th} = \frac{V_j^2 - V_0^2}{2q}$$

For the ideal constant pressure cycle it follows that:

$$\eta_{th} = 1 - \frac{T_1}{T_2} = 1 - \frac{1}{R^{\frac{\gamma-1}{\gamma}}}$$

Where R equals the pressure ratio of the compressor.

The ideal cycle can never be achieved because of friction, non-adiabatic conditions, pressure losses, changes in specific heat and other losses. In practise it is found that for a real cycle the thermal efficiency depends on the overall pressure ratio as well as turbine entry temperature (TET) and component efficiencies. Although it is often argued that thermal efficiency is independent of temperature and whilst this is true, it underestimates the significance of the operating turbine temperature limits and the relationship between the optimum pressure ratio and the turbine inlet temperature.

Over the last 50 years significant improvements have been achieved in all three aspects affecting thermal efficiency. What is worth noting, however, is the fact that for fixed component efficiencies, thermal efficiency increases as pressure ratio and TET rise simultaneously. This has been achieved by developing advanced materials with higher melting points and by using efficient and sophisticated cooling methods. Therefore, if the full benefits of further increasing the pressure ratio and TET are to be realised, improvements in technologies such as cooling methods and materials, must continue to advance. At a constant technology level, where cooling flows will increase with TET and pressure ratio, while component efficiencies will degrade as engine cores become smaller and loadings increase, no real sfc gains can be realised. Moreover, with the increased pressure on engine manufacturers for more reliable and long-lived engines, higher TETs are further constrained. Finally, environmental considerations may limit the maximum operating temperature as well. Overall pressure ratios are similarly constraint by compressor material, structural integrity and turbine cooling limitations, component efficiencies and weight penalties. It is for these reasons that engine manufacturers forecast modest thermal efficiency improvements of about 3-5% over the next 20 years [10].

Component efficiency levels have their own significant role in reducing fuel consumption. With the extensive use of CFD design tools, polytropic efficiencies in excess of 90% have been achieved. Although, there is further potential increase, in an attempt to achieve lower production costs, much of the ongoing research in this area has been towards the increase in component loading and the subsequent reduction of components and parts. It is expected that another 4% reduction in engine fuel consumption can be achieved over the next 20 years.

2.4.2 Propulsive Efficiency

Propulsive, or Froude, efficiency is a measure of how effectively the kinetic energy added in the jet is converted into thrust power. In the case of a fully expanded nozzle propulsive efficiency can be expressed as follows:

$$\eta_p = \frac{2}{1 + \frac{V_j}{V_0}}$$

Improved propulsive efficiency can be obtained through lower specific thrust, defined as the thrust produced per engine mass flow rate, witch in turn corresponds to increased bypass ratio and reduced fan pressure ratio. This is on the basis that propulsive efficiency increases by reducing exhaust velocities, while increasing the airflow through the engine.

For a given bypass ratio there is an optimum fan pressure ratio. The principle applies to both turbofans of the mixing as well as the separate exhaust type. As the bypass ratio increases the optimum fan pressure ratio reduces. In the case of separate exhausts it can be shown that, minimum sfc occurs when:

$$\frac{V_{BYPASS}}{V_{CORE}} = \eta_{FAN} * \eta_{LPTURBINE}$$

Since the increased turbine work, necessary to drive the larger fan, causes a reduction in the nozzle pressure ratio, this causes a fall in both the core and bypass exit velocities.

For the mixed exhaust engine, the static pressure at the mixing plane must be equal for both the air leaving the fan and the air at the outlet of the LP turbine. Therefore, for given values of overall pressure ratio, TET and thrust, the choice of fan pressure ratio fixes the total pressure downstream of the LP turbine and hence the power output form the turbine. Thus, increase in bypass ratio requires the fan pressure ratio to fall.

Assuming constant TET and overall pressure ratio, a substantial and continuous reduction in sfc can be obtained, for the ideal cycle, by reducing specific thrust. However, since component efficiencies cannot be regarded as 100%, and by considering realistic component efficiencies these benefits are significantly degraded, although still considerable.

The effect of reducing the fan pressure ratio, while increasing the diameter, requires a reduction in the rotational speed. This however has detrimental effects on the design of the turbine. If the efficiency of the LP turbine is to be maintained at adequate levels

this requires the addition of extra stages with larger blade diameter and the consequent weight penalties. If on the other hand the use of extra stages is to be avoided, the fan will run at speeds above its optimum while the turbine will run at speeds below its optimum. In addition, the higher torque required to drive the fan will compromise the structural design of the core. Disc bore diameters will increase making the mechanical design more difficult and imposing weight penalties.

Installation effects include pressure losses within the bypass duct and the fan nozzle. With the pressure ratio and exit velocity being reduced as a result of the increased bypass ratio, the aforementioned effects become stronger and reduce the sfc benefit even more. More importantly, however, installation effects include higher amounts of drag along with the impact of increased fan and nacelle weight. Especially the drag penalty may be aggravated considering the interference drag penalties incurred between the larger nacelle and the wing. It is for these reasons that increase of bypass ratio beyond a certain value may indeed impose an sfc penalty.

Over the years different concepts and designs have been developed in order to increase the acclaimed sfc benefit of increased bypass ratio while reducing its associated penalties. As far as component efficiencies are concerned, it has been suggested that the configuration of the conventional direct driven fan, which limits the bypass ratio to a value below 10, may have to change.

An alternative solution to the direct driven fan may be the introduction of a gearbox between the LP shaft and the fan. This allows the LP turbine to operate at its optimum conditions and may provide weight savings. Furthermore, the length of the nacelle may be reduced since it does not have to accommodate the large and long LP turbine of the former design. Yet, such a design introduces unavoidable gearbox losses and large amounts of heat rejection as well as additional weight penalties. Finally the use of a gearbox may lead to maintainability and reliability concerns.

Another method for matching the fan speed with the LP turbine speed is the use of a contra-rotating stator-less turbine. The configuration allows high relative blade speeds, desirable for improved performance, while providing slower absolute blade speeds. The latter is used to directly drive contra-rotating fan blades. The contra-rotating fan offers improved efficiency by reducing swirl and eliminating fan exit guide vane losses. However, since the pitch of the blades is fixed a variable nozzle area may be required to ensure adequate stability and fan surge margins. The contra-rotating aft fan engine requires complex bearing and lubrication arrangements in the hot section of the engine, while it must provide adequate sealing to prevent losses and hot air from

flowing into the fan section. Although the contra-rotating turbine may have lower efficiencies the elimination of the gearbox favours its application.

Implementation of these techniques will allow an increase in bypass ratio to a value around 12. Further reductions in specific thrust are still possible but will only benefit fuel burn if ways can be found to reduce the increased weight and drag associated with such engines. In summary, the improvements discussed above imply an sfc gain of about 10-15% in total by 2020 relative to today's best engines. However these values assume that the engine is optimised for minimum sfc and the aircraft will continue to cruise at or near the minimum sfc thrust level of the engine [27,25].

2.5 Airframe Developments

Although airframe design is determined to a large extend by the propulsion system, an attempt is made here to identify the technologies and advances that are likely to occur over the next 20 years in order to increase the efficiency of the airframe itself. In general, the aim of any airframe designer is to increase the aerodynamic efficiency of the aircraft in question, and/or reduce the weight. Therefore, the technologies and designs that are discussed in this section refer mainly to these two disciplines.

Based on the economic viability of a future large aircraft, discussed in section 2.3 and taking into account the environmental limitations forecasted, airframe developments must achieve an 15-20% reduction in fuel burn per passenger. These developments will include advances in aerodynamics as well as structures and materials. Considering the strong influence of drag reduction on fuel usage (Robert van't Riet [40] states that aerodynamic concepts that reduce drag result in fuel consumption reductions of roughly the same percentage) it is likely and more expected that significant improvements in aerodynamics must be employed.

2.5.1 Improvements in Aerodynamic Efficiency

In the case of a subsonic aircraft, the aerodynamic efficiency is expressed as the Mach number times lift-to-drag ratio. Over the last 40 years the aerodynamic efficiency has improved around 15%. Although, early jets presented good efficiency, when the wide body fuselage and high bypass ratio engines entered service, in the early 70s, there were enough aerodynamic improvements to overcome the increased drag and weight of these designs. Since the 80's, however, improvements in the efficiency became evident [9].

Improvements in the aerodynamic efficiency came as a result of the implementation of advanced wing concepts. These include supercritical airfoils, which permit flight at a given Mach number with greater airfoil thickness and reduced sweep. This in turn allows increased aspect ratio without increased stresses, and the higher aspect ratio can produce the same lift with lower induced drag. Furthermore, winglets, which are near vertical surfaces located at the wingtip, provide a reduction in induced drag, similar to a wing span increase, but with less structural impact.

As mentioned before, the application of latest state of the art to the aerodynamic design of a 747-400, i.e. A330/340 technology level, would already give an all-new design, which would improve the aerodynamic efficiency by around 15%. However, even if the 747-400 is to be scaled using A340 technology levels, significant challenges must be resolved. As the size increases, and due to the square-cube law, weights grow faster than areas, and this alone may eliminate the DOC gains of increased size. Moreover, the large moments of inertia and short tail arms of such an aircraft will lead to unfavourable trim-drag penalties. In addition, the 80m-box constraint and structural considerations lead to an aircraft with aspect ratio around 8.2 instead of 9.3 for the A340. Therefore, for the new large aircraft in order to achieve an aerodynamic efficiency at 0.85M, which is comparable to the A340 at 0.82M a trim tank and a c.g. travel to 45% of the mean aerodynamic chord must be introduced. However it should be remembered that for c.g. positions beyond 25%, the aircraft becomes statically and dynamically unstable and needs a sophisticated close loop stability augmentation system.

Further improvements are possible with a phased approach depending on the maturity and the risk of the technology. At relatively low risk a 7% reduction in drag can come as a result of improved tip devices, restoration of higher aspect ratio by utilising the maximum span limits (in the expense of wing weight), use of riblets and variable camber wings [24].

Intermediate risks are associated to Hybrid Laminar Flow Control (HLFC) on nacelles and the fin. However, these designs will require considerable development and testing before launching. Altogether this yields a drag improvement close to 10 %. Therefore, if the 18% target is to be achieved then a HLFC system on the wings is required. This can yield a further 10% reduction in drag. Although implementation of a HLFC system could save up to 3-5% in DOC, one has to consider the development costs, which will reduce the net profit to a value around 2-3% [51].

In the field of structures and materials, extensive use of Fibre Reinforced Plastics (FRP) for the primary structures could offer a 15% weight reduction in the structure of the wing. Combining the advances in aerodynamics and structures with the improvements in engine sfc over the next 20 years can give a 40-45% reduction in fuel burn, which translates to 8-10% reduction in DOC. It is thus, obvious that the initial target set by the airlines can only be achieved with extensive synergies and developments between the different disciplines of propulsion, aerodynamics and materials.

An interesting point, however, is that although the developments outlined above yield a 40 % reduction in fuel usage compared to the most advanced current designs and around 50% reduction compared with a 747, the relative low fuel prices give a 20% reduction in DOC. It is, thus, obvious that economic viability alone may not justify the introduction of a new large aircraft.

Up to this point only conventional designs have been considered. Indeed if the technologies outlined above are to be implemented, both airframe and engine design would have reached their limits. As far as airframe design is concerned a number of aircraft manufacturers are already considering alternative solutions that may result to further benefits. Out of them the most promising seems to be the Blended Wing Body (BWB) configuration .

The BWB advantage results from the aerodynamic and structural synergies between the wing and the fuselage. The fuselage not only accommodates the payload but also serves as a lifting surface. This reduces the total wetted area of the aeroplane and allows high aspect ratio to be achieved in the expense of little weight penalty. Moreover, the favourable spanwise load distribution can offset the structural inefficiencies of the non-cylindrical fuselage, leading to further savings. Further synergy is realised through buried engines that ingest the fuselage's boundary layer and thus increase propulsive efficiency. The latter coupled with the elimination of the engine's pylons reduces the weight and the drag even more. Initial estimates include 30% reduction in fuel burn along with 25% improvement in lift to drag ratio compared to the conventional configuration described above [53]. Although there are significant challenges to be resolved, the benefits are too big to be ignored.

2.6 Implications of Advanced Engines Integrated on Future Aircraft

Despite the aforementioned gains of the high bypass ratio engines in fuel efficiency, significant installation problems may well prevent the use of such designs. These problems include both physical installation as well as system integration issues.

2.6.1 Physical Integration

The large nacelles are difficult to install in conventional underwing locations, mainly due to the limited space available. These limitations include ground proximity and Foreign Object Damage (FOD) concerns as well as avoiding engine-ground contact during an adverse landing and/or failure to extend the nose landing gear. A way around these problems may include lengthening the main landing gear and/or altering the vertical position of the wing. However, these solutions impose significant weight penalties and therefore they may be regarded as inefficient.

Moreover, aerodynamic interferences between the engine components and the wing, may lead to further inefficiencies. Although it may be possible to move the engine forward, in order to minimise the aerodynamic interaction effects, this would increase the bending moment on the wing. Taking into account the weight of such an engine this issue becomes even more critical.

Overwing installations, on the other hand, may be discarded on the basis that they present fire hazards with fuel leakage. Additional safety concerns may include the possibility that in the event of a crash hot, heavy engines may be mixed with passengers and cargo. Even in the event of emergency evacuation, significant problems may exist due to the proximity of the engines to the fuselage. Moreover, noise problems with respect to passenger comfort may be introduced. Finally exhaust impingement on tail surfaces is also a concern with over-wing installations, unless a "T" tail configuration is used.

Aft body installations require major fuselage and empennage changes. Considering the size of the UHB engines the number of engines used is limited to two and therefore higher thrust levels must be achieved. Finally in the case of a conventional aircraft, it is estimated that by removing the engines from the wing introduces mass penalties in terms of both wing mass and fuselage mass. As far as the wing mass is concerned the penalties are introduced due to the elimination of the favorable bending relief and therefore the need for a heavier structure. In addition, having the engines at the rear of the fuselage requires a longer fuselage, while the cut-outs and the discontinuities in the structural members will add weight to the structure.

Considering the installation issues of high bypass ratio engines on a BWB configuration it seems that some of the constraints and limitations discussed above are removed. Since the engines can be located on the upper surface of the fuselage, ground

clearance and landing gear weight limitations on fan diameter and BPR are removed. Furthermore, the possible elimination of the pylon reduces the drag and weight penalties of such an engine. Finally if boundary layer ingestion is to be implemented, an increase in bypass ratio may be necessary. This will allow the less sensitive fan to ingest the boundary layer, reducing therefore the negative effects of such an action on the performance and stability of the core compressor.

2.6.2 Systems Integration

Today's engines are expected to provide primary as well as secondary power. While the former is used to propel the aircraft the latter is necessary for the aircraft services including air-conditioning, engine and airframe anti-icing, avionics, flight control actuation, etc. There is a concern with respect to the developments in thermal and propulsive efficiencies of aircraft gas turbine engines and whether the latter will remain capable of providing the secondary power requirements of future aircraft. The demands placed on each engine to provide these off-takes is a function of aircraft size, number of passengers and number of engines per aircraft. As the number of passengers increases, the bleed air required for air-conditioning increases proportionally. Power requirements will also increase, although not so rapidly. However, with the increased demand for in-flight entertainment an increase in electrical power requirements must be expected [54].

With HBPR engines, because the power extracted through bleed and shaft off-takes is a larger proportion of the total in the core of the engine, than for a conventional bypass engine, the engine becomes more sensitive leading to increased sfc and thrust penalties. Moreover with the increased sensitivity of HBPR engines on weight and drag, extraction of secondary power can only aggravate these effects [58,59,60] .

2.6.3 Implications of Off-takes on Advanced Engines

Considering the detrimental effects of off takes on engine performance there have been concerns about the capability of advance engines in providing the power requirements of future aircraft. Particularly, bleed requirements may impose such penalties that other options should be considered.

For a properly matched engine, with given pressure ratio, TET and thrust, the bypass ratio is uniquely related to specific thrust. Hence, the higher the bypass ratio, the smaller is the core of the engine required to produce a given thrust.

Previous studies showed that as the specific thrust of the engine reduces, constant off-takes lead to higher sfc and thrust penalties. This is on the basis that since the core of the engine becomes smaller, bleed and shaft power off-takes constitute a greater percentage of the total power within the core. Moreover, with the developments in engine and airframe design an aircraft with fixed payload and mission will become smaller and lighter, thus requiring less thrust. Since the off takes will remain essentially constant the engines will suffer a greater performance loss than for constant thrust. With the introduction of a new large aircraft, however, and the continuous demand for better air quality and in-flight entertainment, the off-take requirements will also increase leading to further performance penalties. Hence, future engines are perceived to have higher off-take penalties than today's engines because off:-

1. Reduced engine core, reflecting the increase in bypass ratio

2. Reduced engine size, reflecting the improved performance of future engines and aircraft
3. More severe off-take requirements to improve passenger comfort and to overcome the increased inertia and the relative short moment arm of a new large aircraft

The only reference obtained, which specifically addresses secondary power implications of HBPR engines, is by N J Peacock[53] of Rolls Royce, Derby. This paper concludes that there is nothing fundamental that makes the penalties of off-takes on HBPR engines greater than on present turbofans, provided they are designed for off-takes at the out-set. However, although the penalties will not increase in absolute terms, percentage increases will occur and thus improved methods of providing systems power should be sought preferably using shaft power from the output shaft. Anti-icing is recognised as a particular problem and this function should be the first to be considered in the replacement of bleed air.

Although other manufacturers have expressed their concern about the limitations of advanced engines to provide secondary power, they all agree in that, if the engine is designed with the off-takes in mind only percentage increases in sfc and thrust penalties will occur, while the absolute values will remain essentially constant. These views are also supported by a series of independent studies performed during the late eighties by Cranfield College of Aeronautics [58, 59, 60].

However since the work of Hopkins[60] and Peacock[63], are the only studies specifically addressing the issues discussed above it is worth considering in more detail some of the points they raised along with their findings.

Peacock, in his paper considered the application of high bypass ratio engines and open rotors to a medium range, 150 seat aircraft. The off takes have been sized accordingly for this class of aircraft and engine. This selection reflects the forecast during the late eighties that major engine developments will occur at the 150-seat class of aircraft. At that time large subsonic transports were not, in general, proposed neither novel configurations. Furthermore, since two engines were required to propel the aircraft considered, the one engine inoperative situation determined the maximum off-takes and therefore performance penalties.

Although the study appreciates the fact that for a properly matched engine, hence one that is designed to provide off-takes at its design point, the penalties will increase in percentages only, it estimates that the 1-2% sfc penalty for the then turbofans will rise to over 4% for advanced engines. However this increase will come as a result of the increased off-take requirements rather than engine inability to provide the off-takes.

Although, the above holds true, one has also to consider the difficulties as well as the penalties incurred even when the engine is designed with the off-takes in mind.

As far as the process of matching the engine for known off-take requirements is concerned, additional problems occur and have to be resolved. The design case for an HBPR powerplant is likely to be different from that for a conventional turbofan. For the later the design case is likely to be for take-off thrust, while for HBPR engines the design case will probably be at the top of climb. The supply of secondary power does have implications when considered along with this change in design case.

At sea level mass flow rates through the core engine of a powerplant are high, particularly at the maximum thrust setting used for take-off. Thus, although bleeding air from the engine will reduce thrust, the proportion of total flow taken for air-conditioning, is small. Mass flow rates at the top of climb will be significantly less than at take-off and although bleed rates for air-conditioning are also likely to fall, the reduction will not be that great. Therefore, in this case the proportion of flow taken as bleed is greater requiring a further 'oversizing' than would be required in a take-off case to retain thrust with off-takes if necessary [49]. It is therefore worth considering the likely off-takes to see whether they follow this pattern.

In the case of a civil transport aircraft, shaft power requirements are likely to be reasonably constant throughout the flight, apart from peaks during operation of high-lift devices and/or extension/retraction of the undercarriage. Thus, although extraction of shaft power increases as a proportion of the total shaft power recirculated within the engine, the difference is not that great.

In descent cases both bleed and shaft off-takes become a much larger proportion of the totals. Also, to provide the required pressures, bleed must be taken at a higher stage from the compressor. However, the total fuel used during descent phases is small compared to that used in climb and cruise. Provided the engine is not taken into an unstable region and that excess thrust is not produced due to faster running of the engine to provide off-takes, operating the engine in a significantly less efficient condition than in climb and cruise can be tolerated.

One airframe service not considered so far is ice protection. On present aircraft, this is provided by using bleed to heat surfaces where ice may build-up. Thermal anti-icing can use as much and, in some cases, considerably more bleed than air-conditioning. Although the flow rates for anti-icing fall with altitude, like air-conditioning, and are not required at altitudes above 30,000ft, the bleed requirement is not present at all times, only in icing conditions. These occur only a small percentage of the time.

To provide enough thrust in icing conditions would require considerable oversizing of an HBPR powerplant. Although the compressor might be allowed to run in a less efficient condition during anti-icing use, due to the small proportion of time required, compromise and oversizing of engine components would still be required and the extra weight of oversizing would be carried at all times. Thus, as pointed out by Peacock[53] anti-icing is a significant problem in its present form.

Obviously, the effects discussed above will become more severe in the case of advanced aircraft and novel configurations like the BWB. Since use of a LFC system can reduce cruise drag, and therefore thrust requirements, by 10% the engine will have to operate at a lower power setting. Hence, the increased power needed for the operation of the LFC system in conjunction with the bleed air, will become a much larger proportion of the totals within the core of the engine. Especially in the case of a BWB configuration the reduced number of engines, most designs proposed consider three engines, will increase the off-take demands applied on each engine.

2.6.4 Installation Implications of Secondary Power Systems on Future Engines

In this chapter so far only efficiency and turbo-machinery design characteristics have been considered with regard to the extraction of secondary power from future powerplants. This section considers the installation implications of the facilities required to provide secondary power on such engines.

In the case of a conventional turbofan engine, the accessory gearbox is usually placed within the nacelle. Moreover the cabin bleed air is carried across the fan stream in a duct within a large fairing downstream of the fan and outlet guide vanes.

If the full benefits of increasing engine's bypass ratio are to be realised, the nacelle should be designed to have minimum dimensions with little space for required structure and no space for other components. In addition the core cowl should also have minimum dimensions as well as the support struts between core, cowl and engine mounting.

In the case of a HBPR removal of bleed air from the core would impose significant pressure loss penalties due to the ducts passing across the wide fan stream. Since the effect of a given pressure loss in the bypass stream increases as fan pressure ratio falls, the penalties incurred from the air-condition ducts will be more severe in the case of a HBPR engine. Air starting would also produce similar problems, due to the requirement of ducts passing across the fan stream and leading to the air starter. Moreover if the AGB is to be placed within the fan cowl, the latter will become larger and heavier implying additional drag and weight penalties.

2.6.3 Engine-Airframe Integrated Performance

When considering the possible developments discussed above one has also to consider the integrated performance of the propulsion-airframe system. Although the negative effects of increased bypass ratio and engine size, on aircraft drag and weight have been outlined, these are not the only implications. Taking into account Breguet's equation[46] for range and rearranging it for the fuel as:

$$W_{Fuel} = W_{initial}\left(1 - e^{-\left(\frac{Rsfc}{V(\frac{L}{D})}\right)}\right)$$

The equation as it stands at the moment appears to take into account the effect of the propulsion system with the term of sfc. However a closer examination reveals that for a given range the engine itself affects both the aerodynamic efficiency and the initial weight. The latter can be taken as the maximum take-off weight which in turn consists of the weight of different items as follows: some of the items (i.e. wing, undercarriage, etc) can be taken as a function "C_1" of the total gross weight. Some other items such as the payload it self, the fuselage and the furnishing can be assumed to be a function "C_2" of the payload. Finally there is the installed engine weight and the weight of the fuel. This gives the following expression:

$$W_{in} = C_1 W_{in} + C_2 W_{payload} + W_{engine} + W_{fuel}$$

The weight of the wing, now, is also a function of engine weight, along with other factors, meaning that the engine affects all the weight items but the weight of the payload. As the bypass ratio increases although the sfc reduces the weight of the engine increases as well, leading to increased W_{in} and therefore increased fuel weight. Furthermore, a large bypass ratio affects to a great extent the aerodynamic efficiency of the aircraft. Hence, although improved sfc can be achieved by reducing the specific thrust of the engine, there is an optimum bypass ratio for fuel weight that represents a balance between improved sfc, weight and drag.

One particular problem with HBPR engines, compared to conventional turbofans, is that the former will have a greater thrust lapse with speed and altitude. Since HBPR engines produce thrust by reducing exhaust velocities, while increasing the airflow, at sea level the mass flow rate through the engine will be greater, reducing with altitude. This reduction in mass flow corresponds to a reduction in thrust produced and therefore it is likely that the thrust required will be determined by the maximum climb conditions. Thus, it may be necessary to oversize the engine with the consequent penalties in mass and drag. Although, lower thrust settings can be achieved through increased aspect ratio and/ or reduced wing loading, the negative effect of these measures on wing weight and the aerodynamic inefficiencies, imposed by the increased wing area during the cruise segment, offset by far any potential savings leaving again increased thrust settings as the more likely solution.

Moreover, the potential introduction of laminar flow control systems could reduce cruise drag, and therefore thrust, by 10%. Since LFC systems have no effect at takeoff, the lapse of thrust is increased even more [27]. Advances in airframe and engine technology may upset the well-defined and understood operating temperature

and pressure relationships of today's engines throughout the mission profile [17]. As a result it will be more difficult to match the engine for minimum sfc at cruise, while providing the increased thrust settings required for take-off and climb. Furthermore, the increased passenger capacity of a new large aircraft and the power needed to drive the LFC system will require higher off-takes from the engines and at lower thrust settings. These actions will aggravate the thrust and sfc penalties normally incurred.

From an environmental point of view, while current regulations are concerned with the landing and take-off cycle only, it is estimated that future regulations will include emissions produced through the cruise segment as well. The emission performance of any aircraft is a close function between the airframe, engine and combustion. The design of the airframe determines both the thrust required, due to its weight and drag, as well as the emissions per passenger. Reducing the drag of the aircraft has an immediate effect on the fuel burn and, thus, H_2O and CO_x emissions, while weight on TET and NO_x. From the engine point of view as the BPR increases, fuel consumption reduces. Although the above holds true, initial estimates show that for the cruise segment it is the low bypass ratio that produces the least NOx emissions. This is on the basis that the greater thrust lapse rate of the HBPR engine requires the engine to run at higher thrust and TET levels.

Today's large turbofan engines have a bypass ratio around 7-9. With current technology levels this represents an optimum in terms of fuel consumption, aircraft and engine weight, drag and finally cost. Moreover the turbine inlet temperature of current large turbofans is near 1700°K at take-off [45]. Again there is a balance between the maximum attainable turbine inlet temperature and its effect on engine weight (increase in turbine entry temperature increases specific power and allows for the design of a smaller core), cost and turbine blade durability. Future requirements will impose additional constraint in both bypass ratio and TET. As far as the bypass ratio is concerned a future driving factor may be noise limitations, which may require an increase in bypass ratio. TET on the other hand, may be limited by emission regulations.

It is thus obvious that future developments in aircraft design along with environmental requirements may compromise the design of the engine driving it away from its optimum for minimum cruise sfc.

2.7 Evolution of the All/More Electric Aircraft Concepts

Although the concept of an AEA was first proposed by Woodford in 1945 a more general interest began in the late 1970's when the oil crisis and the subsequent rise in fuel prices was the driver for a number of studies sponsored by NASA. In June 1981 NASA held a Workshop on Electric Flight Systems to determine their required actions and identify critical technologies.

During the mid eighties NASA contracted work with Lockheed and Boeing to research and produce study reports for an "Integrated Digital/Electrical Aircraft" (IDEA). Both companies used derivative models of their products, Lockheed an advanced 1011 (Tristar) and Boeing a derivative 767.

The Lockheed study [5] indicated 12% savings in fuel block and 7.9% reduction in DOC over a 2500 nm mission. Much of these savings came from the introduction of digital flight controls allowing relaxed static stability with significant reductions in mass and drag. However, this is not an advantage of the AEA concept. This alone makes comparison difficult and therefore inadequate. Additional savings came from improved engine performance -due to elimination of bleed air- and weight reductions. In both cases the results seem to be overoptimistic. In the case of engine performance, the baseline aircraft used bleed air form the High Pressure (HP) port and therefore the penalties incurred for the baseline configuration were higher than normally.

The study used three electrically driven centrifugal compressors with inlet guide vanes. In accordance with other all electric studies the pressure ratio of the compressor was set to 3.2 at cruise conditions. Motor speed and inlet guide vanes were used to control mass flow and keep it constant throughout the operating envelop. Induction motors were used to drive the compressors at approximately 46,500 rpm. The motors were powered by a three-phase 400V 800 Hz supply, which the study has established as the new standard to replace the conventional 200V 400 Hz supply. The induction motors were directly powered from the ac system, so weight, size, cost and thermal management problems were avoided. The estimated weight of the motor and compressor was 100 lb and 75 lb respectively giving a total weight of 525 lb per aircraft.

The conventional air cycle was replaced by some form of vapour cycle pack. The system was selected because of its improved efficiency over the conventional configuration. The study claimed that elimination of bleed air ducting, bleed pre-coolers and engine ducting would enable the total ECS weight to be reduced by 2334

lb. However, especially with the use of a vapour cycle, it is difficult to imagine this weight saving. In the contrary it is more likely that an increase in total weight would occur. Moreover, reliability, cost and maintenance issues of the more efficient vapour cycle have been poorly treated.

As mentioned above the use of relaxed stability did not enable a direct comparison between the use of electrical and hydraulic powered controls. However, given that both systems are driven from the engines as shaft power they have the same effects on the performance and cycle of the engine with the only difference being the actual amount of power extracted. However initial indications are that the difference would be minimal.

The weight and engine performance savings predicted by the use of electrical power enabled a reduction in the fuel burn. However the then high fuel prices exaggerated the savings in DOC. Finally the assumption that the new technology would cost the same per pound of equipment led to more savings.

Boeing[4] considered the use of electrically driven compressors driving a pair of air-cycle packs. High efficiency samarium cobalt electric motors drove three separate centrifugal compressors. The intake was placed at the nose of the aircraft to maximise the ram recovery pressure. An efficiency of 80% was assumed for the compressor. The use of a two-stage compressor with variable diffuser and inlet guide vanes enabled operation at high efficiencies and reduced surge bleed losses. The power required to drive the compressors varied from 350 horsepower for sea level hot day operation to 270 horsepower at 39000 ft ISA conditions. The modifications of the conditioning systems showed a 1080 lb weight increase over the baseline aircraft. In common with other studies the APU was deleted form the baseline aircraft providing an overall 230 lb saving. However such a decision may be unacceptable especially for the type of aircraft examined. The study claimed a 3% improvement in fuel burn and 1.8% DOC savings for the adoption of the electrical system technology. Boeing claimed that of the 3%, about 2% was due to the all-electric system and about 1% due to replacement of bleed air extraction with shaft power extraction. In fact costs analysis results suggest that ECS changes account for 80% of the total fuel savings, i.e. 2.4%. These reductions however came from redesigning the aircraft to allow for reduced fuel and masses of the IDEA version. Moreover, the engine selected by Boeing had low shaft power and high bleed air extraction penalties compared to the then turbofans. Finally, fresh airflow rates per passenger were half those normally used.

During the late eighties Cranfield College of Aeronautics produced its own study based on a hypothetical RB211-524D4 powered A300-600 aircraft[58, 59]. Two zero bleed system variants have been derived and total mission fuel burn results compared with the baseline. A version with electric motor driven compressors as an air source was predicted to give slightly greater savings than a version with compressors directly driven by shaft from the engine. However, in both cases savings were less than 1% for the representative missions examined. Both zero bleed versions were defined to have Electro-Impulse de-icing that led to significant savings over the baseline aircraft during flight phases with icing conditions. Overall mission fuel savings, however, were small due to the limited fraction of time in icing conditions.

Further studies in College of Aeronautics included the implications of advanced engines for the airframe systems of transport aircraft[60]. For the purpose of the study a hypothetical twin Allison 501-M80F-3 propfan powered BAe 146-300 was used as a baseline. Again elimination of bleed air led to modest improvements in fuel usage. The study concludes that although, extraction of equivalent bleed and shaft power led to almost similar penalties, with shaft power being slightly better, increase in TET and loss in thrust was substantially larger for the bleed off-takes.

Although the study accounted for the weight implications of replacing current bleed systems with shaft power off-takes and the thrust loss in the case of retaining bleed air off takes, it failed to account for the effects of resizing the engine to recover the thrust loss. Moreover since the engine used was a propfan, installation effects of secondary power systems on a HBPR have not been studied. Therefore increase in engine weight and drag has not been accounted for.

Chapter 3: Effects of Bleed & Power Extraction on Gas Turbine Performance

The chapter deals with the impact of secondary power on the performance of aero-gas turbines. Effects of off-takes with variations in engine power setting, altitude and Mach No are considered, general conclusions are drawn and the findings are compared with those of previous studies. The chapter concludes with a theoretical analysis on the effects of off-takes and the development of a semi-empirical non-dimensional power factor, which can be used with various engine configurations for estimating the effects of extracting shaft power off-takes.

3.0 Introduction

Extraction and distribution of secondary power affects the design and performance of the engine and may compromise its primary role. Any system mounted on the engine in order to extract and convert power would increase both complexity and weight. The former would affect the reliability of the system and therefore increase maintenance and operational costs. The latter would affect the aerodynamic characteristics of the aircraft and would increase both profile drag, due to an increase in the frontal area of the engine, and induced drag due to the greater lift needed to counteract the increase in weight. Thus, for a given installed thrust fuel consumption will increase.

In addition to these direct effects and from a technical point of view extraction of bleed air and shaft power affects a significant number of engine parameters like turbine entry temperature (TET), sfc and thrust. Furthermore, the operation of the compressor and the handling qualities of the engine are also affected.

Initially, extraction of bleed air or shaft power would appear to be very similar. In both cases the high pressure (HP) turbine supplies the off-take, even though bleed air is indirectly supplied by the compressor. The effect on engine performance will be a reduction in the power available for thrust. Hence, both actions tend to increase turbine engine temperature and/or reduce thrust. Despite their common effects, however, component characteristics and the magnitude of the aforementioned effects will vary considerably and therefore they should be treated individually.

3.1 Background – Generalised Effects of Off-takes on Engine Performance

Compressor characteristics: For all but very low power settings, the HP turbine will operate between choked nozzles. Therefore, by definition the non-dimensional mass flow at the inlet and exit will be constant. This also fixes the pressure and temperature

ratio across the turbine. When shaft power is extracted the turbine will have to produce a higher level of work to cover the higher load. If engine speed is to be maintained the fuel control system will increase fuel flow and thus TET. To satisfy the choked flow condition the increase in TET will cause pressure to increase. The running point will move along the constant speed line towards surge, Figure 3.1 [56]. In practice, there is some reduction in mass flow but this has little effect in offsetting the increase in temperature. Taking bleed air from the compressor also requires an increase in fuel flow and hence temperature. Since the compatibility of work between the compressor and turbine must be satisfied, the change in mass flow through the turbine will require an increase in TET. To ensure the correct value of turbine non-dimensional mass flow, the compressor pressure ratio will fall, Figure 3.1.

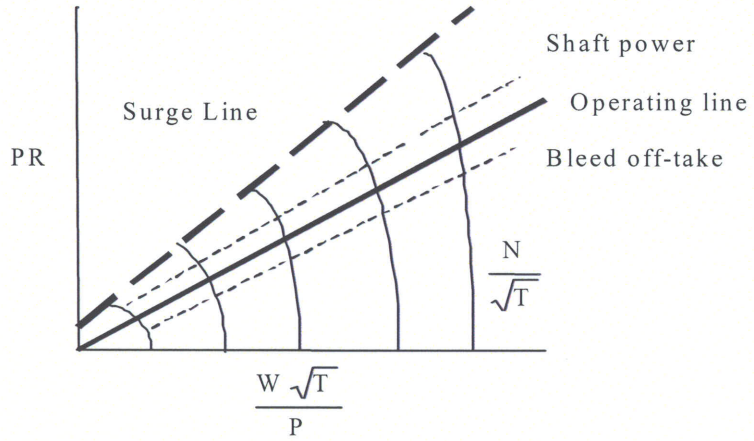


Figure 3.1: Effect of bleed air and shaft power on compressor characteristics

While extraction of bleed air increases the surge margin, shaft power off-takes have the opposite effect. Ideally the running line of the compressor will be positioned for maximum efficiency. Movement of the running line as bleed or shaft power is extracted will thus result in a change of component efficiency. Although the running line can be positioned for maximum efficiency with off-takes, the need to ensure adequate surge margins may well prevent this. In addition if the demand of off-takes varies across the flight mission, which is more likely, positioning of the running line for maximum efficiency is harder to achieve.

Effect on TET: Taking bleed air from the outlet of a compressor causes the delivery pressure and mass flow to drop by approximately the same percentage as the air off-take. Assuming constant TET and turbine pressure ratio, i.e. turbine operates between chocked nozzles, the work output of the turbine drops immediately in proportion to the percentage of the air off-take. If turbine power is to remain constant an increase in

TET is required. This action will also determine the operating point of the compressor re-establishing a new mass flow and pressure ratio, as explained above. When shaft power is extracted the turbine has to deliver extra work at a given speed and this can only occur by increasing TET.

The general effects of bleed and shaft power off-takes can be seen in Figure 3.2 where TET curves are shown over an extremely wide HP shaft speed range for constant flight conditions. It is interesting to note that the minimum TET, which occurs with no off-takes at idle speed, rises with increasing off-takes and occurs at higher speeds. However for a given TET both the trust produced and the running speed reduce. The latter implies that in terms of TET the aerodynamic speed limit will increase with off-takes. Figure 3.2 was taken from reference 56 which claims that for a compressor with small pressure ratio extraction of constant bleed air constitutes a larger percentage of off-take than for a compressor with a high pressure ratio. Since the power output of the turbine will drop in proportion to the percentage of the off-take, air bleeds tend to increase the TET in the case of high pressure compressors and drop this temperature in the case of low pressure compressors. However, this is highly unlikely and as it is shown later in this chapter (paragraph 3.4) TET penalties always increase regardless the pressure ratio of the compressor. The penalties increase in proportion to the percentage of off-take relative to the total power within the core of the engine.

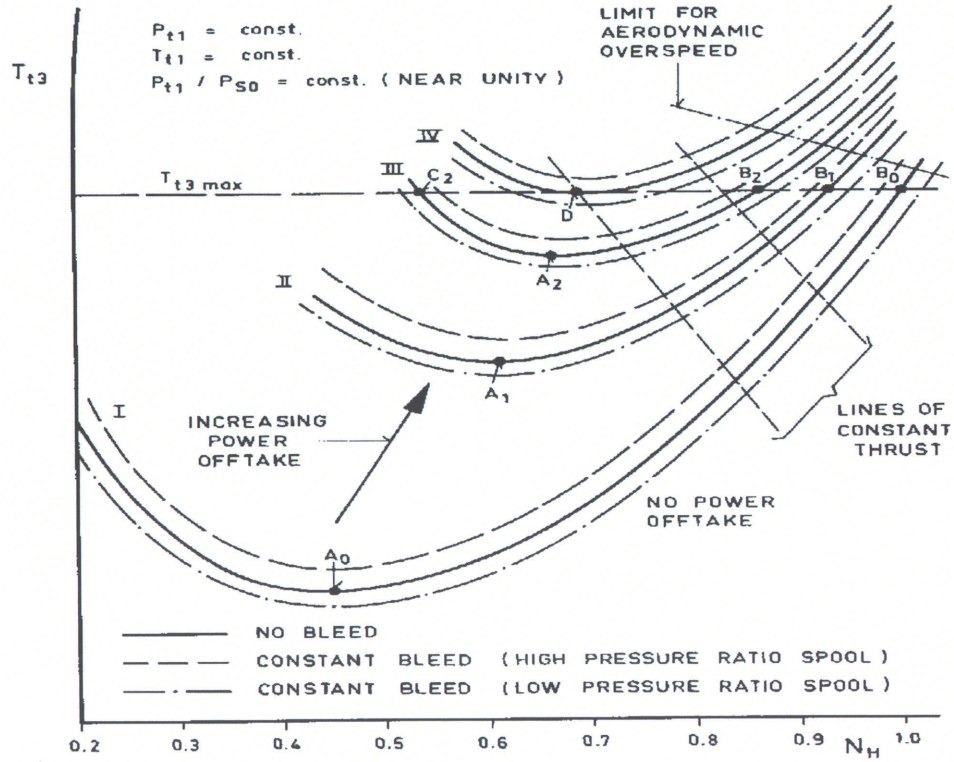


Figure 3.2 General effects of off-takes on TET

Effect on sfc: For the engine to maintain a specified thrust output and also provide secondary power (whether bleed air or shaft power) requires extra energy input. This is provided by burning fuel at higher rate. Thus, sfc increases. The increase in sfc will be a function of the size of the off-take relative to the total power delivered by the turbine and the characteristics of the engine, i.e. overall pressure ratio, specific thrust, TET, etc. Thus for a given off-take the effect will depend on the size of the engine, its configuration and finally its operating conditions. For small off-takes the increase in sfc will vary linearly with off-takes. However, for large off-takes the increase may become non-linear.

In addition to the direct effects discussed above, secondary effects take place and are mainly related to the efficiency of the compressor. Since extraction of secondary power affects the running line of the compressor its efficiency is also affected causing a rise in sfc. Similarly, reductions in other components efficiencies may also occur, though their impact on engine performance is usually less important.

Effect on Net Thrust: When the engine is operating well away from maximum output, whether this is defined by a limit on TET or aerodynamic speed limit, it is possible to maintain constant thrust as well as provide the off-takes. However, if the engine is operating at rated conditions, such as maximum take-off, maximum climb, maximum cruise thrust etc. the need to produce extra power for off-takes will reduce the power available for thrust.

Handling: When the engine accelerates, work has to be done against the shaft inertia, to increase speed and to also increase the temperature of the various elements. This is exactly analogous to taking a shaft off-take and causes the same upward motion of the running line towards surge. Deceleration, on the other hand, has the opposite effect. When designing the engine the running line will be positioned to allow sufficient surge margin for acceleration. With a shaft power off-take, however, the surge margin is reduced with the subsequent danger of surge. Although taking bleed can control this effect, this action restricts the thrust output and the efficiency of the engine.

Windmilling: The windmilling characteristics of an engine are important because they affect the capability to provide power during an engine failure and the ability to relight the engine. The effects concerned are complex but it can be noted that for a given power off-take at constant flight conditions and fixed take-off thrust, the effects are worse for:

- Higher bypass ratio engines
- Lower pressure ratio compressors
- When a compressor has others in-front of it

3.2 Effects of Off-takes under Specific Modes of Engine Operation

For a given engine the effects of off-takes can be studied under specific modes of operation, i.e. by varying alternatively engine power setting, altitude and finally flying Mach number. These variations would change both the power of the engine and its thermodynamic details. Thus, extraction of a fixed percentage of off-take will diminish the effect of off-take/engine size while isolating the effect, if any, of engine parameters such as compressor pressure ratio, TET, specific thrust, etc. Extraction of a fixed amount of off-take, on the other hand, can account for engine size and eventually the percentage of the off-take relative to the core power.

The objective of such a parametric analysis is to provide generalised conclusions and form the basis of a more in-depth and theoretical approach. For this kind of parametric analysis two types of engines have been selected, a low bypass ratio two-spool engine and a two-spool turbojet engine. The selection reflects the ability of the engines to operate satisfactory over a wide range of Mach numbers and altitudes.

3.2.1 Effect of Shaft Rotational Speed and TET

For a properly matched engine operating at constant flight conditions and with no off-takes shaft rotational speed rises with TET. As the rotational speed increases both the mass flow and the non-dimensional mass flow rise. This action also increases the pressure and temperature ratio of the compressor. In terms of thrust, the rise in mass flow combined with the increased pressure ratio and TET lead to a rapid increase of net thrust. Form the above follows that specific thrust increases as well.

Although fuel flow rises with TET there is no mass flow effect on sfc because its effects on thrust and fuel flow cancel out. However, the rise of TET and pressure ratio will improve thermal efficiency. Although propulsive efficiency falls, sfc will experience a generally dropping tendency. The opposing effects of propulsive and thermal efficiencies cancel out as TET continues to increase resulting in increased sfc.

Performance penalties of a Turbomatch model operating with 0.6 kg/s of bleed air from the end of the high-pressure compressor are shown in Figures 3.3 for sfc and thrust. The model represents on a typical military low-bypass ratio engine operating at

typical cruise conditions. It becomes apparent that for a fixed amount of bleed air the greater performance penalties are obtained at the lower TET operations. This increase in engine performance penalties, which can be attributed to the reduction of engine speed, mass flow and therefore total power re-circulated within the engine, found to be common for both shaft power and bleed air off-takes. Similar trends have also been observed in the case of the jet engine model.

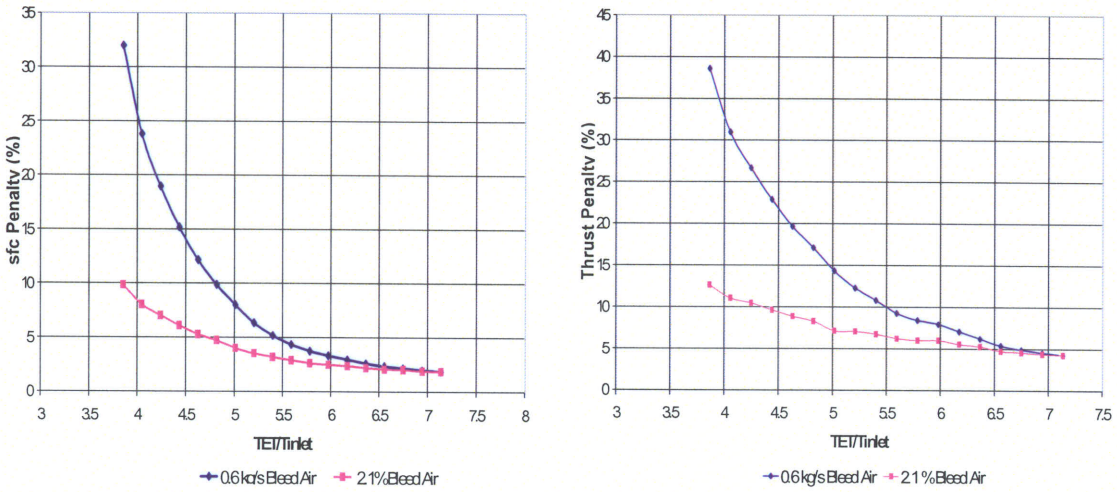


Figure 3.4 Effect of TET on Bleed Air Penalties

Performance penalties imposed by the extraction of a constant percentage of bleed air (Figures 3.4) suggest that although the size of an off-take is the dominant factor, engine parameters such as TET, pressure ratio and specific thrust have also an important second order effect.

3.2.2 Effect of Altitude

The ambient static temperature falls linearly with altitude. Ambient static pressure and density also fall with altitude. With the continuous reduction in pressure and temperature, the mass flow reduces and this is the dominant effect on thrust and engine power. For a given Mach No the total temperature at the inlet of the compressor is proportional to the ambient static temperature. Therefore, the non-dimensional rotational speed and mass flow rise with altitude. Consequently, the engine behaves as its rotational speed rises giving increased pressure ratio. This in turn partly offsets the effect of reduced mass flow. The effect is experienced up to 11.5 Km and from then on the inlet temperature remains constant so that specific thrust no longer improves with increase in altitude, thus leading to continuous thrust reduction.

The rise in non-dimensional mass flow and rotational speed, experienced in the troposphere, increases the pressure and temperature ratio of the compressor causing an increase in thermal efficiency. Although the propulsive efficiency of the engine falls, the former effect predominates and the sfc falls; remaining constant for altitudes higher than 11.5 Km. The fuel flow, on the other hand, reduces continuously with altitude. Once again, the dominant effect in compressor work is the reduction in mass flow and as a result the work produced by the turbine falls with altitude.

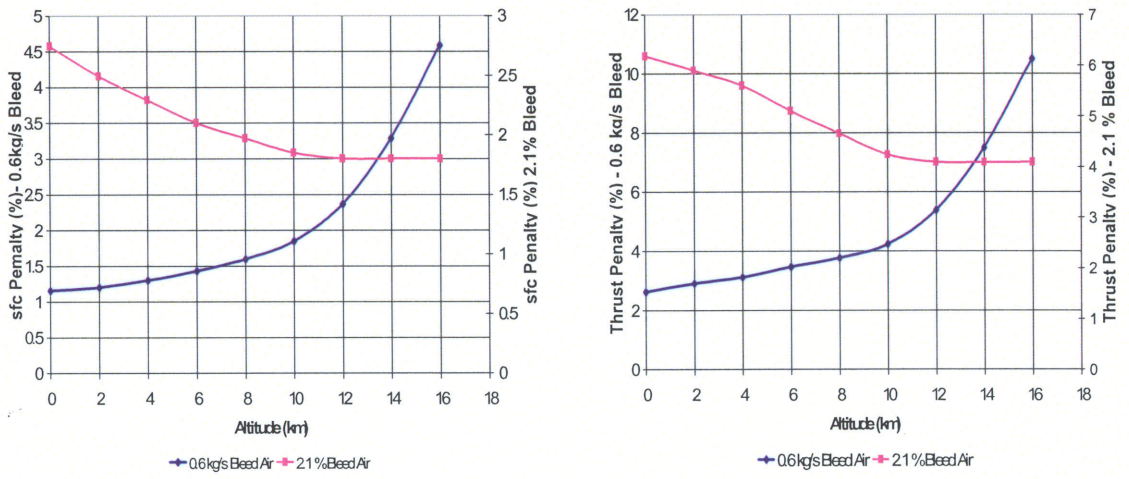


Figure 3.5 Effect of Altitude on Bleed Air Penalties

The effect of altitude on the low bypass ratio engine model operating at rated TET, 0.9 Mach and 0.6 kg/s of bleed air is shown in Figure 3.5. These plots indicate that for a fixed amount of bleed air engine performance penalties increase with altitude. However, these penalties are the net effect of two main counteracting factors: increased TET to engine inlet temperature ratio and increased percentage of bleed air. Once again the size of the off-take relative to the power of the engine is the predominant factor leading to higher performance penalties as altitude increases. Yet, it can be seen that up to an altitude of 11.5 km the effect of increased TET to engine inlet temperature ratio (hence, increased pressure ratio, specific thrust and thermal efficiency) partly offsets the effect of rising percentage of bleed air. This conclusion is also supported by the performance of the engine under constant percentage of bleed air.

3.2.3 Effect of Flight Velocity

As the flight velocity of an aircraft changes, the performance of the engine is affected. Three main effects are taking place with increase in speed:-

- Momentum drag
- Ram compression
- Ram temperature rise

Momentum drag increases with Mach number and reduces the net thrust of the engine. Ram compression, on the other hand, has a double effect. It increases the pressure at the inlet and therefore flow density and mass flow into the engine rise. The second effect is of increasing nozzle pressure ratio and gross thrust. In terms of net thrust, ram compression and momentum drag are the primary effects. At low speeds, up to Mach numbers of 0.3-0.5, the effects of ram compression are relatively small and momentum drag predominates leading to reduced thrust. As Mach No increases, ram compression becomes more significant leading to increased thrust. In the case of a turbofan engine, however, the momentum drag effect is more pronounced due to the lower jet velocity and the larger inlet area.

In terms of non-dimensional performance, ram compression and temperature rise will result in reduced non-dimensional shaft speed, mass flow, pressure ratio and therefore specific thrust and thermal efficiency. In accordance with its natural tendency the sfc curve rises as Mach number increases. As far as compressor work is concerned, although pressure ratio reduces, ram compression and temperature rise are the dominant factors leading indeed to increased compressor work.

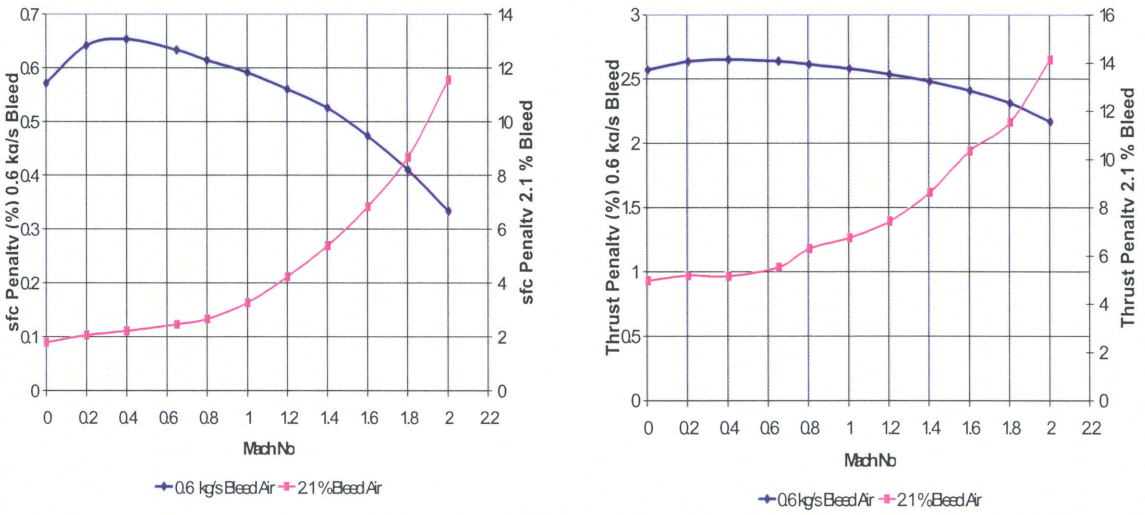


Figure 3.6: Effect of Mach No on Bleed Air Penalties

The effect of flight Mach No on the performance of the low bypass ratio model operating at rated TET and with the above-mentioned off-takes is shown Figure 3.6. As previously discussed, increase of Mach No leads to more air pumped into the engine

reducing the proportion of the off-take relative to the amount of air through the core. Therefore, performance penalties for fixed off-takes reduce with Mach number. However in the case of constant percentage of bleed air the detrimental effect of compressor inlet temperature and pressure on pressure ratio, non-dimensional power setting and specific thrust implies increased performance penalties with Mach number.

In general, performance penalties for the jet engine model were in good agreement with the results presented in the case of the low bypass ratio model. However, this was not the case with Mach number variation. More specifically, extraction of a fixed amount of off-take led to essentially constant percentage penalties, Figure 3.17. It is shown in paragraph 3.4 that this finding represents the net effect of reduced off-take size relative to the power re-circulated within the engine and reduced non-dimensional power setting expressed as the ratio of TET/T_2. These two counteracting factors balance out leading to constant percentage penalties. This effect is attributed to the design point of the engine and its relatively low value of pressure ratio and TET.

3.3 Review of Previous Studies

3.3.1 AGARD's Report on Extraction of Auxiliary Power

The work described by Bauerfeind [56] investigates the sensitivity to bleed and power off-takes for a range of engine configurations. A parametric study is presented and deals with the effects of non-dimensional power and air off-takes from the high-pressure spool for nine different engine cycles. For the purpose of the study the detrimental effects of the two types of off-takes were presented with reference to TET and compressor non-dimensional speed. This choice was made because TET together with aerodynamic over-speed and idle conditions present fixed limits on the operation of the engine, while additional compressor surge margin can always be provided. The engine configurations and assumptions made were arbitrary and in some cases the pressure ratio of the high-pressure spool was modest by current standards. However, the majority of the cycles investigated were realistic and relate mostly to the then military turbojet and turbofan engines. The levels of power and bleed off-takes were given as non-dimensional parameters related to the gross thrust of the engine at take off conditions as follows:

$$\frac{\left[\frac{Power}{P_1\sqrt{T_1}}\right]}{F_{TO}} \quad and \quad \frac{\left[\frac{M_{Bleed}}{P_1\sqrt{T_1}}\right]}{F_{TO}}$$

Where:
Fto = design thrust
M_{Bleed} = bleed flow rate
P_1, T_1 = inlet pressure and temperature

The analysis did not consider changes of compressor or turbine efficiency with the displacement of the running line caused by bleed or power extraction. All the performance investigations were carried out at operating conditions where the propulsion nozzles were always choked.

The study showed that the sensitivity of the various engine types to air and power off-takes differed substantially. The actual physical limits for the off-takes were represented by the maximum permissible TET, aerodynamic over-speed and the flight idling speed in this order of importance with the additional surge margins required following as a result of these off-takes. It is interesting to note that the maximum power off-take, limited by maximum attainable TET, was relatively low in the case of the two high bypass ratio engines examined. Although the author of the study did not comment on this finding, it is reasonable to assume that because of the high bypass ratio the power extracted was larger relative to the total power re-circulated within the core of the engine, and the high-pressure spool in particular. Of the nine engines modelled none of them experienced problems due to the potential surge of the high-pressure compressor. In the contrary, the work showed the low-pressure compressor to be particularly sensitive leading to premature surge before the engine reaching limiting values of TET, high-pressure surge, etc. The latter effect was more noticeable in the case of the high bypass ratio engines and also the configurations that involved a moderate pressure ratio for the high-pressure spool. Once again, the study failed to comment on the findings.

The findings are somewhat surprising and in most cases it would be the high-pressure shaft that would require additional surge margin as a result of shaft power extraction. Without knowing the exact operating conditions and component characteristics used, it is very hard to comment or speculate on the reasons that may led to these findings. However, it should be noted that the amounts of off-takes considered by Bauerfeind were extremely high and did occur at relatively high altitude - low Mach number operations. In such conditions the low design pressure ratio for the fan will give a steep working line making it more susceptible to surge at high values of non-dimensional speed. As explained before, the effect of a given off-take on the performance of a compressor will depend upon the size of the off-take relative to compressor power and also the characteristics of the compressor. In practice, high-

pressure compressors tend to surge at low values of non-dimensional speed and pressure ratio.

Hence, it is more likely that the findings reflect on the sensitivity of the low-pressure shaft to operate safely at conditions of high non-dimensional speed, which is also exaggerated by the extraction of extremely large amounts of secondary power, and should not necessarily be presented as the critical case for the provision of extra surge margin with extraction of shaft power off-takes. This conclusion is also supported by the findings presented in the next chapter, which deals with the effects on the performance of a typical military engine of driving an afterburner pump.

From an engine point of view, extraction of shaft power will be limited by the reduction in surge margin. At high values of non-dimensional speeds, i.e. high altitude-low speed operations, it is likely that the low-pressure compressor would impose this limit. At low values of non-dimensional speed, i.e. low altitude-high speed operations, the HP compressor would be prone to potential surge. However the above assumes that the maximum TET is reached and therefore there is a thrust penalty to be paid. In fact this will become the critical case form an aircraft point of view. In such a case, the reduction in thrust will constrain the net performance and the flying qualities of the aircraft limiting its operational envelope. Again, such a condition is likely to occur at high levels of off-takes during high altitude low speed operations, Figure 3.8 .

Although, Bauerfeid [56] identified compressor aerodynamic over-speed as a limit, this is highly unlikely. The effect of an off-take at constant TET would be a reduction in rotational speed leading to increased over-speed limits in terms of TET (Figure 3.2 taken from Bauerfeid[56]). Therefore, it is more likely that aerodynamic over-speed would limit the performance of a clean engine, i.e. no off-takes, while in the case of off-takes maximum TET would be once again the limiting factor. In the extreme case of very high off-takes the rotational speed at maximum attainable TET may become the minimum self-sustained speed preventing further increase of off-takes. The latter is shown in Figure 3.2 curve No. IV.

Bauerfeind[56] also suggests that by relating the bleed and power off-takes to engine size and inlet conditions a direct comparison can be made of the different sensitivities of the various cycles to air and power off-takes. Although neither of the above-mentioned parameters is actually non-dimensional, they both take into account the size of the engine and its operating conditions in terms of altitude and Mach number. However, the graphical representation of the results show that the parameters fail to

account for different engine configurations, i.e. bypass ratio and number of spools, as well as the power setting of the engine.

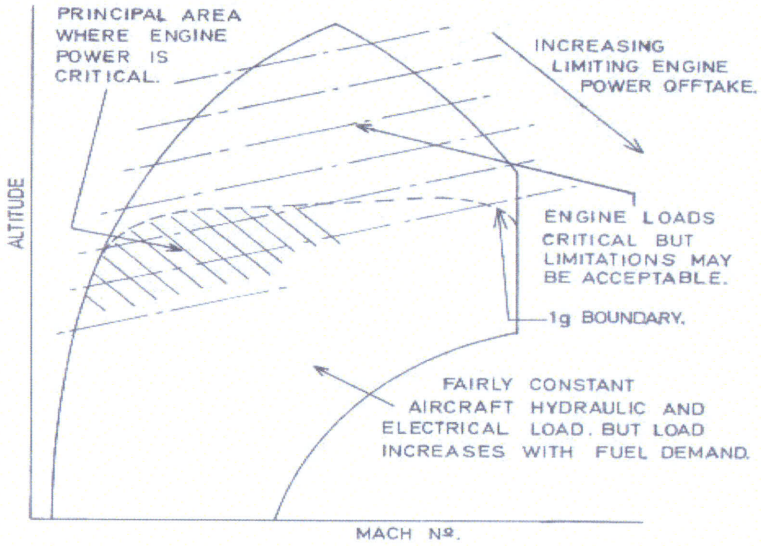


Figure 3.8: Effect of Off-takes on Aircraft Operational Characteristics [61]

To conclude, the size of shaft power off-takes considered by Bauerfeind[56] was extremely high and untypical of transport aircraft secondary power requirements. Even in the case of military aircraft some of the off-takes did occur at unrealistic conditions and therefore, cannot be considered as representative. However, the study is indicative of the trends for secondary power extractions and its limitations with respect to reduction in surge margin, increased TET, flight idle speed to avoid engine run down and finally rotor over-speed. The analysis, however, did not considered changes of sfc with extraction of secondary power. Although these variations will have a second order effect in the case of military engine, the criteria for acceptable performance for a transport aircraft would need to include these variations and the sensitivity of sfc to off-takes.

3.3.2 NACA's Reports on the Effects of Heat and Power Extraction

During the early fifties NACA Lewis laboratory participated in a program evaluating various means of extracting secondary power form jet engines. As part of this program the effects of heat and power extraction on turbojet engine performance was also investigated. The analysis and the findings were published in a series of technical notes[96-99], which included two main phases:

- The method of analysis and the development of generalised performance charts for a turbojet engine with extraction of bleed air, shaft power and finally heat.

- The effect of the above mentioned off-takes over a range of engine operations.

The study was based on experimental component characteristics derived from an analysis of unavailable data. These data were established from altitude wind tunnel investigations of a complete turbojet engine with several tail – pipe nozzles. Component characteristics were idealised into a set of curves for several power settings, ram pressure ratios and altitudes. Compressor and turbine characteristics were plotted as functions that contained the variables mass flow (W), enthalpy change (ΔH) and rotational speed (N) so that the turbine and compressor characteristics could be directly related through the following relationships:-

$$W_{Compressor}\Delta H = W_{Turbine}\Delta H \text{ and } W_{Turbine} = (1 + FAR)W_{Compressor}$$

Thus, a matching procedure was established and involved the simultaneous evaluation of compressor and turbine variables such as mass flow, pressure and temperature ratio and rotational speed. Both compressor and turbine characteristics were plotted on the same coordinates to establish the operating points of the clean turbojet. Extraction of bleed air and shaft power altered these variables and therefore the performance of the engine. By expressing the power off-take in a convenient manner, i.e. ratio of bleed air to mass flow and shaft power to compressor power at sea level rated conditions; changes in the matching of the components and in mass flow were evaluated.

Comparisons between the results of the analytical investigation and experimental results showed that reasonably accurate predictions could be achieved. The results presented are applicable to engines with component characteristics, pressure ratios and TET similar to the particular engine examined. Indeed, the main drawback of the method is the fact that component characteristics must be known in order to be able to apply the derived methodology or make use of the generalised performance charts.

The study also examined the effects of bleed and shaft power off-takes under various engine operations. In general, the findings are in good agreement with those presented in the previous paragraphs of this chapter. Thus, reductions in the TET and/or TET to compressor inlet temperature ratio at given flight conditions increased the adverse effects of bleed and power extraction. As far as changes in altitude are concerned the study concluded that for a fixed percentage of off-take the thrust ratio and sfc ratio approached unity as altitude increased. The findings also showed that the detrimental effects of fixed off-takes increased with altitude. This effect was attributed to the reduced density of the air with increasing altitude. Finally, for the range of Mach numbers investigated, the detrimental effects of off-takes on thrust and sfc increased

slightly as the Mach number increased. It must be noted, however, that the range of Mach number investigated was rather small.

In general, this series of technical notes presents an interesting method for evaluating performance penalties associated with extraction of secondary power. However, the application of the method is limited due to the fact that engine component characteristics must be known prior to any investigation. Although the method can be applied to any turbojet engine in the case of a turbofan engine allowances must be made for the matching procedure of the fan. This should involve changes in bypass and fan pressure ratio. Inevitably, this action will increase the complexity of the method. Even in the case of a turbojet engine, the method requires the estimation of a significant number of parameters and becomes rather complex, confusing and time consuming. One of the main drawbacks of the analysis is that no attempt was made to investigate the effect of engine configuration and parameters on performance penalties. Finally the assumptions made and the nature of the analysis limit the application of the performance charts derived to turbojet engines with given component characteristics and engine parameters.

3.3.3 Study on the Implications of Advanced Engines for the Airframe Systems of Transport Aircraft

Godner[74] for his thesis investigated the implications of advanced engines for the airframe systems of transport aircraft. A Turbomatch two-spool turbofan model was used to investigate the effects of bleed air and power extraction. Bleed air was taken from the interstage position of the high-pressure compressor while shaft power was extracted from the high-pressure spool. The engine design point was fixed with a bleed of 1.75 lb/s for a sea level static thrust of 100 KN and TET of 1450 K. The off design performance was simulated for various values of bleed and shaft power at typical cruise conditions of 35,000 ft and 0.85 Mach. The work was repeated at four different overall pressure ratios so that the range of cycle pressure ratio of 25:1, 30:1, 35:1 and 40:1 was considered. In addition, a range of TET covering 1250 K to 1650 K for a 35:1 pressure ratio was considered. Finally, the bypass ratio of the engine was increased up to 30:1 with corresponding reductions in the fan pressure ratio. Only design point cases were studied to ensure that the compressor map did not suffer from the steep loss of efficiency.

Extraction of bleed air and shaft power in excess of the design values led to a linear increase of sfc, TET and thrust loss. Increased engine pressure ratio led to a small reduction of thrust loss and TET rise. The difference, however, was small and Godner

concluded that the increase of TET and thrust loss as secondary power is extracted from the engine is not dependent on the cycle pressure ratio. Extraction of bleed air, however, led to increased sfc penalties with engine overall pressure ratio. Godner argued that the actual fuel burnt to support the extra power requirements should remain constant for each of the engine pressure ratios, with some small difference arising from the thermodynamic advantage of burning fuel at high pressure. Thus, the apparent difference in sfc was attributed to the superior performance of the clean engine as the pressure ratio and thermal efficiency increased.

The additional fuel flow required to produce the equivalent bleed and shaft power was the same. However, the increase in TET and loss of thrust was significantly higher in the case of bleed air extraction. Godner also stated that the majority of the arguments presented in the case of increased engine pressure ratio apply equally when the TET increases with constant pressure ratio. In general, increased TET gave rise to small percentage penalties for sfc, thrust loss and TET rise required to recover the thrust loss. Finally, the study concluded that off-take penalties increased with bypass ratio. These penalties however were very small relative to the improved performance achieved with increased bypass ratio.

Notwithstanding the reasoning given by Godner the results suggest that for fixed off-takes and engine size, performance penalties are not particularly sensitive to the pressure ratio and TET of the cycle. On the other hand, increased bypass ratio has a detrimental effect on the performance of the engine with power extraction. The findings of the parametric analysis presented in paragraph 3.2 showed that for fixed off-takes and engine size, performance penalties depend mainly on the size of the off-take relative to the power of the high pressure spool. However, simultaneous increase of TET, PR and specific thrust had an important second order effect

At the design point of the engine and for fixed thrust and TET, increase of pressure ratio would produce a so-called fishhook curve when plotted against specific thrust. At first, as the pressure ratio of the cycle increases so does specific thrust. Continuous increase of pressure ratio, however, will eventually result in reduced specific thrust. For given thrust, bypass ratio and TET specific thrust is a direct measure of core power. Thus, for fixed off-takes any change in pressure ratio will have a double effect on core power and will affect the percentage of off-take relative to the total power re-circulated within the high-pressure spool. This is also true in the case where TET is increased along with specific thrust. This action would reduce the mass flow through the core of the engine and consequently the power re-circulated within the high-pressure shaft. Once again the dominant factor should be the amount of off-take

relative to the power of the high-pressure shaft with the increase of TET and pressure ratio having a second order effect. Although it is very hard to isolate the individual effects of the above-mentioned parameters, based on the findings of the following paragraph it can be argued that the results presented in reference 74 represent the net effect of two counteracting factors (i.e. effect of pressure ratio and core power in one case and TET and core power in the second), which may have been underestimated or overlooked by Godner. In the case of increased bypass ratio, performance penalties should be attributed to the reduction of core power and the extraction of a higher percentage of secondary power.

3.3.4 SAE Fuel Penalty Evaluation Method

Recognising the detrimental effects of extracting secondary power, the SAE Thermodynamic manual[80] includes a section on the evaluation of the penalties. The method of analysis includes the installation and operating penalties for the air-conditioning and engine bleed air anti-icing systems. The penalties are expressed as a reduction in range for fixed aircraft weight and an increase in take off weight for a specified range. The evaluation of bleed air and shaft power penalties are given as an extra fuel flow required to maintain the same thrust (bleed air only) and extra take off weight (both bleed and/or shaft power). The require increase in fuel flow required to maintain the same thrust with bleed air is given as :

$$\Delta W_f = 0.0335[TET/2000]W_b$$

Where
ΔW_f = Fuel flow
Wb = Bleed air flow (lb/hr)

The quality of the results is limited as different engine configurations and levels of bypass flow are not considered.

3.4 Theoretical Analysis on the Effects of Off-Takes

3.4.1 Introduction

The importance of understanding and optimising the detrimental effects of extracting bleed and shaft power off-takes is widely recognised. However, no such work has been found in published literature that addresses both in a qualitative and quantitative

manner the exact implications of extracting secondary power in terms of engine performance penalties.

For these reasons and the purpose of the study, a more detailed analysis was undertaken and concentrated on the effects of engine overall configuration and performance penalties associated with extraction of secondary power. The aim of the analysis was to isolate the individual effects of engine parameters and to develop a general method for estimating the performance penalties of different off-takes on various engine configurations. Two different approaches were possible and both of them are presented and discussed in turn in the following paragraphs. The first method deals with the effects of fixed percentages of off-takes on thrust loss, sfc, TET and compressor characteristics. The method is derived by expressing the performance of the engine and its matching procedure in a non-dimensional form. The second approach is based on a simplified form of the pi-theorem and examines the effects of different magnitudes of shaft power off-takes on various engine configurations.

3.4.2 Analysis

The case of a simple turbojet engine operating at static conditions is considered. Figure 3.8 shows a simplified diagram of the engine along with its station numbering.

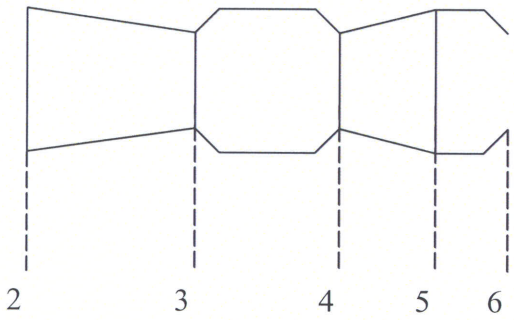


Figure 3.8 Single Spool Turbojet Engine

The engine operates at steady state conditions. Therefore, at any power setting compressor power equals turbine power:

$$W_{Air} * Cp_{cold} * (T_3 - T_2) = W_{Gas} * Cp_{hot} * (T_4 - T_5) \qquad (3.1)$$

and

$$W_{Gas} = W_{Air} * \left(1 + \frac{W_{Fuel}}{W_{Air}}\right)$$

where:
W_{Air} = Air mass flow through the compressor (kg/s)
Cp = Specific heat for air and gas (J/kgK)
η_C = Compressor isentropic efficiency
η_T = Turbine isentropic efficiency
W_{Gas} = Gas mass flow through turbine (kg/s)
W_{Fuel} = Fuel mass flow (kg/s)
T= Stagnation temperature (Kelvin)

Assuming that fuel to air ratio is small compared to the total mass flow then (3.1) becomes:

$$Cp_{cold} * (T_3 - T_2) = Cp_{hot} * (T_4 - T_5) \tag{3.2}$$

Consider the same engine operating with any given amount of bleed air extracted from the end of the compressor: Then power equation (3.1) becomes:

$$W_{AIR_B} * Cp_{cold} * (T_3 - T_2)_B = W_{Gas_B} * Cp_{hot} * (T_4 - T_5)_B$$

or,

$$W_{AIR_B} * Cp_{cold} * (T_3 - T_2)_B = (W_{AIR_B} - W_{BLEED}) * Cp_{hot} * (T_4 - T_5)_B \tag{3.3}$$

where suffix "B" denotes the case of bleed air extraction. Let W_{BLEED} be the bleed flow extracted so that:

$$\beta = \frac{W_{BLEED}}{W_{AIR}} \tag{3.4}$$

then equation (3.3) becomes:

$$Cp_{cold} * (T_3 - T_2) = (1 - \beta) * Cp_{hot} * (T_4 - T_5)_B \tag{3.5}$$

The degraded performance of the engine operating with bleed air can be generalised with reference to the clean engine by dividing (3.5) with (3.2). This yields:

$$\frac{(T_3 - T_2)_B}{(T_3 - T_2)} = \frac{(1 - \beta) * (T_4 - T_5)_B}{(T_4 - T_5)} \tag{3.6}$$

The modified power equation of each engine shown in equation (3.6) can also be expressed in terms of engine non-dimensional power setting T_4/T_2, so that:

$$\frac{\left(\frac{T_3}{T_2}-1\right)_B}{\left(\frac{T_3}{T_2}-1\right)} = \frac{(1-\beta)*T_{4B}\left(1-\frac{T_5}{T_4}\right)_B}{T_{4B}\left(1-\frac{T_5}{T_4}\right)} \tag{3.7}$$

For the purpose of this study turbine characteristics can be generalised into a single curve relating non-dimensional mass flow at the inlet and turbine pressure ratio as shown in figure 3.9a and curve 1. The non-dimensional mass flow at the exit of the turbine can then be related to the inlet conditions as follows:

$$\frac{W_5\sqrt{T_5}}{P_5} = \frac{W_4\sqrt{T_4}}{P_4}\left(\frac{P_4}{P_5}\right)^{1-\frac{\eta(\gamma-1)}{2\gamma}} \tag{3.8}$$

Therefore, for constant turbine polytropic efficiency the assumed generalised turbine characteristics give rise to a single curve for the outlet conditions when plotted against turbine pressure ratio, Figure 3.9a curve 2. The generalised characteristics of the propulsive nozzle, on the other hand, are shown in Figure 3.9b. The matching procedure of the turbine and exhaust nozzle dictates that each point of the exhaust nozzle characteristic in figure 3.9b corresponds to a point on the turbine curve 2 on figure 3.9a. For a choked exhaust nozzle the operating point of the turbine is fixed with constant pressure and temperature ratio.

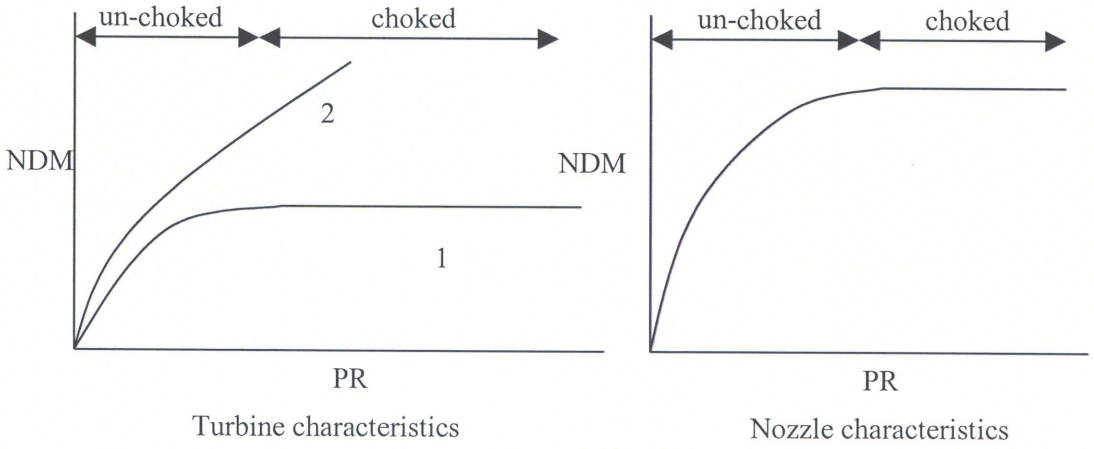


Figure 3.9: Generalised Turbine and Nozzle Characteristics

Therefore:

$$\left(1-\frac{T_5}{T_4}\right) = \left(1-\frac{T_5}{T_4}\right)_B \tag{3.9}$$

hence, for constant TET and choked propulsive nozzle equation (3.7) is reduced to:

$$\frac{\left(\frac{T_3}{T_2}-1\right)_B}{\left(\frac{T_3}{T_2}-1\right)}=(1-\beta) \tag{3.10}$$

Therefore, the pressure ratio of the engine with bleed air extraction can be estimated as follows:

$$PR_B=\left[\frac{\eta_{CB}}{\eta_C}*(1-\beta)*\left(PR^{\frac{\gamma-1}{\gamma}}-1\right)+1\right]^{\frac{\gamma}{\gamma-1}} \tag{3.11}$$

where:
PR: clean engine pressure ratio
η_C: compressor efficiency of clean engine
η_{CB}: compressor efficiency with bleed air

The non-dimensional mass flow through the compressor can be matched to the non-dimensional mass flow at the inlet of the turbine through the following equation:

$$\left(\frac{W_2\sqrt{T_2}}{P_2}\right)_B=\left(\frac{W_4\sqrt{T_4}}{P_4}*\frac{P_4}{P_3}*\frac{P_3}{P_2}*\frac{W_2}{W_4}*\sqrt{\frac{T_2}{T_4}}\right)_B \tag{3.12}$$

Assuming fuel to air ratio is relatively small compared to total the air mass flow, equation (3.12) equals to:

$$\left(\frac{W_2\sqrt{T_2}}{P_2}\right)_B=\left(\frac{W_4\sqrt{T_4}}{P_4}*\frac{P_4}{P_3}*PR*\frac{1}{(1-\beta)}*\sqrt{\frac{T_2}{T_4}}\right)_B \tag{3.13}$$

If combustion pressure losses are constant and the engine operates with the exhaust nozzle chocked, the non-dimensional mass flow at the inlet of the turbine is constant and equation (3.13) can be written with reference to the clean engine to give:

$$\frac{W_{2B}}{W_2}=\frac{PR_B}{PR}*\frac{1}{(1-\beta)} \tag{3.14}$$

Provided that the mass flow and pressure ratio of the clean engine are known, equations (11) and (14) can be combined to give the mass flow into the engine for any given bleed air off-take:

$$W_{2B} = \frac{\left[\frac{\eta_{CB}}{\eta_C} * (1-\beta) * \left(PR^{\frac{\gamma-1}{\gamma}} - 1 \right) + 1 \right]^{\frac{\gamma}{\gamma-1}}}{PR} * \frac{1}{(1-\beta)} * W_2 \qquad (3.15)$$

Equations (3.11) and (3.15) can also be used to establish the operating point of the compressor. For given flight conditions and fuel calorific value, sfc is a measure of engine overall efficiency. Specific fuel consumption is expressed as follows:

$$sfc = \frac{Fuelflow}{Netthrust} \text{ (kg/Ns)} \qquad (3.16)$$

Once again, the aim of the analysis is to express sfc as a group of non-dimensional parameters. In the case of the clean engine operating at static conditions, where net thrust equals gross thrust, sfc can be expressed with reference to non dimensional mass flow at the inlet of the compressor, fuel to air ratio and finally non-dimensional gross thrust:

$$\frac{\left(FAR * \frac{W_{AIR}\sqrt{T_2}}{P_2} \right)}{\left(\frac{F_G}{A_6 * p_6} \right)} \qquad (3.17)$$

where:
FAR: Fuel to air ratio
F_G: Gross Thrust (N)
A_6: Nozzle area (m^2)

By definition fuel to air ratio is non-dimensional and can be expressed as follows:

$$FAR = \frac{C_P(T_4 - T_3)}{\eta_{COMBUSTION} * FHV} \qquad (3.18)$$

where:
C_P: Specific heat at constant pressure
$\eta_{COMBUSTION}$: Combustion efficiency

FHV: Fuel heating value

For an engine operating with chocked nozzle gross thrust equals:

$$F_G = W_6 * V_6 + A_6 * (p_6 - p_0) \tag{3.19}$$

Therefore,

$$\frac{F_G}{A_6 p_0} = \frac{W_6 \sqrt{T_6}}{A_6 P_6} * \frac{P_6}{p_0} * \frac{V_6}{\sqrt{T_6}} + \frac{p_6}{p_0} - 1 \tag{3.20}$$

or

$$\frac{F_G}{A_6 p_0} = \frac{P_6}{p_0} * \left(\frac{W_6 \sqrt{T_6}}{A_6 P_6} * \frac{V_6}{\sqrt{T_6}} + \frac{p_6}{P_6} \right) - 1 \tag{3.21}$$

equation (19) also gives:

$$F_G + p_0 A_6 = W_6 V_6 + p_6 A_6 \Rightarrow \frac{F_G + p_0 A_6}{P_2 A_6} = \frac{\frac{F_G}{A_6 p_0} + 1}{\frac{P_2}{p_0}} \tag{3.22}$$

To simplify the analysis and the representation of the results, equations (3.21) and (3.22) can be combined to give a non-dimensional parameter ϕ that is used to express gross thrust as a function of engine inlet conditions, flight speed and finally the Mach number at the throat of the nozzle:

$$\phi = \frac{\frac{F_G}{A_6 p_0} + 1}{\frac{P_2}{p_0}} = \frac{P_6}{P_2} * \left(\frac{W_6 \sqrt{T_6}}{A_6 P_6} * \frac{V_6}{\sqrt{T_6}} + \frac{p_6}{P_6} \right) \tag{3.23}$$

Therefore, the sfc ratio of the engine operating with bleed air to the clean engine can be expressed as:

$$\frac{sfc}{sfc}_{BLEED} = \frac{\left(\frac{(T_4 - T_3) * \frac{W_2\sqrt{T_2}}{P_2} * (1-\beta)}{\left(\frac{P_6}{P_2}\left(\frac{W_6\sqrt{T_6}}{A_6 P_6} * \frac{V_6}{T_6} + \frac{p_6}{P_6} \right) \right)} \right)_{BLEED}}{\left(\frac{(T_4 - T_3) * \frac{W_2\sqrt{T_2}}{P_2}}{\left(\frac{P_6}{P_2}\left(\frac{W_6\sqrt{T_6}}{A_6 P_6} * \frac{V_6}{\sqrt{T_6}} + \frac{p_6}{P_6} \right) \right)} \right)_{CLEAN}} \tag{3.24}$$

The pressure ratio over the engine can be written as follows:

$$\frac{P_N}{P_2} = \frac{P_N}{P_5} * \frac{P_5}{P_4} * \frac{P_4}{P_3} * \frac{P_3}{P_2} \tag{3.25}$$

For any given engine operating at fixed flight conditions with choked exhaust nozzle, constant combustion pressure losses and fixed nozzle area, both gross thrust and non-dimensional factor ϕ are proportional to the pressure ratio of the compressor. Compressor pressure ratio will fix the ratio of P_n/P_2, equation (3.25).

In addition,

$$\frac{W_6\sqrt{T_6}}{P_6 A_6}, \frac{V_6}{\sqrt{T_6}}, \frac{p_6}{P_6}$$

are all constant so that, when the pressure thrust is relatively small compared to the total gross thrust, changes in gross thrust can be expressed as:

$$\frac{F_{GB}}{F_G} = \frac{\phi_B}{\phi} = \frac{PR_B}{PR} \tag{3.26}$$

equation (3.24) simplifies to:

$$\frac{sfc_{BLEED}}{sfc_{CLEAN}} = \frac{\frac{(T_4 - T_3)_B \left(\frac{W_2\sqrt{T_2}}{P_2} \right)_B (1-\beta)}{PR_B}}{\frac{(T_4 - T_3)\frac{W_2\sqrt{T_2}}{P_2}}{PR}} \tag{3.27}$$

For any value of β equations (3.27) and (3.14) give:

$$\frac{sfc_{BLEED}}{sfc_{CLEAN}} = \frac{(T_4 - T_3)_B}{(T_4 - T_3)} \tag{3.28}$$

However:

$$T_3 = T_2 \left(1 + \frac{\left(\frac{P_3}{P_2}\right)^{\frac{\gamma-1}{\gamma}} - 1}{\eta_C} \right) \tag{3.29}$$

equations (3.11), (3.28) and (3.29) give:

$$\frac{sfc_B}{sfc} = \frac{\left(T_4 - T_2 * \left(1 + \frac{(1-\beta) * \left(PR^{\frac{\gamma-1}{\gamma}} - 1 \right)}{\eta_c} \right) \right)}{\left(T_4 - T_2 * \left(1 + \frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{\eta_c} \right) \right)} \tag{3.30}$$

Because of the nature of the analysis, the derived relationship presented in (3.30) is valid for off-design conditions and any turbojet engine operating at static conditions with chocked exhaust nozzle. Therefore it has been proven that for fixed percentage of bleed air, sfc penalties depend on TET, temperature at the inlet to the engine and finally compressor pressure ratio. Gross thrust penalties, however, depend on pressure ratio only. To take into account the effect of pressure thrust and flight velocities greater than zero, equations (3.11) and (3.15) can be used to calculate compressor pressure ratio and engine mass flow. Then equations (3.28) or (3.18) and (3.29) can be applied to calculate FAR. Finally, net thrust can be calculated from equations (3.15), (3.26), (3.23) and equation (3.31) shown below.

$$F_N = F_G - M_2 * V_0 \tag{3.31}$$

In the case of shaft power extraction, the power off-take can be expressed as a function of compressor power so that:

$$\beta = \frac{Power}{W_2 * Cp * (T_3 - T_2)} \tag{3.32}$$

equation (3.5) becomes:

$$Cp_{COLD} * (T_3 - T_2) * (1 + \beta) = Cp_{HOT} * (T_4 - T_3) \tag{3.33}$$

Therefore a similar analysis as the one presented above will give:

$$PR_B = \left[\frac{\eta_{CB}}{\eta_C} * \frac{1}{(1+\beta)} * \left(PR^{\frac{\gamma-1}{\gamma}} - 1 \right) + 1 \right]^{\frac{\gamma}{\gamma-1}} \tag{3.34}$$

for the engine mass flow, assuming FAR is relatively small compared to the total air mass flow equations (3.14) and (3.15) become:

$$\frac{W_{2B}}{W_2} = \frac{PR_B}{PR} \quad \text{and} \quad W_{2B} = \frac{\left[\frac{\eta_{CB}}{\eta_C} * \frac{1}{(1+\beta)} * \left(PR^{\frac{\gamma-1}{\gamma}} - 1 \right) + 1 \right]^{\frac{\gamma}{\gamma-1}}}{PR} * W_2 \tag{3.35), (3.36}$$

the sfc ratio can then be calculated as follows:

$$\frac{sfc_B}{sfc} = \frac{\left(T_4 - T_2 * \left(1 + \frac{\frac{1}{(1+\beta)} * \left(PR^{\frac{\gamma-1}{\gamma}} - 1 \right)}{\eta_c} \right) \right)}{\left(T_4 - T_2 * \left(1 + \frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{\eta_C} \right) \right)} \tag{3.37}$$

as in the case of bleed air extraction, the gross thrust ratio can be calculated from:

$$\frac{F_{GB}}{F_G} = \frac{\phi_B}{\phi} = \frac{PR_B}{PR} \tag{3.38}$$

The above analysis considered the effects of bleed air and shaft power off-takes on the performance of a turbojet engine operating with constant TET and choked exhaust nozzle. If the increase in TET is required a similar analysis can be followed. In this case, the objective is to establish a relationship for the increase in TET required to recover the thrust loss shown above. The new operating point for the compressor will also be established together with the sfc of the engine.

Assuming the effect of pressure thrust is small, the ratio of gross thrust for the two engines can be expressed in terms of the ratio of the so called *"gross thrust parameter"* ϕ. If the thrust lost is to be recovered, this ratio must equal unity and this can only happen if the pressure ratio of the engine for the two cases is the same. Because of the choked nozzle conditions, it follows that the non-dimensional mass flow and non-dimensional jet velocity at the nozzle - the latter is shown in equation (3.21) as $V_N/T_N^{0.5}$ - remain constant. Therefore, an increase in compressor pressure ratio corresponds to increased mass flow, TET, jet velocity and static pressure at the throat of the nozzle. In the case of bleed air, the increase in TET can be established with reference to equation (3.6). Assuming constant component efficiencies and neglecting the effect of fuel on mass flow, it can be shown that:

$$\varphi = \frac{T_{4B}}{T_4} = \frac{1}{(1-\beta)} \tag{3.39}$$

and for the mass flow into the engine:

$$\frac{W_{2B}}{W_2} = \frac{1}{(1-\beta)} * \frac{1}{\sqrt{\varphi}} \tag{3.40}$$

substituting (3.39) and (3.40) in (3.24) yields:

$$\frac{sfc_B}{sfc} = \frac{\left(\frac{T_4}{(1-\beta)} - T_2\left(1+\left(\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right)\right)}{\left(T_4 - T_3\left(1+\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right) * \frac{1}{\sqrt{1-\beta}}} \tag{3.41}$$

It can be shown that in the case of shaft power extraction (3.39), (3.40) become:

$$\varphi_{Power} = \frac{T_{4B}}{T_4} = 1 + \beta \ , \quad \frac{W_{2Power}}{W_2} = \frac{1}{\sqrt{\varphi}} \qquad (3.42), (3.43)$$

and the sfc ratio yields:

$$\frac{sfc_B}{sfc} = \frac{\left(\left(T_4 * (1+\beta) \right) - T_2 \left(1 + \left(\frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{\eta_C} \right) \right) \right)}{\left(T_4 - T_2 \left(1 + \frac{PR^{\frac{\gamma-1}{\gamma}} - 1}{\eta_C} \right) \right) * \left(\sqrt{1+\beta} \right)} \qquad (3.44)$$

The appearance of the squared root *power factors* (1+β) and (1- β) for shaft power and bleed air respectively, is attributed to the fact that as the TET increases the detrimental effects of the off-take on sfc diminish.

The above-derived relationships for sfc and TET penalties are also valid for bypass ratio engines. Therefore, for fixed percentages of off-takes, relative to the power re-circulated within the high-pressure compressor or the core of the engine, performance penalties are not dependent on the bypass ratio of the engine.

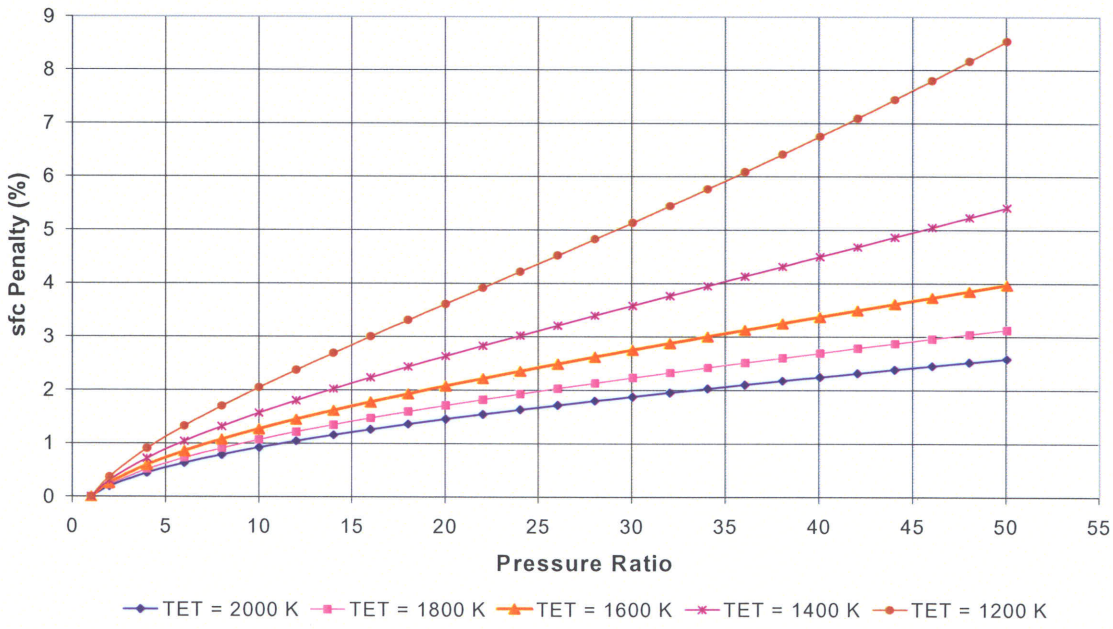


Figure 3.10 Effects of Engine PR and TET on sfc Penalties for β=0.05

Equations (3.26), (3.28) and/or (3.30) can be used to generalise the effects of engine parameters such as pressure ratio and TET and also study the performance of various

jet engine configurations with bleed air extraction. Therefore, thrust penalties reduce as the pressure ratio of the engine reduces, while sfc penalties reduce with increased fuel to air ratio. More specifically, in the case of sfc, equation (3.28) reveals that penalties reduce as the TET increases and/or the pressure ratio of the engine reduces. The effects of cycle pressure ratio and TET on sfc and thrust are plotted in Figures 3.10 – 3.12 for β=0.05 and γ=1.4; the effects of compressor isentropic efficiency have been neglected.

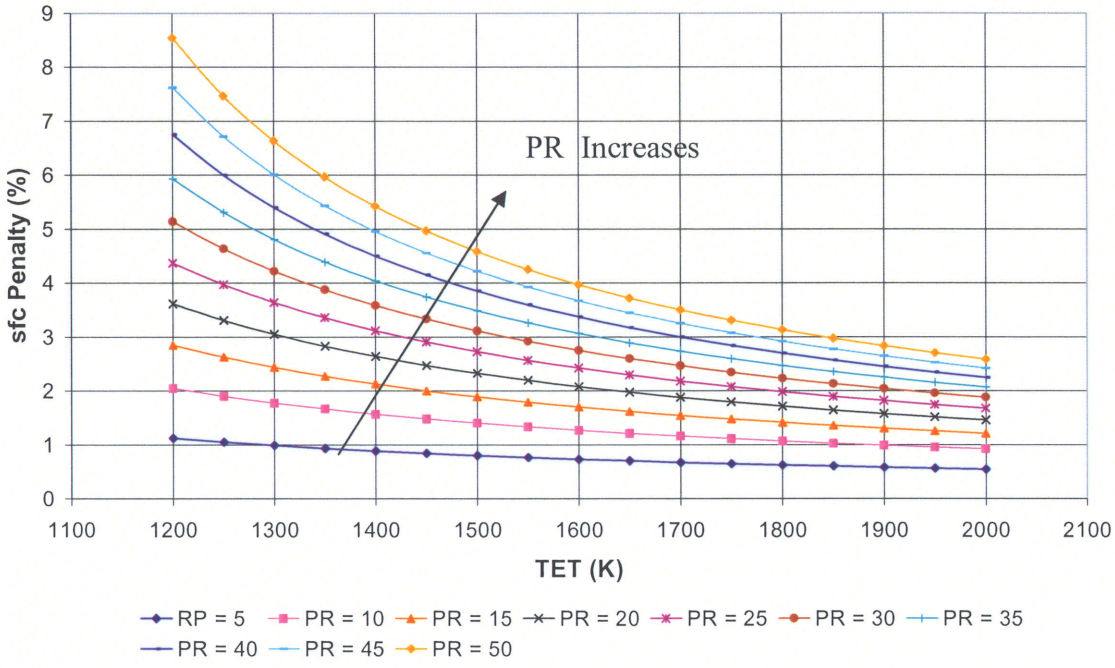


Figure 3.11 Effects of Engine PR and TET on sfc Penalties for β=0.05

The above-mentioned equations can also be used to study the effects of increased percentage of bleed air. The effects are shown in figure 3.12 (a & b) for fixed pressure ratio and TET respectively. In figure 3.12a, a pressure ratio of 25 was selected, while in Figure 3.12b TET was kept constant at 1600 degrees Kelvin. From the plots it can be concluded that for fixed TET, pressure ratio and compressor efficiency the sfc penalties increase linearly with off-takes. However, the penalties increase rapidly as the cycle TET reduces and/or pressure ratio increases. Once again, due to the nature of the analysis effects of compressor isentropic efficiency and pressure thrust have been neglected.

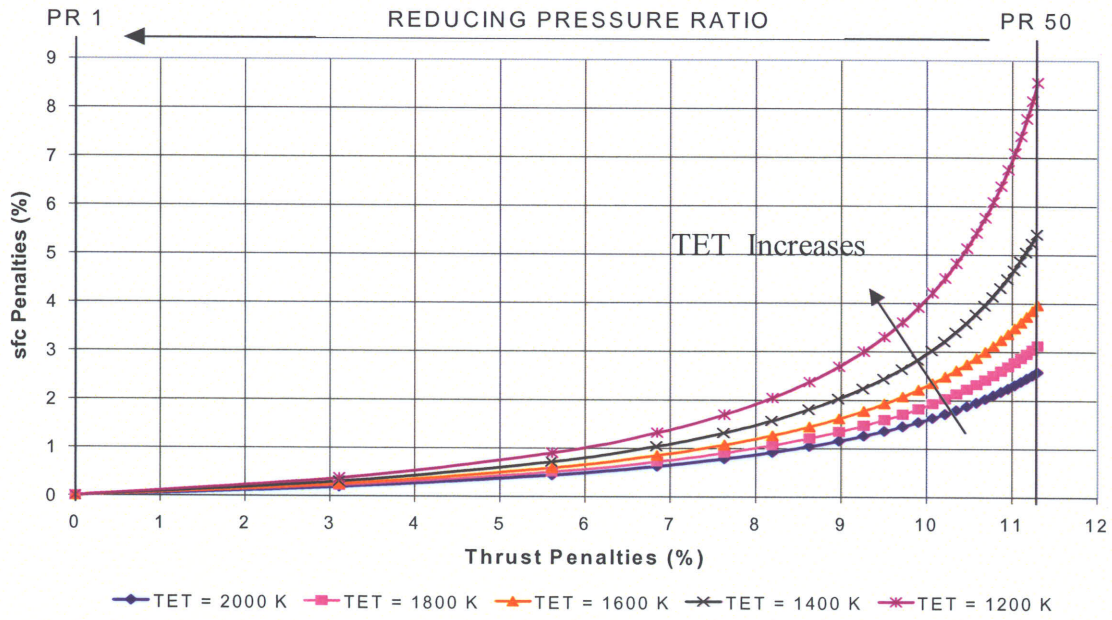


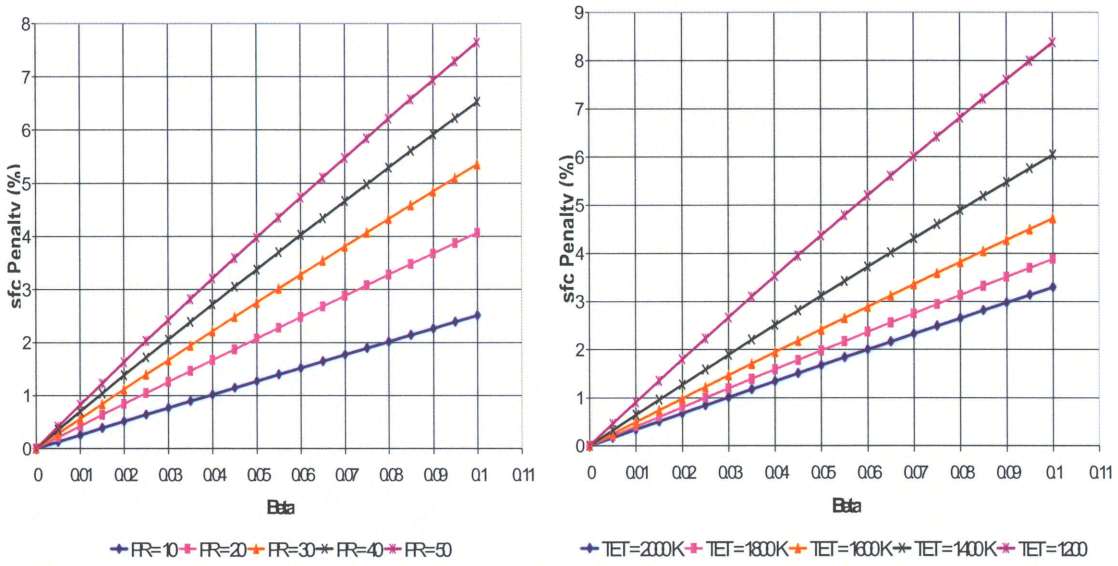
Figure 3.11 Effects of Engine PR and TET on sfc and Thrust Penalties for β=0.05

Although the results underestimate the effect on pressure thrust and assume that the engine operates at static conditions, similar trend lines will be obtained by taking into account the full effect of pressure thrust and flight velocity, however it must be noted that the magnitude of the penalties will differ.

This also holds true when comparing equivalent amounts of bleed air and shaft power off-takes. Therefore, performance penalties will follow the same pattern in both cases, however the magnitude of the penalties will be higher in the case of bleed air off-takes.

3.4.2 Validation

The accuracy of the theoretical method was compared against the results obtained from a Turbomatch model. For this purpose an arbitrary single spool turbojet engine was considered. The engine design point was fixed for a sea level static thrust of 92 KN, a mass flow of 100 kg/s, TET of 1550 degrees Kelvin and an extremely high pressure ratio of 30. Although, such a compressor pressure ratio is unrealistic for a single spool engine and will affect the stability of the compressor, it was considered a representative extreme case for studying the validity of the theoretical method over wide range of engine configurations and parameters. Changes in thrust and sfc for different amounts of bleed air are shown in Figures 3.12, 3.13 respectively.



(a) (b)

Figure 3.12: Effect of Off-take Size on sfc Penalties

From figure 3.13, it can be seen that as the percentage of bleed air increases the deviation of the theoretical results increases as well. This is mainly due to the fact that equations (3.23) and (3.26) used to model the effects of bleed air on thrust, underestimate the effect of off-takes on pressure thrust. This error is also accumulated in the case of sfc penalties were a similar deviation form Turbomatch results can be observed. During the analysis it was noted that the error was proportional to the percentage of the pressure thrust component relative to the total thrust. Therefore, the accuracy of the method can be improved by correcting the penalties with the percentage of pressure drag or by modelling the effects of off-takes as outlined in Figures 3.15 and 3.16.

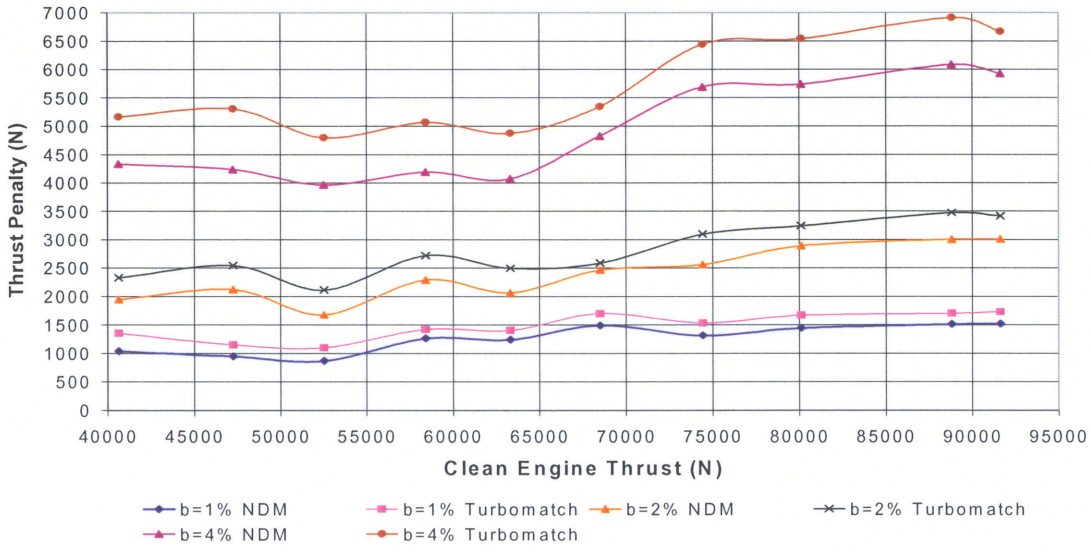


Figure 3.13: Thrust Penalties for Single Spool Turbojet Engine

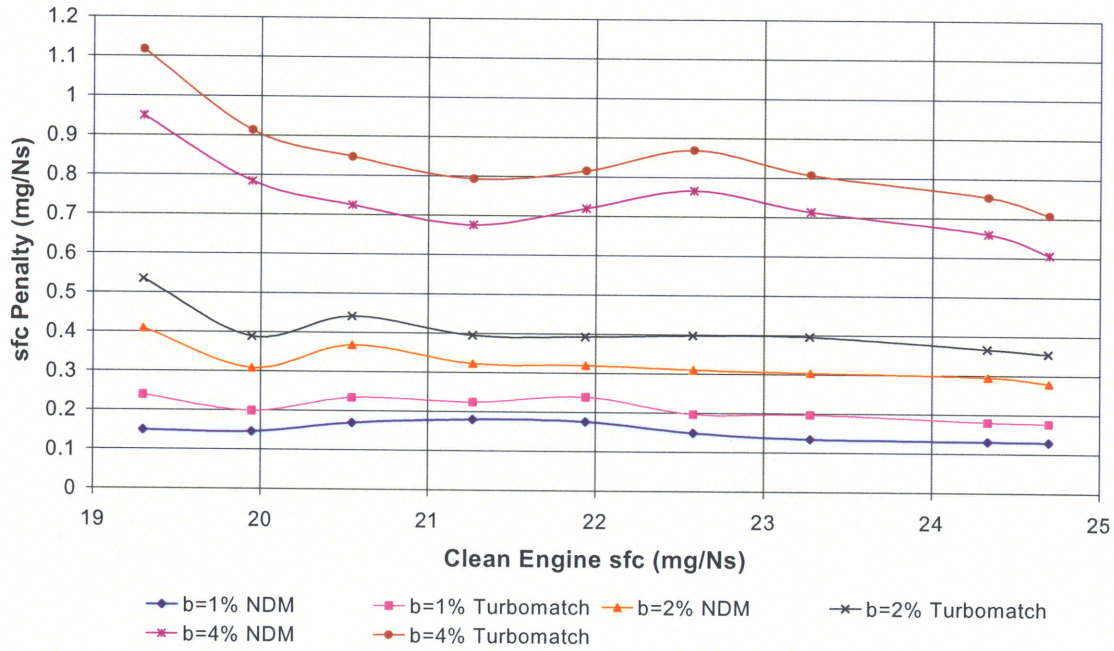


Figure 3.14: sfc Penalties for Single Spool Turbojet Engine

Despite this deviation, in all cases the results follow the expected trend lines, giving a clear indication of the effects of off-takes on thrust and sfc penalties, i.e. both Turbomatch and theoretical results predict a linear increase of penalties with increased off-takes.

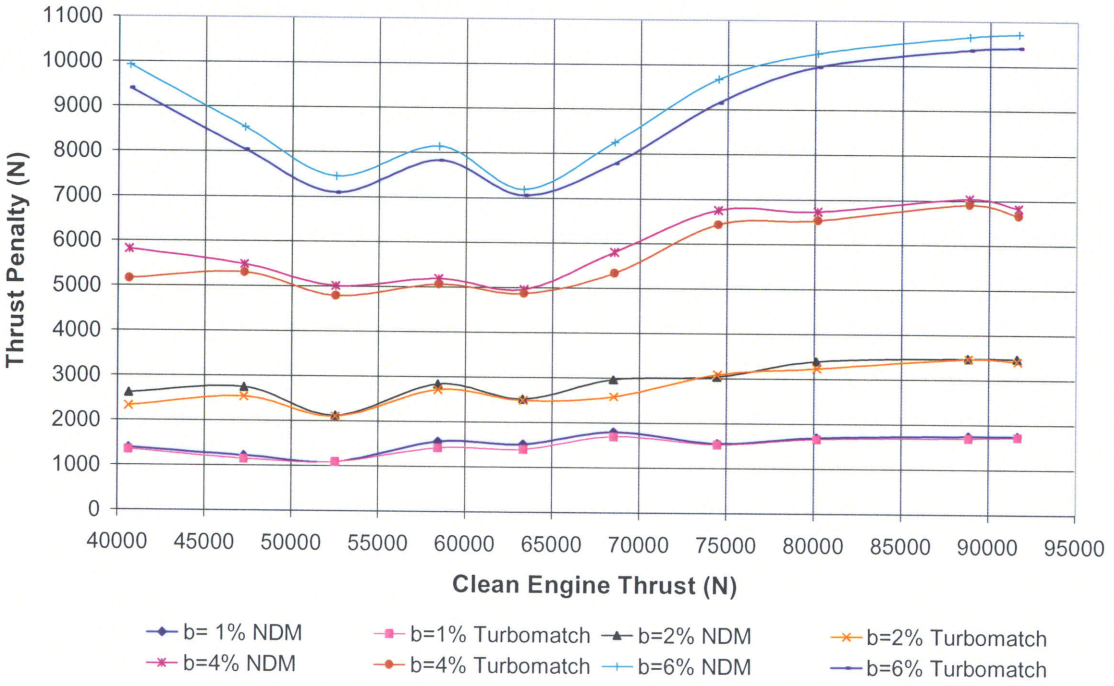


Figure 3.15: Corrected Thrust Penalties for Single Spool Turbojet Engine

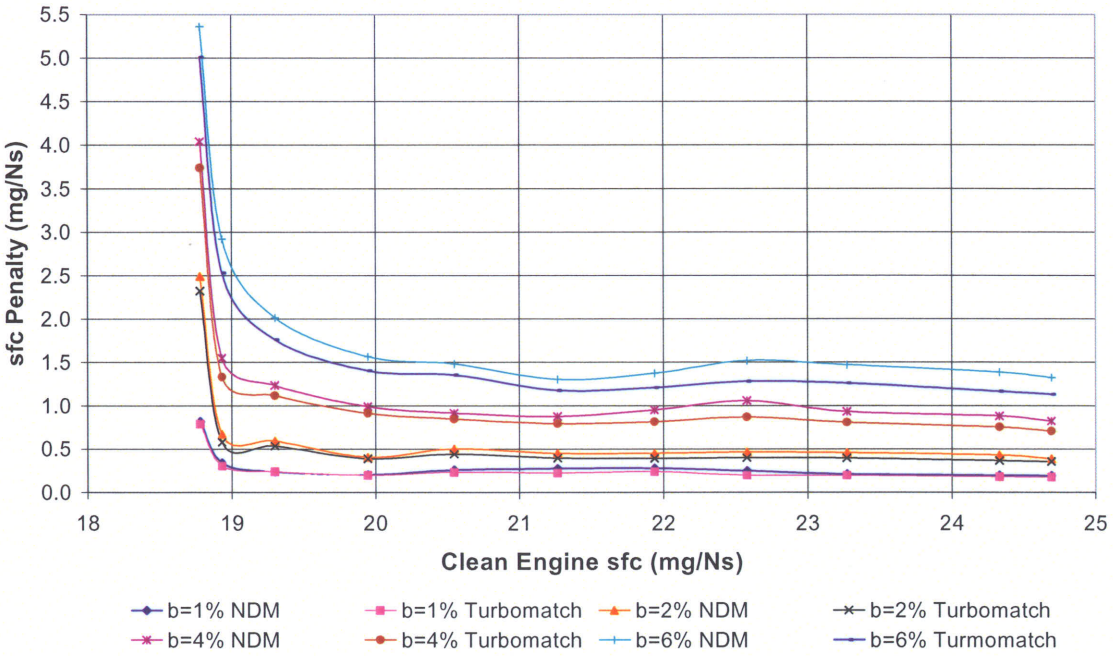


Figure 3.16: Corrected sfc Penalties for Single Spool Turbojet Engine

The movement of the compressor running line with extraction of various amounts of bleed air is shown in Figure 3.17. The plots include the findings from Turbomatch and the results derived from equations (3.11) and (3.15) as solid lines and operating points respectively. The results give a clear indication of the displacement of the running line with bleed air off-takes, as discussed in paragraph 3.1.The analysis can be used at the early stages of the design process of the engine to study the movement of compressor running line with extraction of off-takes and the provision of adequate surge margins.

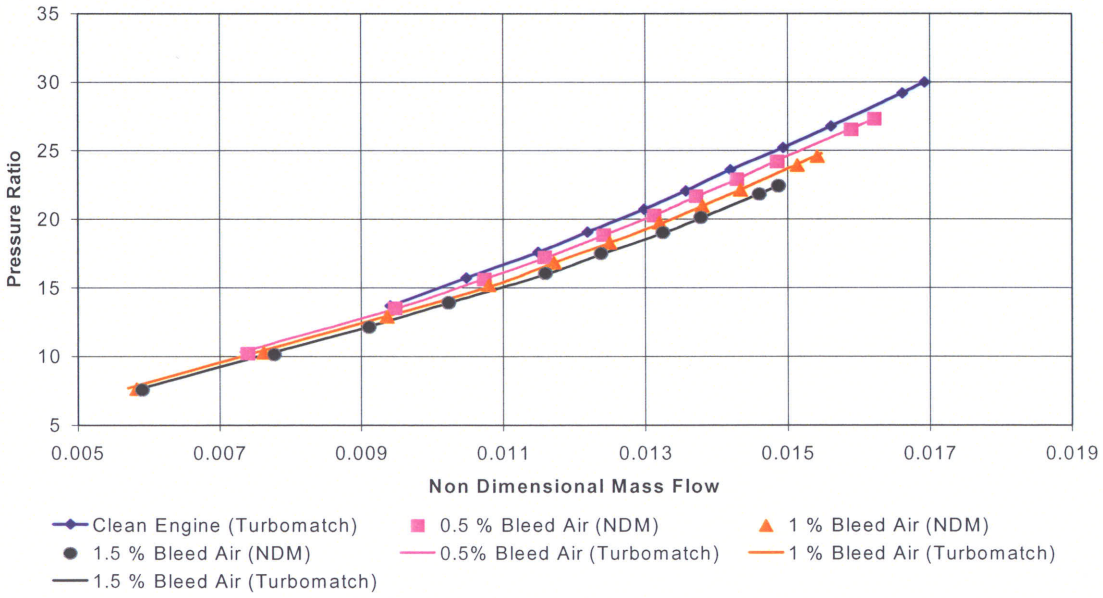


Figure 3.17: Operating Line for Clean Engine and Bleed Air (Turbomatch Results Vs Analytical NDM Results)

Finally, equation (3.28) can be used to analyse the trend lines of sfc penalties when secondary power is extracted, i.e. cases examined in paragraph 3.2. For fixed percentages of off-takes performance penalties increase as the difference of TET and compressor delivery temperature reduces. Therefore penalties reduce with engine power setting and altitude but increase with Mach number. For altitudes greater than 11.5 Km, the term (T_4-T_3) remains constant leading to constant sfc penalties.

The same equation can also be used to study the effects of fixed amounts of off-takes. It was noted in paragraph 3.2.3 that, in the case of the turbojet engine extraction of a fixed amount of off-take with increase in Mach number led to essentially constant percentage penalties, Figure 3.18. For fixed TET, the nominator of equation 3.28 reduces as the Mach number increases. The denominator, however, increases with reduced percentage of off-take. Therefore the constant penalties are the net result of two counteracting factors: (i) reduced non-dimensional power setting and (ii) reduced percentage of off-take. The same argument also holds true for the findings reported by Godner and discussed in paragraph 3.3.3.

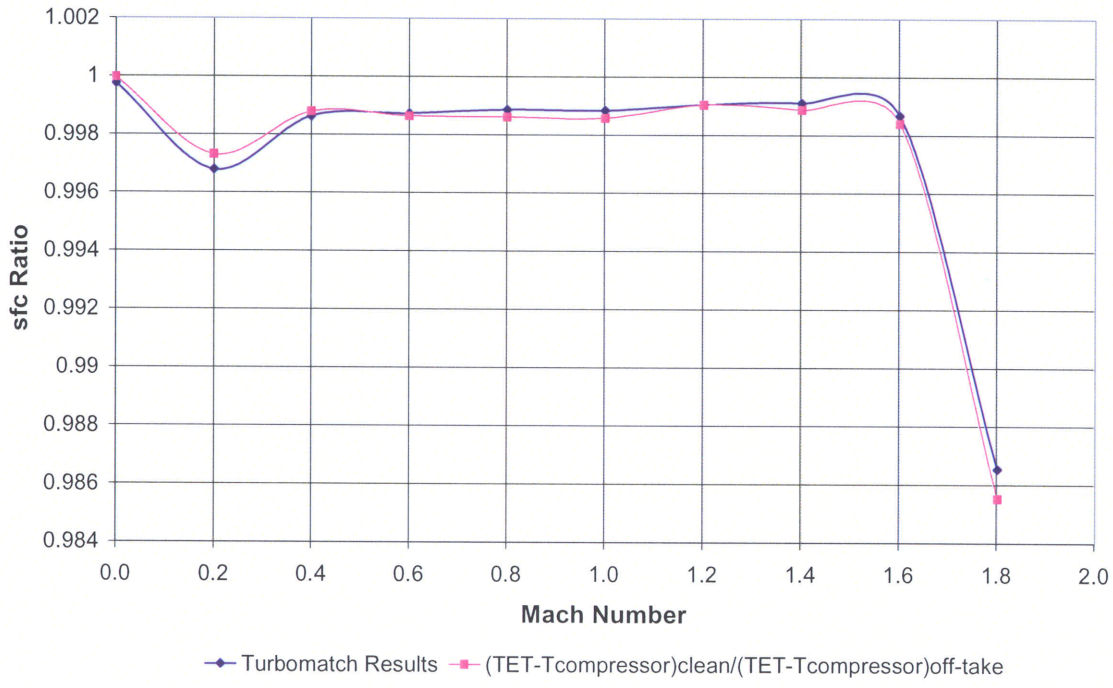


Figure 3.18: Effect of Fixed Amount of Off-Take on Turbojet Engine

3.4.3 Discussion and Limitations of the Method

Although fuel flow is often described as a quantity flow rate, such as mass flow, in reality it is an energy flow rate, i.e. power input:

$$WFE = W_F * FHV$$

Where W_F is the fuel flow rate and FHV is the calorific value of the fuel. This energy if put into a gas can be expressed as:

$$\text{WFE} = \text{FAR}* \text{W}_{\text{AIR}} * \text{FHV or } WFE = \frac{C_P \Delta T}{FHV} * W_{GAS} * FHV \Rightarrow WFE = C_P * W_{GAS} \Delta T$$

Both shaft power and bleed air off-takes are supplied by this energy flow rate and at the expense of available power for the production of thrust, i.e secondary power is supplied by the fuel and limits the efficiency of the engine in terms of thrust delivered. Therefore, the effect of any off-take can be expressed in terms of power input into the engine. For any given compressor pressure ratio, shaft power and bleed air off-takes can be expressed as a percentage of the mass flow going though the engine and subsequently the mass flow of the gas. Therefore, from the above equation it follows that for given Cp performance penalties will depend on the temperature rise of the gas, which is described by the term (T_4-T_3).

The method described in the previous paragraphs of this chapter provides both a qualitative and quantitative analysis on the effects of TET and compressor pressure ratio when secondary power is extracted. Although a number of previous studies on the subject identified the effect of off-take size as a primary factor, they all failed to isolate the individual effects of engine configuration and the thermodynamic details of the cycle. The present analysis identified the significance of these factors for the first time and provided a useful physical explanation of all the effects that take place. Although the analysis is based on a number of assumptions, i.e. constant component efficiency and chocked exhaust nozzle, it is accurate enough for estimating performance penalties and compressor characteristics at the early stages of the design process. Finally, although the expression in equation (3.28) has limited application, i.e underestimates the effect of pressure thrust and its accuracy is limited for static conditions only, it can be used for a qualitative analysis and to identify and/or explain performance trend lines.

The main drawback of the method is that it requires a number of parameters and more importantly the expression of off-takes in terms of engine core power, which in turn may not be known. As a consequence, although the method of analysis is applicable to all engines, the presentation of the results do not merge into a single curve but vary with different percentages of off-takes and different engine configurations. For this

reason a semi-empirical expression was derived and is presented in the following paragraph.

3.5 Development of Non-Dimensional Semi-Empirical Power Factor

For a number of years the author tried to express sfc penalties associated with the extraction of shaft power in terms of a non-dimensional power factor. The method was based on a simplified form of the "pi-theorem" The objective of the analysis was the development of a non-dimensional expression that would include all the parameters that affect the performance of the engine. Based on the findings of the literature review and prior to the theoretical analysis, the emphasis was placed on an expression that would relate the magnitude of any given amount of off-take to the core power of the engine. For this reason the following parameters were selected:

- Thrust
- Specific Thrust
- PR
- TET
- Power Off-take

All the above parameters can be expressed in terms of the three basic units: mass, length and time. For the purpose of the analysis Temperature can be expressed as velocity squared and specific thrust as velocity. Therefore the ratio of specific thrust to TET squared is non-dimensional. Pressure ratio is non-dimensional by definition. Thrust and shaft power can be expressed as the quasi-dimensional parameters shown below:

$$\frac{Shaft\ \ Power}{P_1\sqrt{T_1}} \text{ and } \frac{Thrust}{P_1}$$

These parameters were re-arranged to give the following expression:

$$\frac{ST \times PR \times \sqrt{\frac{Thrust}{P_1}}}{\frac{Shaft\ \ Power}{P_1\sqrt{T_1}} \times \sqrt{TET}}$$

Several Turbomatch models were used to plot the sfc penalties against this relationship. However, the results merged into a single curve only in the case of the

same engine or different engines with similar thermodynamic details. The deviation increased when turbojet or small bypass ratio engine were compared against typical high bypass ratio turbofan engines.

Taking into account the findings of the theoretical analysis, fuel flow and shaft power off-takes can be combined to give a quasi-dimensional expression as follows:

$$\beta = \frac{Fuel\ Flow \times ST \times PR}{\frac{Shaft\ Power}{P_1\sqrt{T_1}} \times \sqrt{TET}} \quad or \quad \beta = \frac{(TET - T_3) \times PR \times \frac{Thrust}{(1+BPR)}}{Shaft\ Power \times \sqrt{TET}}$$

Consider any given engine with the same station numbering as in figure 3.8. The power off-take can be expressed in terms of the energy flow rate into the gas. Therefore at station No 4:

$$\beta = \frac{\frac{W_4 \times \sqrt{T_4} \times (\Delta T)}{P_4 \times T_4}}{\frac{Power\ Off - Take}{P_1 \times \sqrt{T_1}}} \Rightarrow \beta = \frac{W_4 \times (\Delta T) \times P_1 \times \sqrt{T_1}}{Power\ Off - Take \times P_4 \times \sqrt{T_4}}$$

Let BPR be the bypass ratio of the engine varying from zero to infinity, then:

$$\frac{W_4}{W_2} = \frac{1}{1+BPR} \Rightarrow W_4 = \frac{1}{1+BPR} \times W_2$$

also:

$$\frac{P_2}{P_4} = \frac{P_2}{P_3} \times \frac{P_3}{P_4} = \frac{1}{PR} \times \frac{P_3}{P_4}$$

Therefore, assuming constant combustion losses:

$$\beta = \frac{W_2 \times \frac{1}{1+BPR} \times (\Delta T) \times \sqrt{T_1}}{Power\ Off - Take \times PR \times \sqrt{T_4}}$$

Re-arranging pressure ratio and substituting specific thrust gives:

$$\beta = \frac{\frac{Thrust}{1+BPR} \times (\Delta T) \times PR}{Power\ Off\ -Take \times \sqrt{T_4}}$$

Figure 3.19 plots the sfc ratio against the quasi-dimensional relationship for several engines operating at various conditions and with extraction of different amounts of shaft power off-takes.

Three areas of interest are identified. At relatively high values of power factor, i.e. low off-take and bypass ratio and/or high TET, pressure ratio and thrust, sfc penalties increase linearly and remain relatively low. Continuous reduction of power factor causes an adverse, non-linear sfc penalty. Further reduction of power factor causes penalties to increase rapidly.

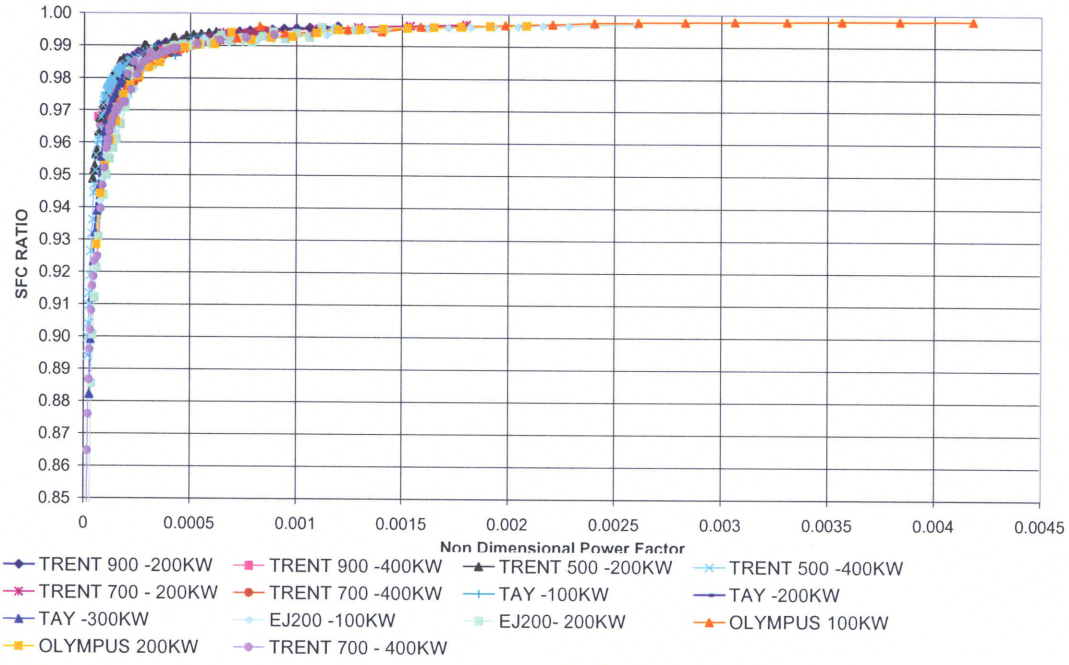


Figure 3.19: sfc ratio Vs Non-Dimensional Power Factor

For a fixed percentage of off-take relative to the core power, sfc penalties are dependent on the temperature rise across the combustion chamber. The latter is a function of compressor pressure ratio and TET. The percentage of the off-take will be a function of thrust, compressor pressure ratio, TET and bypass ratio. However, all this parameters can be related to each other through specific thrust. Therefore, for a properly matched engine with fixed thrust, pressure ratio and TET it follows that specific thrust is uniquely related to bypass ratio and the mass flow through the core of the engine. The latter implies that core power reduces with bypass ratio and therefore

a fixed amount of shaft power will constitute a higher percentage of off-take. Therefore, for fixed TET, thrust and pressure ration, increased bypass ratio causes power factor to reduce.

A useful tool to explain the effects of compressor pressure ratio is the design point map, Figure 3.20. This map has on its axes the two main variables of performance, sfc -representing fuel efficiency- and specific thrust -representing size. As the so called fishhook curve of constant TET moves clockwise in the direction of increasing pressure ratio, it is seen that at first sfc diminishes and specific thrust increases until point A is reached, where specific thrust is at its maximum. Further increase of pressure ratio causes specific thrust to fall, but sfc continues to fall until point B is reached. Further increase of pressure ratio causes specific thrust to fall and sfc to rise.

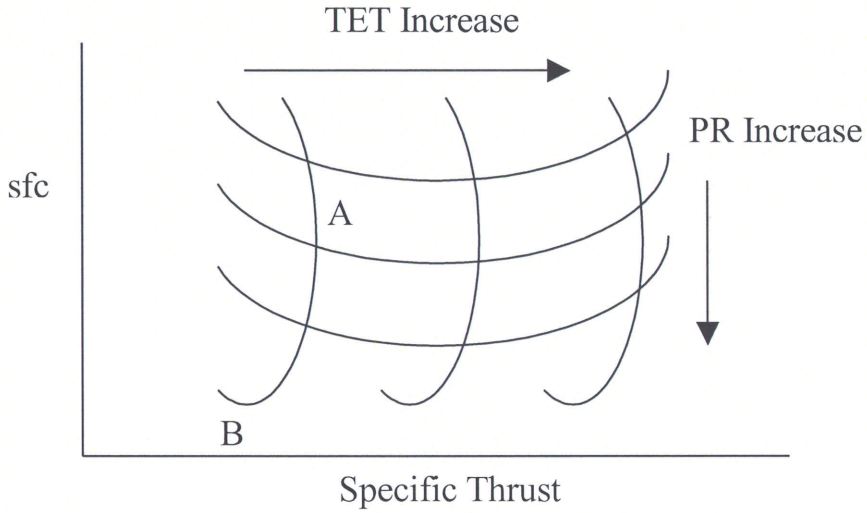


Figure 3.20: Design Point Map (Fixed Bypass Ratio)

Therefore, for a given engine with fixed thrust, bypass ratio and TET, compressor pressure ratio will affect both the core power of the engine and the temperature rise across the combustion chamber. These counteracting effects of pressure ratio are combined in the above-mentioned expression by the overall change of specific thrust, pressure ratio and fuel flow.

TET will also affect the core power of the engine and the temperature rise across the combustion chamber. Once again the counteracting effects of TET are considered by fuel flow, specific thrust and the squared root term. The latter appears in the denominator of the power factor to take into account the reduction in mass flow and core power as TET increases.

The derived power factor is a useful tool that enables the designer to consider the effects of shaft power extraction at the early stages of the design process and provides a better mean for optimising the overall performance of the engine by taking into account efficiency, size and effects of off-takes. The parameters required are the same with those at the design map thus, enabling a direct comparison of the sensitivities of the various cycles to power off-takes. Finally, the nature of the analysis enables the inclusion of the magnitude of the off-take and not its size relative to the core power. The latter simplifies considerable the analysis.

Peackock[53] claimed that future turbofan engines would experience a 5% sfc penalty due to increased off-takes. He also mentioned that penalties could be reduced to 2% by optimising the method of extracting secondary power. It is shown in Chapter 8, that these claims are in good agreement with the findings of this study. From Figure 3.18, it is seen that theses sfc penalties correspond to low power factor and lie in *design areas* where sfc penalties could increase rapidly. These issues are further discussed in Chapter 8.

Chapter 4: Performance Investigation & System Integration for the More Electric Military Aircraft

The chapter deals with the application of the MEA/E concepts on a typical military aircraft. It is shown that due to the unique design characteristics of this type of aircraft, special consideration must be given to its performance and system architecture. The chapter concludes with the findings of a case-study reviewing the effects on the performance of a typical military engine of driving an afterburner pump. The alternative systems examined for the provision of the power required include: i) a mechanical drive ii) electrical drive with varying efficiency iii) the use of an air turbine receiving air from various engine stations. The work does not strengthen the argument for adopting a bleed system to drive the afterburner pump. However, it is concluded that the main drawback of adopting an electrical system would be its weight

4.0 Introduction

Modern military aircraft have developed into complex flying systems making increasing use of secondary power to enhance both their flying qualities and fighting effectiveness.

Similar to the case of civil aircraft, present military aircraft use three forms of secondary power: electrical, hydraulic and bleed air or pneumatic. Due to the size constrains, however, secondary power is distributed through an airframe-mounted gearbox that drives hydraulic pumps and generators by a direct shaft link to the engine. To account for an engine failure in the case of a twin-engine aircraft, a cross drive is used to link the two gearboxes. Although this design offers high efficiency, the high levels of shaft power and the complexity of the design limit its overall effectiveness. Over the years, alternative concepts have been proposed including the use of pneumatic and/or electric links.

Although, the present study focuses on civil applications of the MEA/E concepts, special consideration should be given into the performance and systems design issues of a military aircraft. These issues are reviewed in the followings paragraphs of this chapter. It was shown in the literature review (Chapter 2) that the acclaimed benefits of the more electric concepts include improved engine performance and system mass. These issues are reviewed in the followings paragraphs of this chapter.

4.2 Implications of Electrical Secondary Power Systems on Military Aircraft

Despite the advantages of replacing present forms of secondary power with electrical power, the high levels of shaft power off-takes and the additional handling problems may affect the design and performance of the engine. Moreover the high power consumption of the afterburner pump, coupled with the fact that it is used for such a small proportion of the flight, means that an electric drive system for this pump may not be the optimum solution. An alternative to an electric drive is to retain the current system –i.e. direct shaft power off-take via a gearbox- or to use an air turbine powered by air bleed from the engine compressors.

The aim of the study is to determine the effect on the performance of a typical military engine of driving an afterburner pump by:

- A direct shaft power off-take via a gearbox
- An electrical drive via an embedded or external electrical generator and motor
- An air turbine taking bleed air from the engine compressor

The study includes the following tasks:

(a) Review various Secondary Power Systems (SPS) and identify possible candidates for comparison

(b) Define a baseline aircraft/engine system

(c) Define typical power requirements for the baseline aircraft

(d) Define fuel flows and power requirements for the operation of the afterburner pump

(e) Evaluate engine performance penalties and implications incurred by the provision of power requirements

(f) Extent the analysis to include aircraft performance

For the purposes of the study the EJ-200 powered Eurofighter has been defined as the baseline aircraft. However, since the configuration of the engine –i.e. bypass ratio, number of spools etc.- affects the performance penalties imposed by extracting secondary power, comparisons have been made where possible with the RB-199 engine.

As far as performance penalties are concerned the study concentrates on the sfc and thrust penalties as well as increase in Turbine Entry Temperature and variation in surge margin.

4.3 Secondary Power System Implications of Military Aircraft

Unlike the case of a civil aircraft and because of the low utilisation of bleed air, secondary power requirements for military applications are often dictated by shaft power off-takes. Extraction of secondary power, for both the engines and the airframe systems, will affect the performance of the engine and therefore it is necessary to design the engine and system components for the minimum possible penalties. To be able to minimise penalties, the size and type of off-takes and how these vary with engine and airframe conditions must be known. As explained before, extraction of shaft power may limit the surge margin of the compressor and therefore, the overall performance of the aircraft.

For fixed off-takes, whether bleed air and/or shaft power, the running line can be correctly placed at the engine design stage for maximum efficiency. Off-takes are unlikely to remain constant but a compromise between different operational phases will allow the running line to be placed in about the correct position. This however becomes much more difficult in the case of a military aircraft where the off-takes vary greatly in magnitude between different phases.

One situation where the magnitude of off-takes is likely to be both large and very difficult to balance is during engine failure. In this case, the remaining engine must not only provide increased thrust but larger proportion of the secondary power. This will cause performance penalties but in addition handling problems may become extremely severe.

Considering engine failure in the case of a military aircraft, the accessories are shaft driven by the remaining engine. Given that it is not always possible to shed the power requirements in the way of non-essential loads, the running line will move towards surge. This may adversely affect the engine's ability to operate safely in conditions of

inlet distortion or rapid engine transients for which compressor surge margin is necessary. Although bleed air requirements for air-conditioning may also increase it is unlikely to counter the effect on the running line moving towards surge.

4.4 Typical Secondary Power Systems on Military Aircraft

Secondary power on most present military aircraft is used to perform the following functions:

- Engine starting
- Hydraulic Pumping
- Electrical Power Generation
- Environmental control
- Auxiliary Power Unit (APU) starting

4.4.1 Secondary Power Sources

Main Engine: As in the case of civil transport aircraft the main source of secondary power in the case of military aircraft is the main propulsion unit. Secondary power is extracted in two forms:

- Shaft power
- Bleed air

Shaft power from the high-pressure compressor is transmitted to the accessory gearbox via a power off-take shaft. Bleed air is normally provided by extracting air from the end of the high-pressure compressor. Unlike the case of a civil transport aircraft, a military aircraft incorporate a single port for tapping bleed air. This method is usually adopted to avoid the complexity, size and mass penalties of increased number of ports. The primary aim of military aircraft design is to minimise the size and mass of the aircraft with fuel efficiency having a lower priority.

Auxiliary Power Unit: The APU can be regarded as small gas turbine engine used to provide both bleed air and shaft power. APU's are primarily used on the ground to make the aircraft self sufficient and independent of external power sources. Pneumatic power is normally provided by bleeding air from the main compressor of the engine or by using a free turbine design to drive a load compressor. The latter case incorporates a gearbox that is driven by the free turbine and is used to supply power the load compressor and other aircraft accessories. Such a configuration allows the gas turbine compressor to run at its optimum pressure ratio for either maximum power or fuel efficiency, and the load pressure compressor is matched according to the requirements of the environmental control system. In the first case however, the pressure ratio of the gas generator is dictated and compromised by the bleed air pressure requirements.

Additional secondary power sources include a ram air turbine, emergency power units and batteries.

4.4.2 Accessory Gear Box

The accessory gear box is used to drive the aircraft's and engine's accessory including: hydraulic pumps, electrical generators, fuel pumps and oil pumps. The accessory gearbox cab either engine mounted or airframe mounted.
Engine Mounted Gearbox: In this case the accessories are mounted directly onto the engine wheel case. Although such a design could reduce the net mass of the system it increases the frontal area of the engine and implies to poor maintainability.

Airframe Mounted Gear Box: In the case of an airframe-mounted gearbox, a power off-take shaft is used to transmit the power from the engine to a remotely mounted gearbox. Despite its limitations in terms of efficiency and mass this configuration has several advantages over the conventional engine mounted gearbox. The main advantages are listed below:

- Reduced engine and aircraft frontal area
- Simplified engine installation
- Better access to the engine and the accessories
- Accessories are located away from the engine's hot spot
- Flexibility to drive the accessories from a separate source

4.4.3 Hydraulic Power

Hydraulic power is required to drive flight controls, flaps, undercarriage, brakes, etc. The source of this power will be mechanically or electrically driven pumps and its loading at any time will be dictated by:

- Pump efficiency – this depends on pump speed but in practise will not vary significantly over the engine speed range

- Efficiency of the hydraulic system and the leakages across flying control selectors – these are dependent on system pressure and are usually independent of flight conditions

- Specific loads to drive certain services – these loads tend to be of small duration and depend on altitude and flight speed

In general hydraulic loads would vary significantly across the flight envelop with peaks occurring during take off, landing and combat phases. The total power required would be a function of flow rates and control forces. Although both of them are depend on speed and altitude, their specific requirements are very different.

Control forces tend to increase with speed and reduce with altitude. Flow rates, on the other hand, tend to increase with altitude and reduce with Mach number. The opposing effects of altitude and speed on flow rates and forces required, can lead to constant power requirements. Although most military aircraft operate at 4000 psi there are examples of aircraft operating at even higher pressures. While increasing system pressure for a given force output reduces the power requirements at high-speed conditions it has the opposite effect at low speed operations. Therefore, rather than using a high-pressure system during the whole of an operation, problems for loads, seals etc. may be reduced by increasing the pressure during periods of high demand. Adoption of variable hydraulic pressure could reduce power loads at different flight conditions. Such an approach is particularly useful on fighter aircraft where there is a large range in the output forces and flow rates required between high speed – low altitude and low speed – high altitude operations.

4.4.4 Electrical Power

Electrical power generation is an essential service that powers items such as fuel pumps, avionics, instruments, flight control system equipment, weapons system, etc.

Provided that reheat pumps are not electrically driven the variations in total demand vary by small amounts over the whole flight envelope and are usually not affected by altitude and Mach number. However, in the case of an electrically driven de-icing system power demands would increase during icing conditions. In the case of un-reheated engines fuel pumps power requirements are a function of engine condition and could be matched to engine speed rather than use of constant speed AC pumps.

In present, electrical power generation and distribution is mainly accomplished by Constant Speed Drive (CSD) driven generators at 220/115V, 400 Hz, 3 phase AC power. Some 28V DC is also used. To obtain constant voltage and frequency, a hydromechanically driven constant speed unit is normally fitted between the accessory gearbox and the generator. The constant speed drive unit maintains a constant generator speed regardless of the changes in engine speed by adding or subtracting speed as required. Modern military aircraft tend to use a combined brushless generator an constant speed drive known as the integrated drive generator. This configuration reduces the combined mass and size of the individual units and allows for a common lubrication system. The CSD unit of such a system is inevitably complex and introduces inefficiencies.

Generation systems which provide constant frequency AC supply without the use of a CSD unit through solid state conditioning of the output of a directly driven generator are already in service in a number of aircraft. Such systems are known as Variable Speed Constant Frequency (VSCF) systems. Although there are a number of ways of controlling the output, the main two types are the Cycloconverter or DC link. Each has its own advantages and disadvantages but both have the potential to be more reliable than the complex CSD and can provide higher system efficiency.

A further reduction in generation component mass and power requirements could be achieved if the power could be distributed in a Constant Voltage/Variable Frequency or even a Variable Voltage/Variable Frequency (VV/VF) form. Both fuel system pumps and electric anti-icing are candidates for variable frequency power use. Since it is unnecessary (and in some cases impractical) to convert the whole output of a VV/VF generator to high quality power, the ability to operate some of the loads directly from the generator could lead to improved power management.

4.4.5 Fuel System

Today's combat aircraft are rated to carry internal fuel loads of about a quarter of their take off mass. Even when the stowage problem is solved, changes in weight and

moments of inertia result in CG and stability problems. To ensure survivability and airborne stability fuel must be stored in a number of tanks increasing the complexity and power requirements of the fuel system. While engine mounted fuel pumps are usually hydraulically or shaft driven, transfer pumps and the rest of the fuel system components are usually electrically powered. However, in some cases, such as the Tornado, fuel transfer is accomplished by tank pressurisation.

During normal operation, i.e. unreheated, any aircraft fuel pump driven by the engine will require relatively small loads and these will be a function of fuel flow demand and driving speed. This will generally mean a reducing power with altitude but increasing with forward speed. Although unreheated fuel system loads will be fairly small compared to hydraulic and electrical, they constitute a large proportion of the engine accessory loads. Finally in the case of a reheat pump the loads become enormous and may compromise the performance of the engine and the design of the secondary power system.

4.4.6 Secondary Power Systems Configurations

To account for an engine failure in the case of a twin-engine aircraft, a cross drive is used to link the two gearboxes. Over the years, alternative concepts have been proposed including the use of pneumatic hydraulic electric and hybrid links.

Mechanical link: Shaft power is transmitted to the gear box via a clutch or a hydromechanical torque converted. Either the remaining engine or the APU can supply the shaft power. The power transmitted can be used to start the engine or drive the accessories in the case of an engine failure. Despite its high efficiency the configuration is complex and heavy.

Pneumatic link: In this case bleed air from the remaining engine or the APU can be supplied to an air turbine mounted on the accessory gearbox. The power output from the turbine can be used to start the engine or drive the aircraft's accessories. The system requires high amounts of bleed air and has a large effect on the performance of the engine. In addition, the ducting required from and to the APU leads to large mass penalties.

Electrical Link: Electrical power can be supplied form an engine or APU driven generator to an electric motor mounted on the accessory gearbox. Once again the power output of the motor can be used to drive the remaining aircraft accessories or start the engine. The system requires the generators to be oversized and this may lead

to mass penalties. Penalties, however, can reduce with the use of combined starter-generators. The configuration also assumes the use of electrically driven fuel and hydraulic pumps. Such a design can simplify the design of the accessory gearbox and permits the hydraulic pumps to be remotely placed.

Hydraulic Link: Hydraulic power is supplied from an engine or APU driven pump to a hydraulic motor located on the accessory gearbox. The system is complex and introduces large mass penalties. In addition hydraulic systems are known to suffer from poor maintainability.

4.5 Use of Engine Performance Program Study Secondary Power Penalties on a Typical Military Engine

An engine performance synthesis program (Turbomatch) developed at Cranfield University was used to evaluate the above mentioned off-take effects on a typical military engine.

The Turbomatch program enables an engine to be modelled by using different blocks, each one representing engine components such as, compressors, turbines, burners, etc. The program can be used to define an engine for a given design point and then to study its off-design performance.

Turbomatch has been used to evaluate the performance penalties and implications imposed by extracting different amounts and modes of off-takes for driving the afterburner pump of a typical military aircraft. In addition, the program was used to determine the fuel flows, and thus power requirements, for the operation of the afterburner pump over a wide range of flight conditions.

i) A two spool turbofan of 60 KN dry static thrust, BPR 0.75 and PR 26, representing on EJ-200

ii) A three-spool turbofan of 50 KN dry static thrust, BPR 1.5 and PR 24, representing on RB-199.

The study focused on the first engine but comparisons have been made where possible with the second model.

Fuel flows and power requirements for the afterburner pump are shown in Figures 4.1 and 4.2 respectively. Both fuel flows and power requirements increase with speed and reduce with altitude.

Air bled form the engine compressor can be expanded through a turbine to provide power. For the EJ-200 model, at sea level and various Mach numbers, extracting 1.1 kg/s of air form the IP port and expanding it to ambient conditions at 80% efficiency can produce 150 kW.

When comparing the performance penalties of extracting given amounts of bleed air and shaft power for the EJ-200 and RB-199, it was found that the latter experienced significantly higher penalties. For both shaft power and bleed air off-takes the higher performance penalties came as a result of the followings:

- Higher bypass ratio and therefore lower mass flow rates through the core of the engine

- The use of three spools reduce the power output of the HP turbine and therefore a given off-take constitutes a higher percentage of the total power delivered to drive the compressor

- Lower TET, leading to reduced available power for the turbines

- Lower mass flow through the engine

The penalties were even higher in the case of shaft power off-takes where the power required to drive the HP compressor was significantly lower compared to EJ-200.

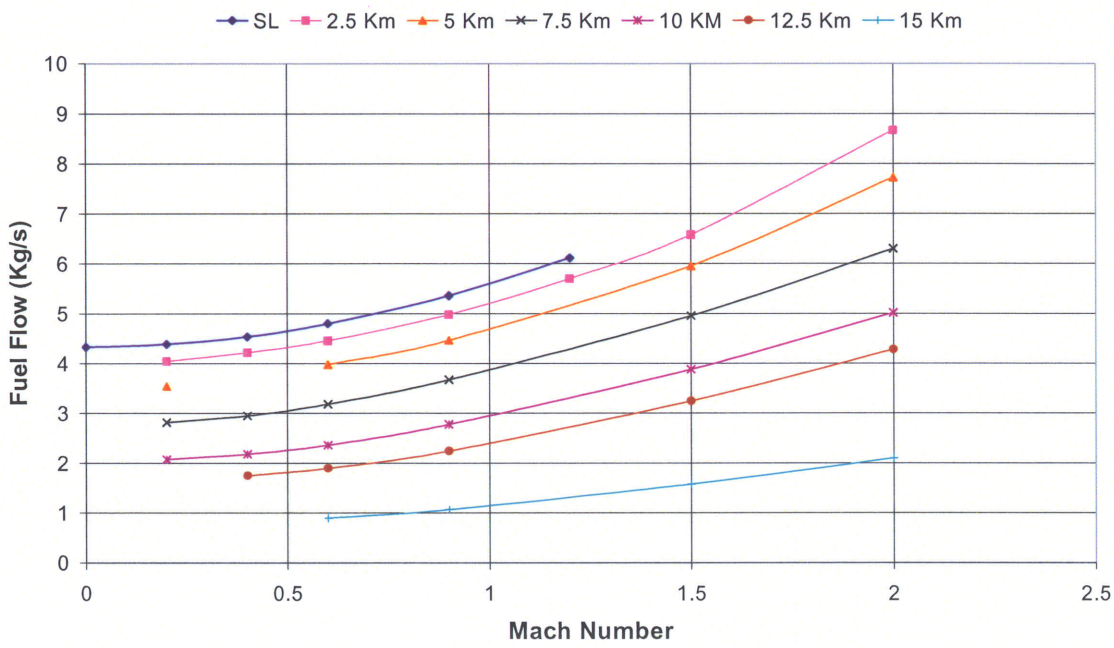


Figure 4.1: Afterburner Fuel Flow

4.5.1 Parametric Analysis-Presentation of the Results

Following the above findings, a parametric analysis has been performed to evaluate engine performance penalties over a wide range of operations. The cases considered include typical flight conditions where the afterburner will be employed either to accelerate the aircraft or to maintain constant speed at high drag operations. The former refer to different altitudes at Mach numbers of 0.6-0.9-1.2. The second refers

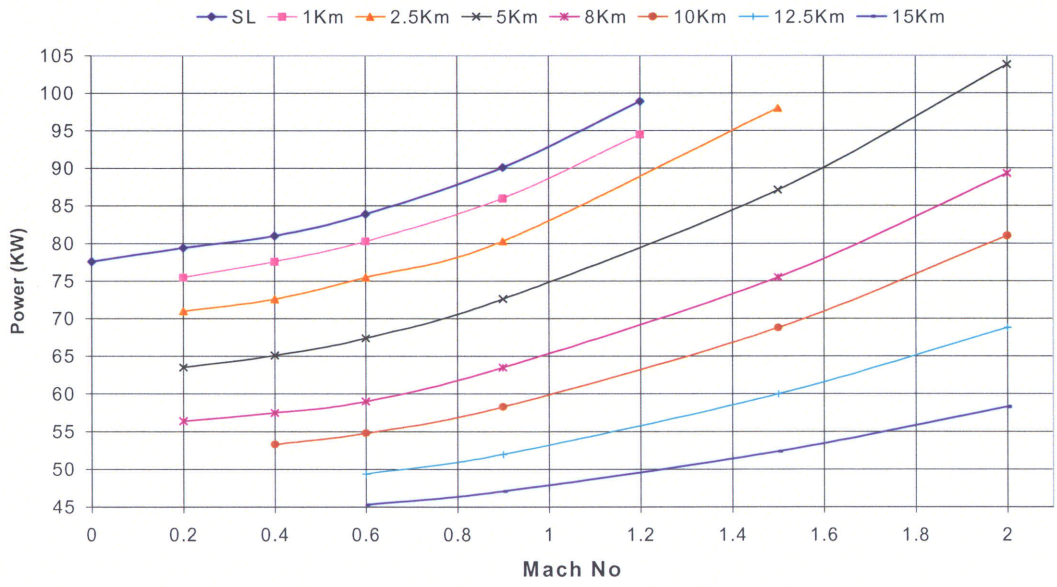


Figure 4.2: Afterburner Power Requirements

to high-speed operations around 2.0 Mach and over a wide range of altitudes. Finally the take off conditions at sea level and 0.2-0.4 Mach have also been considered.

For the purposes of the analysis, a number of different modes of extracting and producing power have been considered:-

Direct Shaft Drive: This is the conventional method of providing secondary power. In this case 95% efficiency has been assumed.

Electrical Drive: In accordance with the general trend of adopting a More Electric and/ or All Electric Aircraft (MEA/AEE) configuration, an electric drive has been considered. Different values of efficiencies, ranging form 65-85%, have been considered.

Pneumatic Drive: Since there have been concerns over the weight and size of a generator required to supply the secondary power needed for the operation of airframe and engine systems, an alternative method of driving at least some of the accessories involves the use of a turbine receiving air form the engine compressors. This scheme may be particularly useful for the drive of the afterburner pump and/ or during an engine failure situation. The cases examined include:

- Bleed air form LP port being expanded to ambient conditions
- Bleed air from IP port being expanded to ambient conditions
- Bleed air form IP port being expanded through a turbine and then returned downstream the LP turbine. For this scheme the total pressure of the air being returned must equal the pressure downstream the LP turbine
- Bleed air form HP port being expanded through a turbine and then returned downstream the LP turbine. Again the same conditions as above must be met.
- Air being bled between the HP and LP turbines and then returned downstream the LP turbine.

Initially it was intended to study the effects of extracting air from the IP port and then expanding it to ambient. This selection reflected the fact that most military engines incorporate a single bleed port usually located at the front stages of the HP

compressor. The exact location of this port would be defined by the operation of the Environmental Control System (ECS) and the Anti/De-Icing requirements in the case of a thermal de-icing system. A preliminary analysis revealed that by locating the IP port at the front stages of the HP compressor, so that it delivers air at 8.4 atmos and 571°K during take-off conditions, the pressure and temperature of the air delivered would be sufficient under most operating conditions.

However, feedback from the industry suggested to return the bleed air back into the engine. For this to be possible the air could only be returned downstream the LP turbine. To ensure sufficient pressure ratio for the operation of the turbine, air could be extracted from the IP port, the end of the HP compressor and finally after the HP turbine. At this point it was also decided to extract air downstream the LP compressor and expand it to ambient. The different options are shown diagrammatically in Figures 4.3-4.8.

For all the cases, engine performance penalties were plotted in terms of both power produced and amount of air extracted. For the purposes of this study, the isentropic efficiency was taken at 80%. For more details regarding the design and performance of the turbine reference 81 should be consulted. The values given here are based on thermodynamic relationships only and do not consider limitations, imposed from the design of the turbine, i.e. rotational speed, mechanical integrity, size, etc.

In all cases 0.3 Kg/s of air have been extracted from the IP port for the operation of the Environmental Control Unit along with 150 kW of shaft power for the operation of the remaining airframe and engine systems. These values considered to be the baseline loads and represent the peak loads applied during combat and with both engines operating. Typical results are presented in Figures 4.9-4.10 for sfc and thrust penalties respectively. The highlighted points on each graph include the penalties imposed for the operation of the afterburner pump. Performance penalties for the operation of the afterburner pump can be found by subtracting the baseline penalties from the value corresponding to each highlighted point. Increase in TET required to recover the thrust loss is shown in Figure 4.11

A parametric analysis studying the production of various amounts and modes of secondary power is presented in Figures 4.12-4.15 for sfc thrust TET and surge margin penalties respectively. The study focused on three configurations. Extractions of shaft power -either mechanical or electrical- bleed air and finally a hybrid system where

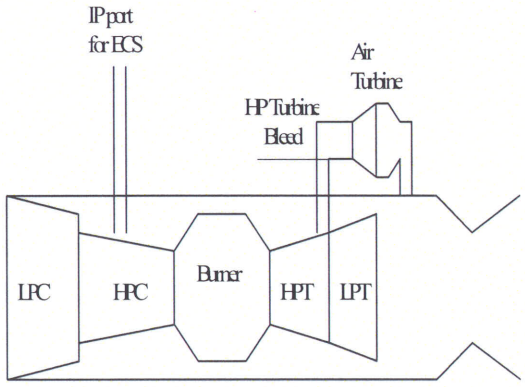


Figure 4.3: Bleed air downstream the HPT and returned downstream

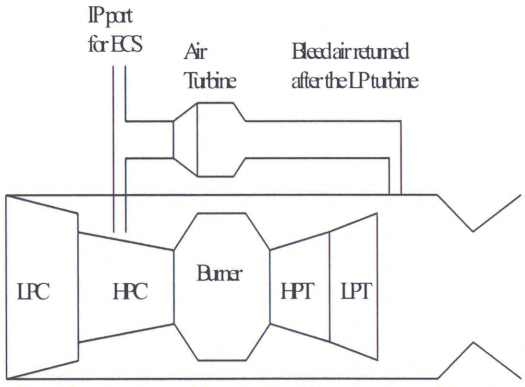


Figure 4.4: Bleed air from IP port returned downstream the LPT

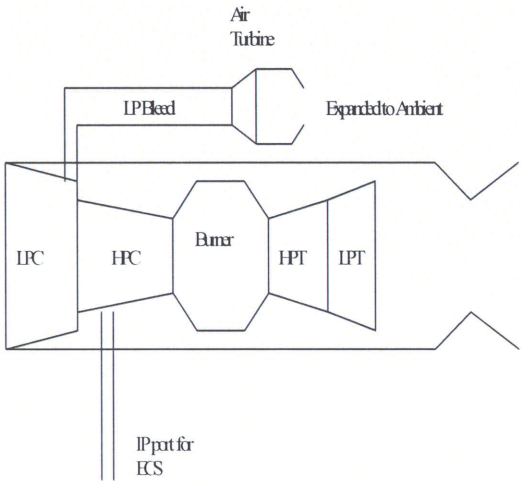


Figure 4.5: Bleed air downstream the LPC expanded to ambient

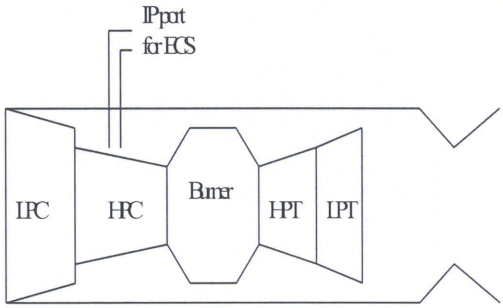


Figure 4.6: Bleed air from IP port for ECS

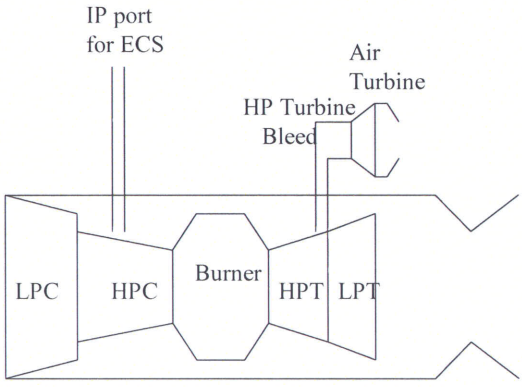


Figure 4.7: Bleed air downstream the HPT expanded to ambient

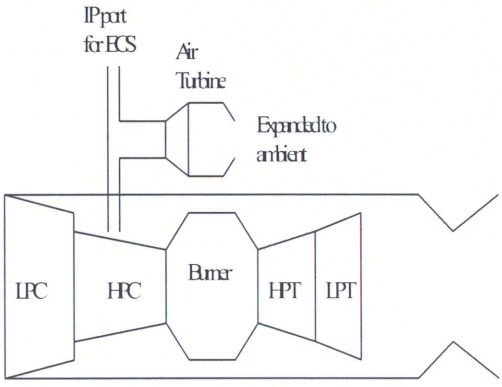


Figure 4.8: Bleed air from IP port expanded to ambient

different amounts of power are produced by a combination of shaft and bleed power. It is noted that for a given amount of power output extraction of shaft power led to lower engine performance penalties. No problems concerning compressor surge were experienced.

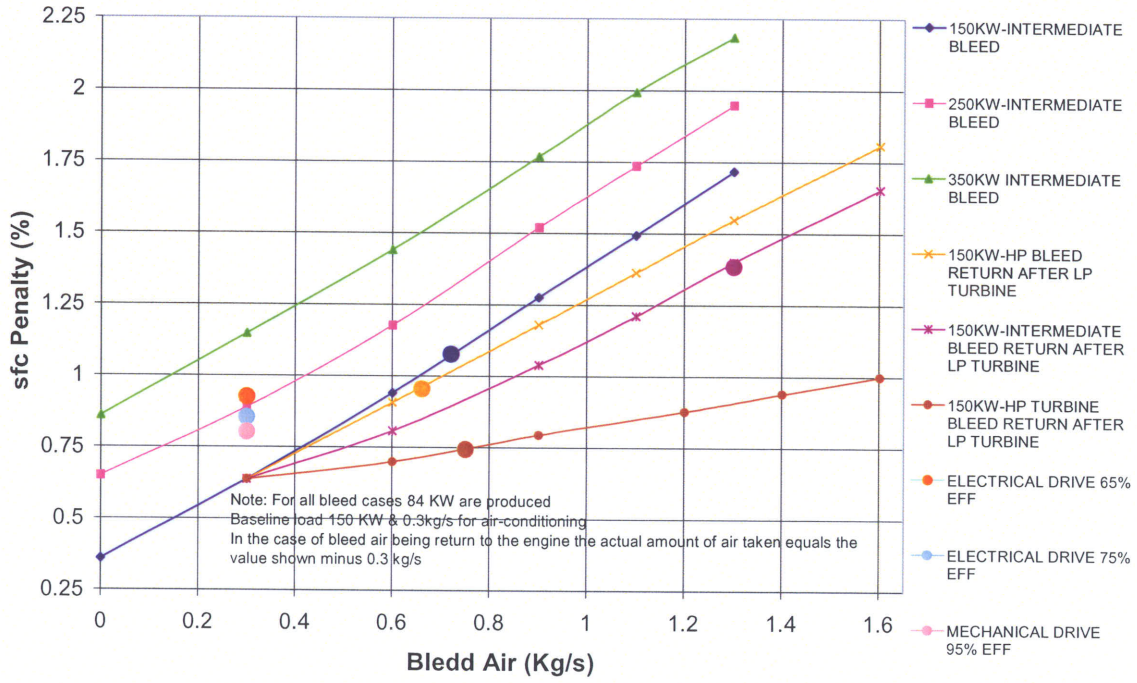


Figure 4.9: sfc Penalties at Typical Take-off Conditions – Sea Level 0.2 Mach

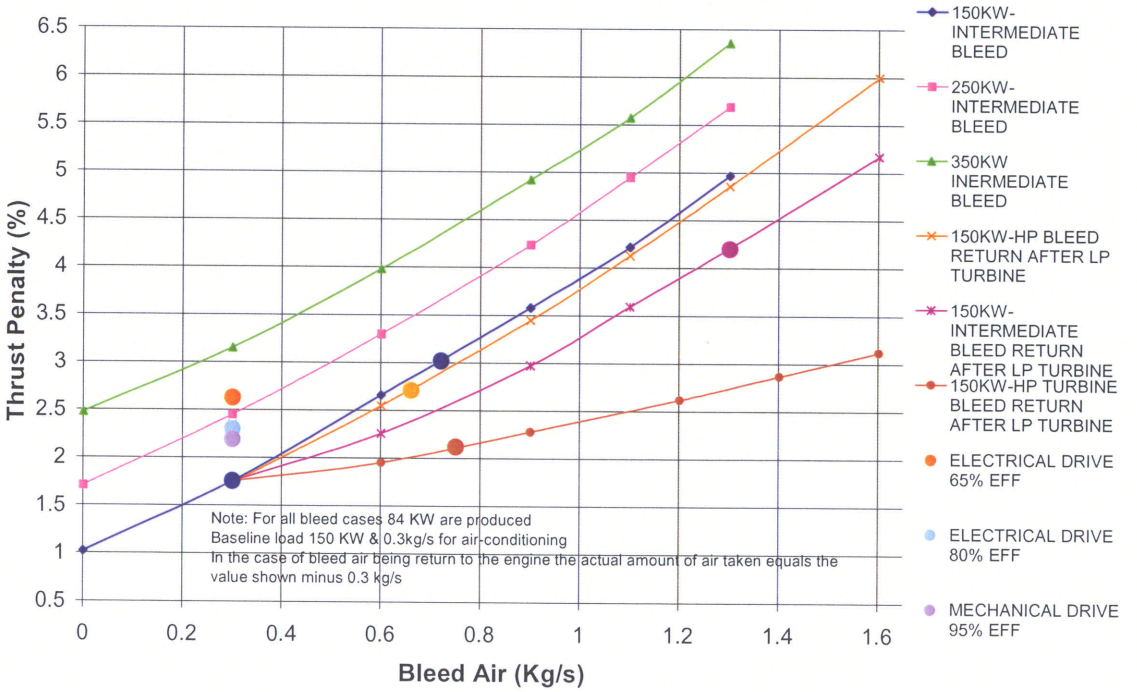


Figure 4.10: Thrust Penalties Penalties at Typical Take-off Conditions – Sea Level 0.2 Mach

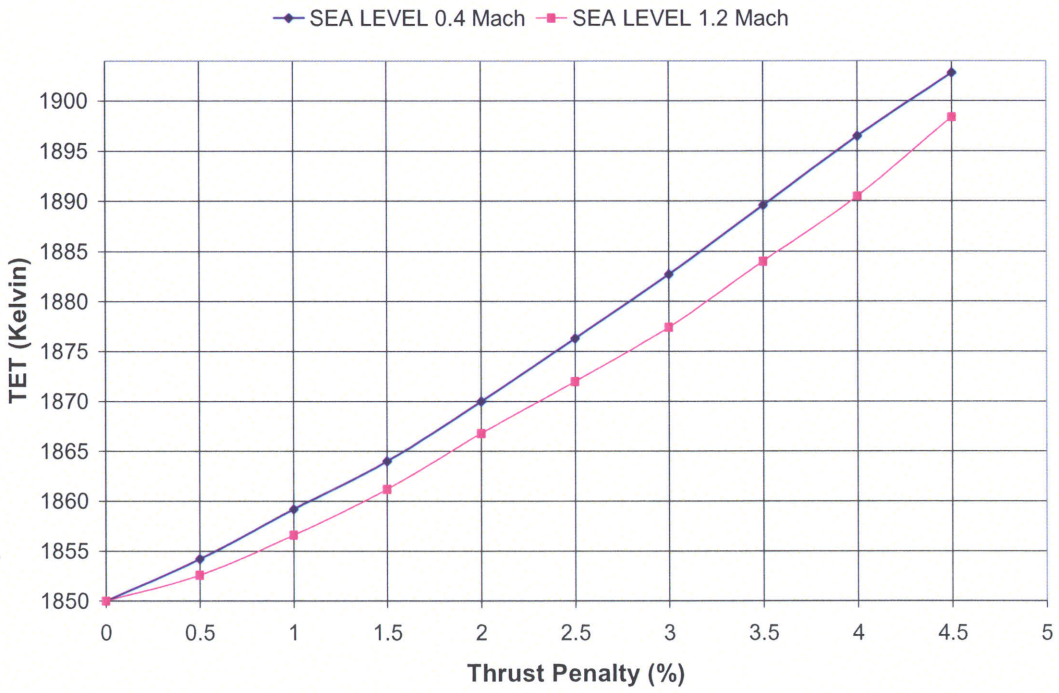


Figure 4.11: Increase in Turbine Entry Temperature at Sea Level Conditions

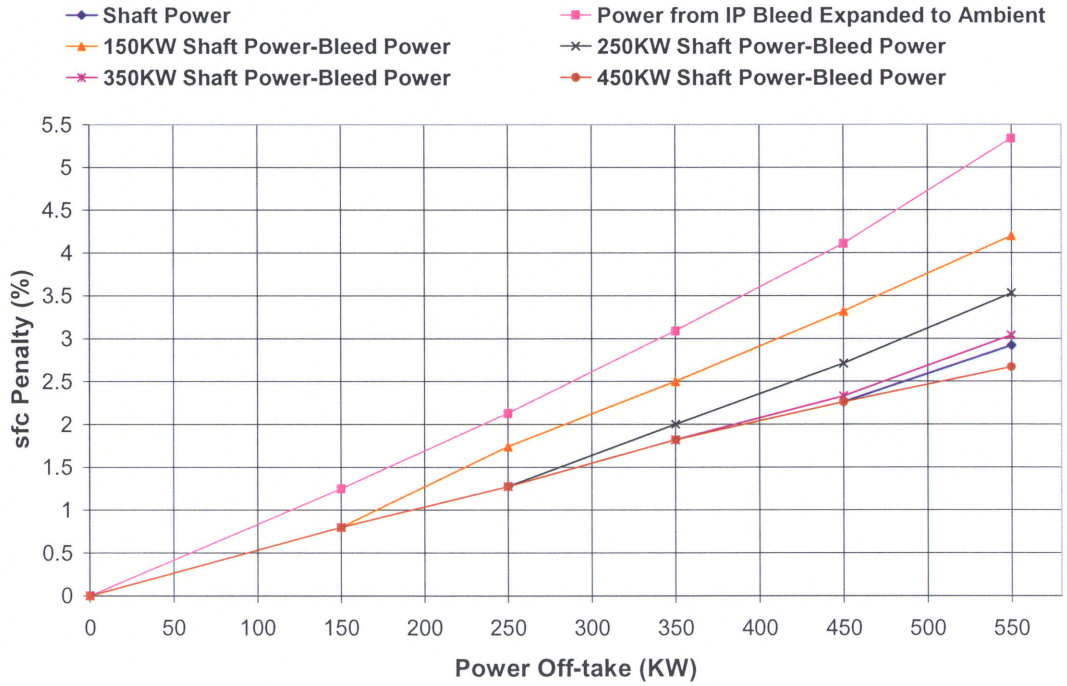


Figure 4.12: Parametric Analysis - sfc Penalties at 10000 meters and 0.6 Mach

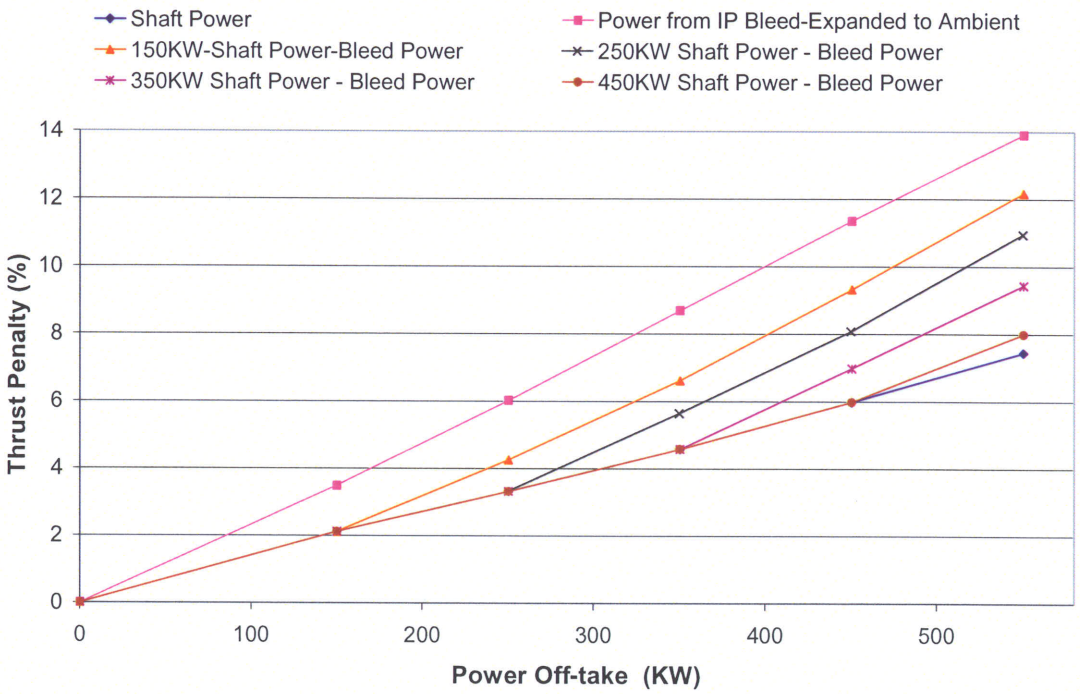


Figure 4.13: Parametric Analysis – Thrust Penalties at 10000 meters and 0.6 Mach

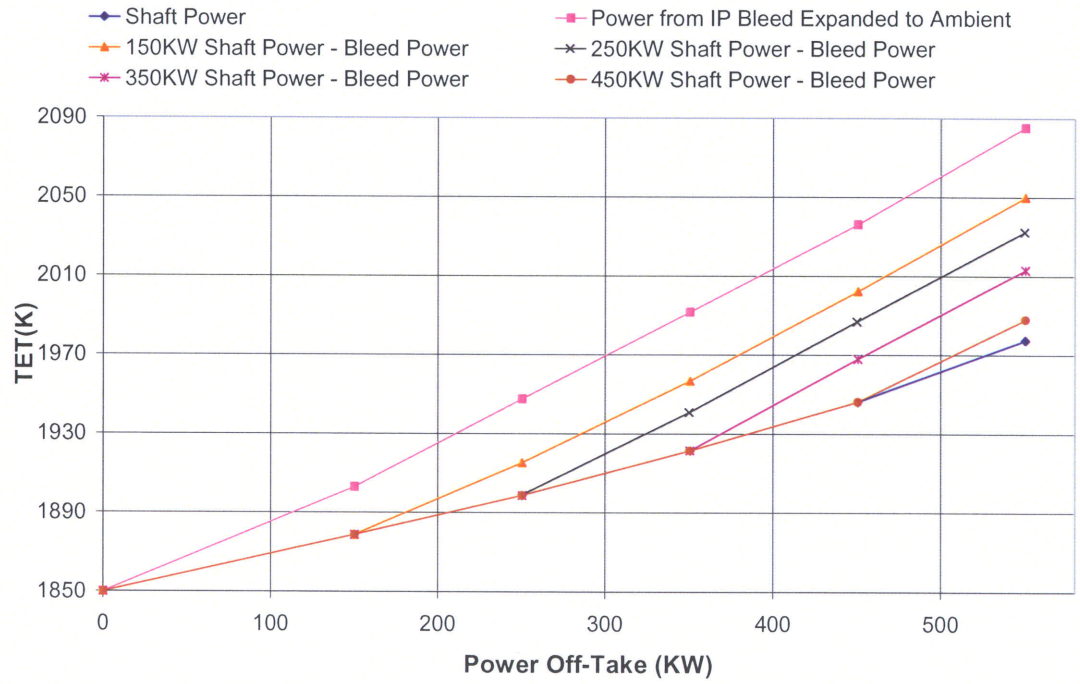


Figure 4.14: Parametric Analysis – Increase in Turbine Entry Temperature at 10000 meters and 0.6 Mach

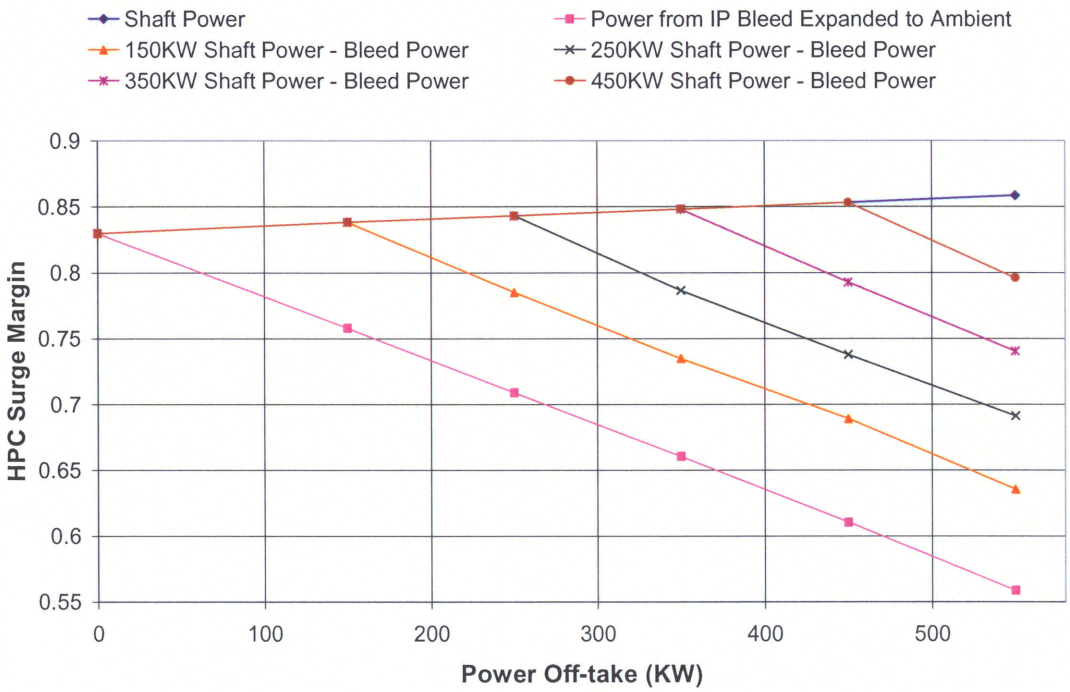


Figure 4.15: Parametric Analysis Effect on High Compressor Surge Margin at 10000 meters and 0.6 Mach.

4.6 Investigation of Mechanical, Electrical and Bleed Air Systems on EJ-200 Turbomatch Model.

4.6.1 Definition of Critical Cases

Since it would not be practical to include all the flight conditions reviewed, for the purposes of this report a selection has been made and only representative cases have been included. These cases are termed critical in terms of power requirements for the operation of the afterburner pump, engine performance penalties, engine handling considerations and finally likelihood of using the afterburner pump.

Power Requirements: In terms of power requirements the critical case proved to be at 5000m and 2.0 Mach. At these flight conditions the power required to drive the afterburner pump would be around 100KW with the actual value depending on the overall efficiency of the driving system, the rotational speed of the pump and the diameter of the impeller. This case would be the design point for sizing the generator and the motor in the case of an electrical drive, as well as sizing the turbine in the case of a pneumatic system.

Assuming a rotational speed of 30,000 rpm and 7.5 cm impeller diameter, the power required to drive the pump would be 90 KW. The total power extracted from the engine would vary from 140 KW, in the case of an electrical drive with overall efficiency of 65%, to 95 KW in the case of a mechanical drive with 95% efficiency. Considering the use of an air turbine the rotational speed increased to 32,500 rpm giving a power output of 108 KW.

Engine Performance Penalties-Thrust/sfc: For all the off-take methods considered, the highest thrust penalties of driving the pump occurred at 5000 meters and 0.6 Mach. As explained earlier on, for a given off-take, performance penalties increase with altitude and reduce with Mach No. However, in the case of driving the afterburner pump the power requirements increase with speed and reduce with altitude. This opposite effects of altitude and speed on engine performance penalties and power requirements cancel out at around 5000 m and 0.6 Mach, leading to higher thrust penalties.

In the case of sfc, on the other hand, maximum penalties occurred at different flight conditions for each system. This is largely because of the opposite effects of altitude and speed on propul sive and thermal efficiencies. However once again the higher penalties occurred at 5000 meters and around 0.6-0.9 Mach.

Considering, however, the total power requirements –i.e. base loads plus afterburner pump loads- the critical case for both thrust and sfc penalties occurred at 10000 meters and 0.6 Mach.

Increase in TET: The increase in TET required to recover the thrust lost would be a function of the thrust penalty, speed and altitude. Once again for a given percentage of thrust penalty, the rise in TET increases with altitude and falls with speed. However the dominant factor proved to be the percentage of thrust lost and therefore for all methods of driving the pump the higher penalties occurred at 5000 meters and 0.6 Mach. The increase in TET varied from 30°K in the case of a turbine receiving air from the LP port and expanding it to ambient, to 10° K in the case of a mechanical system with 95% efficiency. This value increased to 15° K in the case of an electrical drive with 65% efficiency.

Engine Handling Qualities: As explained in Chapter 3, shaft power extraction may be limited by the reduction in surge margin. At high values of non-dimensional speeds, i.e. high altitude-low speed operations, it is likely that the LP compressor would impose this limit. At low values of non-dimensional speed, i.e. low altitude-high speed operations, the HP compressor would be prone to potential surge. However, despite

initial concerns no problems concerning surge have been experienced. This is largely due to the fact that when the afterburner is employed the engine operates at, or near, rated conditions and thus the power produced is high, diminishing therefore the effect of an off-take on compressor line movement.

Although the surge margin of the HP compressor reduced significantly with increase in Mach No this came as a result of the operating condition of the engine and not the extraction of shaft power required to drive the afterburner pump. Even in the case of an electrical drive where the low efficiency of the system would require higher loads, the proportion of the off-take proved to be very low compared to the total power re-circulated within the engine and therefore, the movement of the running line was essentially unaffected. At low speed high altitude operations, on the other hand, the low power requirements did not impose significant movements on the running line of the LP compressor.

4.6.2 Consideration of Engine Performance Penalties with Mechanical, Electrical and Bleed Systems

Bleed Air: For a given amount of bleed air engine performance penalties reduce when the air is extracted either form the very late stations across the engine – i.e. downstream the HP turbine – and/or from the early stages of the compressor – i.e. LP compressor. In addition, the penalties reduce when the air being bled is returned within the engine. However, for a given amount of power output it was found that the lower penalties occur when higher pressure and temperature air is extracted and then expanded to ambient, with the only exception being the case where air is extracted after the HP turbine. In terms of engine performance, the most efficient method of driving the afterburner pump would be by extracting air downstream the HP turbine and then returning it after the LP turbine. Indeed this method of driving the pump proved to be slightly more efficient than a direct drive system. The rest of the bleed air systems introduced larger performance penalties even when compared with an electrical drive having 65% efficiency. From the remaining bleed air options the most effective ones proved to be extraction of bleed air from the HP port and returning it downstream the LP turbine, bleed air from the IP port and expanding it to ambient, bleed air from the IP port and returning it after the LP turbine and finally bleed air from the LP port and expanding it to ambient. However, as mentioned above all this options introduced higher performance penalties when compared to the mechanical and electrical drives.

Nevertheless, when considering the mechanical constraints imposed by the design of the turbine and the integrity of the system, Backus [80] in his report suggested the adoption of a system that would receive air from the LP compressor and then expand it to ambient. Despite the higher performance penalties, according to Mr Backus this represents the optimum design in terms of turbine size and weight.

Shaft Power Off-Takes: Since the engine does not discriminate between an electrical and mechanical drive the only difference in terms of engine performance penalties would be the efficiency of the system and the total amount of power extracted from the engine. A wide range of efficiencies varying form 65% to 95% was considered. The findings suggest that changes in system efficiency did not have a major effect on engine performance. Once again, system efficiency in terms of power requirements proved to be more significant during high-speed low-altitude operations where the fuel flows and power requirements were higher. However at those operating conditions the power extracted would constitute a small percentage of the total power re-circulated within the engine leading, therefore, to small variation in performance penalties with system efficiency. During low speed high altitude operations, where performance penalties tend to be more severe, the relatively small fuel flows and power requirements diminish the effect of system efficiency.

Over the years a number of studies[32, 58, 74] suggested that extracting shaft power from the LP shaft could reduce engine performance penalties. Moreover, since the afterburner is employed at high power settings the wide speed variation of the LP shaft would not impose such a great problem under these conditions. However, it was found that the savings were very small compared to the use of the HP shaft and therefore the complexity and the weight of system would not justify such a change.

4.6.3 Engine Performance Penalties

In the case of the mechanical system thrust penalties varied from 0.64% at 5000m and 0.6 M to 0.2% at 5000m and 2.0 M. These penalties increased to 1% and 0.366% respectively for the electrical system and to 2.36% and 0.66% in the case of the bleed system. The maximum increase in TET for the mechanical system was 10°K. The electrical and bleed systems experienced an increase of 15°K and 30°K respectively. In terms of sfc penalties, for the mechanical drive these varied from 0.154% to 0.09%. The electrical system experienced penalties varying from 0.28% to 0.15%, while the bleed system experienced penalties from 0.659%-0.5%.

4.6.4 Size of the Electrical Generator and Motor

A number of reports [82-84, 86-90] were used for the estimation of the electrical power requirements of a typical modern military aircraft. In the case of the Eurofighter Typhoon aircraft it is reasonable to assume that two generators each one rated at 60 KVA could be used. However it is also possible that the aircraft incorporates generators rated at 40KVA each, which in turn are overloaded during small periods of high power requirements. According to the power requirements for driving the afterburner pump shown in Figure 4.12, the peak loads would be around 140 KW in the case of an electrical system having 65% efficiency reducing, however, to 110 KW in the case of a system of 85% efficiency.

The overall efficiency of the electrical system would depend on the system layout and the advances in electrical technology. If a constant speed drive is used to drive the pump then the overall efficiency could be considerably lower than the 65% efficiency mentioned above. The efficiency of the CSD can be taken around 75% while according to Jones[57, 58, 59] the efficiency of the generators and motors could be as high as 95%. Therefore the overall efficiency reduces to 67%. However, when considering the use of less efficient generators and motors (say in the region of 80-85%) this figure could reduce to 50-60%. However given that VS/CF generators are regarded as proven technology and are already in use in a number of aircraft, it is highly unlikely that EJ-200 would include a CSD system. Therefore this option has been discarded.

Jones[57] considers the use of an electrical system for driving air compressors for the ECS of a subsonic transport aircraft. Despite the different application and system philosophy some of the points raised in that report are applicable to the current study. The analysis presented below is largely based on the finding of reference 59.

Driving most generators from a variable speed source, produces power whose frequency and voltage is directly proportional to the driving speed, though according to Jones[59], it is possible to keep voltage constant. Since the amounts of power required to drive the afterburner pump are typically around 2 times present electrical power requirements, the suitability of a Variable Voltage/Variable frequency power for this task should be considered. The conditioned power for other services can then be supplied using solid states converters.

Of the types of motor, which might be considered to drive a pump, all have the basic characteristic of producing speed that is almost proportional to frequency. Thus if a VV/VF generator is connected directly to a motor driving the afterburner pump the

resulting system would have essentially the same speed variation as the engine and/or a direct shaft drive system. However, since the afterburner is usually employed at rated conditions, i.e. maximum TET and HP shaft speed, the frequency and speed variation at those conditions would be minimal. When the afterburner is required at lower power settings, e.g. to accelerate the engine and the aircraft, then the fuel flow required to be delivered depends on the operating conditions of the engine and therefore the speed of the pump may be taken as being proportional to the speed of the engine's HP shaft. Although some form of control may be required a VV/VF system may still be suitable. The author feels that, since Lucas/TRW are already considering the application of VF/VV generators and motors to aircraft fuel pumps, in principal there is no reason why such machines should not be considered for driving the afterburner pump. Although in the past there have been concerns over the control of high output power generators, as required for this study, work outlined in reference 59 made use of 180 KVA permanent magnet machines.

However if precise speed control is required, some form of frequency control must be used. For reasons explained above this frequency control can in fact be relatively crude, since a well-conditioned sine wave power is not necessary. Reference 59 when considered the application of electrically driven air-compressor for the ECS of a transport aircraft proposed the use of a simple control circuit containing 12 semiconductor power switches.

The same reference states that at the time of the study the most promising type of machine for generators and motors was the Inside Out Permanent Magnet (IOP) machine. These machines are claimed to have a specific weight of 0.3 kg/KVA and efficiency 90 to 95%. However prospects for lower specific weight are also likely due to improvements in rare earth permanent magnets.

As far as the converters and controllers are concerned a 150 KVA demonstration unit was mentioned in the same reference to have a mass of 68 kg. The efficiency of the system was in the region of 90-95%.

Assuming that the electrical drive system would be a VV/VF system then the only components required would be two generators and two motors per aircraft. Taking the efficiency of each component to be around 90-95% the overall efficiency would reduce to 81%. Therefore each machine should be rated at 120 KVA. However in the case of the generators the machines should be rated at 180 KVA each, to be able to provide the power required to drive the afterburner as well as the remaining electrical system components. Using a specific weight 0f 0.3kg/KVA for both the generators and the

motors the increase in weight would be around 133kg per aircraft. However this value does not include the weight of the cables.

If however converter/controllers were to be used an extra 163 kg per aircraft should be added to the previous value. This assumes that a system will be used with a mass of 0.68kg/KVA, taken from reference 59.

For the purposes of this study the weight of the system assumed to be 133kg. It is noted, however, that the figure of 133 kg could be further reduced, using smaller generators and motors, which in turn will be overloaded during high power operations.

4.6.5 Definition of Mass Penalties

For the purposes of this study the mechanical system served as the baseline and weight penalties have been defined as the extra weight added to the existing conventional system. Backus[81], suggests that the weight of the turbine would be around 10-20kg, excluding any ducts, valves and other equipment necessary for the implementation of the bleed system This gives a total of 40 kg per aircraft. It should be noted that if the existing engine incorporates a single bleed port –that being the IP port, the final weight and complexity of the system would increase considerably from the predicted value. In the case of an electrical system preliminary weight estimations described above yielded a total weight penalty of 133 kg per aircraft.

4.6.6 Formulation of the Fuel Penalties of an Aircraft System

One method of comparing the merit of different systems is by evaluating the fuel weight penalties associated with each system. In practise, three forms of fuel penalties are present and include the following:

a) Fuel is required to carry the system's fixed weight and the dead weight of the fuel up to the point where it is expended

b) Extra fuel is required to compensate for the increase in fuel flow due to the extraction of off-takes. Once again the extra fuel is required to be carried the up to the point where it is expended

c) If the system in question results to drag penalties, then for a given amount of installed thrust fuel consumption increases requiring therefore extra fuel to be carried.

For the purposes of this study, however, only the first two cases were considered. In general, the flight operation would usually consist of several phases at different altitudes and speeds. The fuel weight penalty for the complete flight would be the sum of the individual fuel weight penalties for each phase. However, the present study considers the penalties during the cruise and/or loiter phases as well as the phases where the afterburner is employed. During loiter and/or cruise the fuel penalties considered include the weight of the system as well as the dead weight of the fuel needed to compensate for the off-take penalties of driving the afterburner pump. During the phases where the afterburner would be employed, and because of the short period of operation, only the fuel penalties due to the off-takes were considered. The following derivation is based on the method given in reference 57.

The formulation of the fuel weight penalty of an aircraft system can be based on the assumption that the drag is of the form:

$$\text{Drag} = \frac{Weight}{Lift / Drag} \tag{4.1}$$

When the aircraft cruises at constant speed and height, the distance dR covered during time dt is given by:

$$dR = aMdt \tag{4.2}$$

where 'a' and 'M' represent the speed of sound and the Mach No respectively. During a period dt the aircraft will consume an amount of fuel given by:-

$$(f + \Delta f_W + \Delta f_D + \Delta f_S)\, dt = -d\,(W_F + \Delta W_F) \tag{4.3}$$

where f = Rate of fuel use for aircraft without system
Δf_W = Fuel use increase due to system weight
Δf_D = Fuel use increase due to system drag
Δf_S = Fuel use increase due to system off-take
W_F = Fuel Weight (without system)
ΔW_F = Fuel weight (for system)

The negative sign in (3) indicates that the final weight decreases with increasing time. Equation (3) can be rearranged and dt substituted into (2) to give:-

$$dR = \frac{-aM * d(W_F + \Delta W_F)}{(f + \Delta f_W + \Delta f_D + \Delta f_S)} \tag{4.4}$$

Specific fuel consumption, c, can be written:-

$$c = \frac{f + \Delta f_W + \Delta f_D + \Delta f_S}{Drag(Total)} \tag{4.5}$$

Without the system onboard Drag could be written as

$$Drag = \frac{Weight}{Lift / Drag} = \frac{W_1 + W_F}{r} \tag{4.6}$$

With the system onboard empty weight is increased by the system weight ΔW_1 and the fuel weight ΔW_F. Taking lift/drag ratio, r, to be constant this effect leads to:-

$$\text{Drag} = (W_1 + \Delta W_1 + W_F + \Delta W_F)/r \tag{4.7}$$

However, the system will also increase the drag directly by an amount ΔD and cause as effective increase in drag to be overcome by the engines due to its power off-take, given by $\Delta f_S/c$. Thus the total drag with the system onboard is given by:-

$$\text{Drag (Total)} = \{(W_1 + \Delta W_1 + W_F + \Delta W_F)/r\} + \Delta D + \Delta f_S/c \tag{4.8}$$

Thus substituting in (5):-

$$c = \frac{f + \Delta f_W + \Delta f_D + \Delta f_S}{(W_1 + \Delta W_1 + W_F + \Delta W_F)\frac{1}{r} + \Delta D + \frac{\Delta f_S}{c}} \tag{4.9}$$

and rearranging

$$f + \Delta f_W + \Delta f_D + \Delta f_S = c/r\,[W_1 + \Delta W_1 + W_F + \Delta W_F + \Delta D r + \Delta f_S\, r/c] \tag{4.10}$$

which can be substitute into (4) giving,

$$dR = -aM\,\frac{r}{c}\left[\frac{f + \Delta f_W + \Delta f_D + \Delta f_S}{(W_1 + \Delta W_1 + W_F + \Delta W_F + r\Delta D + \frac{r\Delta f_S}{c})}\right] \tag{4.11}$$

Integration of (11) yields

$$R = aM\frac{r}{c}\ln\left(\frac{W_{FO} + \Delta W_{FO} + W_1 + \Delta W1 + r\Delta D + \frac{r}{c}\Delta f_S}{W_1 + \Delta W_1 + r\Delta D + \frac{r}{c}\Delta f_S}\right) \tag{4.12}$$

where W_{FO} = Fuel weight to fly range R(without system)

ΔW_{FO} = Addition fuel weight required with system

Without the system (12) reduces to,

$$R = aM\frac{r}{c}\ln\left(\frac{W_{FO} + W_1}{W_1}\right) \tag{4.13}$$

However since the range in (12) and (13) are the same, it follows that the logarithmic terms are also equal, thus:-

$$\frac{W_{FO} + W_1}{W_1} = \left(\frac{W_{FO} + \Delta W_{FO} + W_1 + \Delta W1 + r\Delta D + \frac{r}{c}\Delta f_S}{W_1 + \Delta W_1 + r\Delta D + \frac{r}{c}\Delta f_S}\right) \tag{4.14}$$

Rearranging and manipulating (14) yields:-

$$\Delta W_{FO}\,(\Delta W_1 + \Delta Dr + \Delta f_S\, r/c) = W_1\,(\Delta W_F) \tag{4.15}$$

or

$$\frac{W_{FO}}{W_1} = \left(\frac{\Delta W_{FO}}{\Delta W_1 + r\Delta D + \frac{r}{c}\Delta f_S}\right) \tag{4.16}$$

However W_{FO}/W_1 can be directly obtained from (13) as follows:

$$\frac{R}{aM\frac{r}{c}} = \frac{c}{r}t = \ln\left(\frac{W_{FO}}{W_1} + 1\right) \tag{4.17}$$

where 't' is the time taken to cover range R, then:-

$$\frac{W_{FO}}{W_1} e^{ct/r} - 1 \tag{4.18}$$

This finally gives the increase of fuel required by the system:-

$$\Delta W_{FO} = (\Delta W_1 + \Delta Dr + \Delta f_S \, r/c) \left(e^{ct/r} - 1\right) \tag{4.19}$$

The total fuel used by the aircraft, with the system fitted, is given by:

$$W_{FO} + \Delta W_{FO} = (W_1 + \Delta W1 + \Delta Dr + \Delta f_S \, r/c) \left(e^{ct/r} - 1\right) \tag{4.20}$$

The fuel weight increase caused by the various components can be taken directly from (20) as follows:

Increase in fuel weight due to system weight (ΔW_{FO}) = $\Delta W_1 \left(e^{ct/r} - 1\right)$ (4.21 a)

Increase in fuel weight due to system drag (ΔW_{FO}) = r $\Delta D \left(e^{ct/r} - 1\right)$ (4.21 b)

Increase in fuel weight due to off-take (ΔW_{FO}) = $\Delta f_S \, r/c \left(e^{ct/r} - 1\right)$ (4.21 c)

Where c = Basic specific fuel consumption (lb/lbf/hr)
r = Lift/drag ratio
D = Increment in drag (lbf)
f_S = Increment in fuel flow due to off-take(lb/hr)
t = Time period considered (hr)

Provided the required data is available equations (4.21 a, b, c) can be used to directly calculate the fuel weight penalties of a system. Alternatively the method developed and described in the following paragraph can be used for use in rapid estimation of fuel weight penalty of an aircraft system in the case of the EJ 200 engine.

4.6.7 Method for Use in Rapid Estimation of Fuel Weight Penalty of an Aircraft System for the EJ200 Turbomatch Model Engine

In order to use equations (4.21 a-c) for the system fuel penalty evaluation several basic parameters are required. It is obviously best if these can be accurately calculated or measured for the case concerned. However, if selections of types of systems are being made during the design stage this is not be always possible and for this reason the author decided to incorporate and present a method that allows for a simple and rapid estimation of the fuel penalties involved. As stated in the previous section direct drag effects have been ignored. Instead the method accounts for the indirect drag effects of system weight (i.e. increase in lift dependant drag) and off-take penalties. It should be noted that sfc penalties only apply if the thrust lost imposed by an off-take is

recovered at the expense of increased TET. Otherwise the system weight penalties should be considered alone. This is on the basis that although an off-take would introduce unavoidable sfc penalties, it would also impose thrust penalties. Since sfc is defined as fuel flow per unit thrust, it becomes apparent that the trust penalty would imply a reduction in the absolute fuel flow. However if the thrust lost is recovered, the fuel flow would increase , thus leading to an off-take fuel penalty.

4.6.7.1 Increment in Fuel due to an Off-take

According to equation (21 c) the fuel penalty of an off-take would be a function of the basic sfc, aerodynamic efficiency (lift/drag ratio), increase in fuel flow due to off-take and finally time period considered. Thus, for given flight conditions (i.e. speed, altitude, lift/drag ratio) sfc and time are known. It is therefore possible to rearrange equation (21 c) so that the fuel penalty is expressed in terms of sfc penalty, lift/drag ratio, and finally time period considered. Figures 4.16-4.17 plot the fuel penalty as a function of sfc penalty for a number of flight conditions.

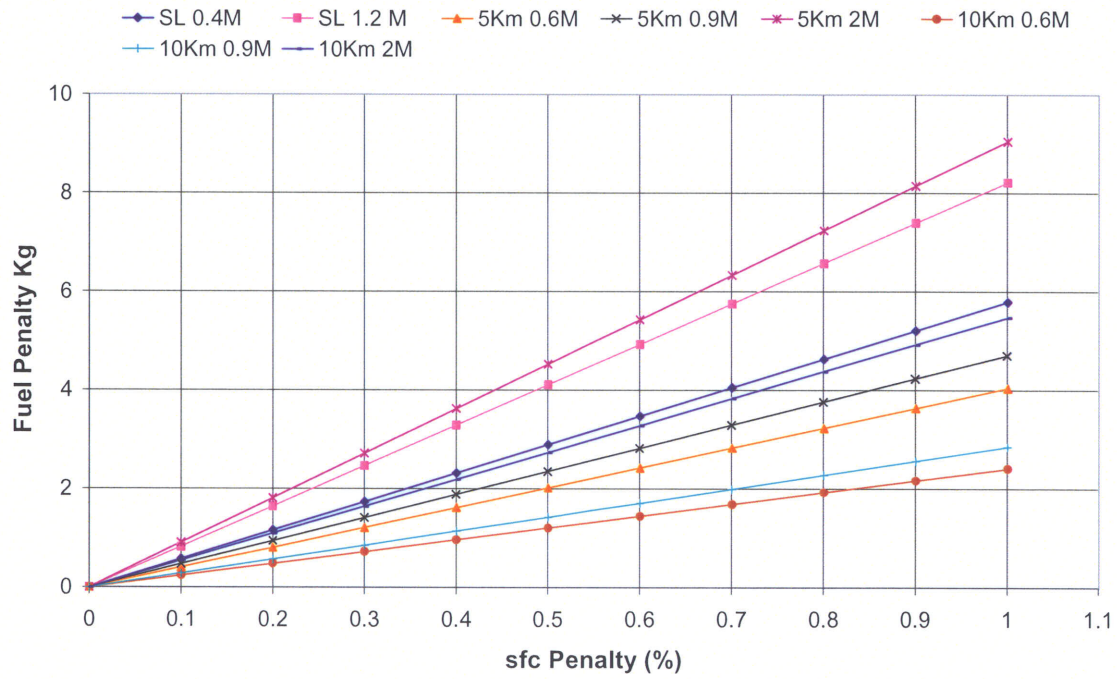


Figure 4.16: Fuel Penalty for 1 Minute of Afterburner Operation

Due to the limited time of operation of the afterburner the penalties are to time. Therefore, for a given sfc penalty at constant flight conditions the fuel weight penalty over a period of 5 minutes will almost equal five times the penalty imposed during 1 minute of operation. Provided that the basic fuel flow is also known, Figures 4.16 and

4.17 can be further simplified by expressing all the parameters involved in a non-dimensional form. More specifically, for given sfc penalties and constant period of operation the fuel weight penalty can be expressed as a percentage penalty of the basic fuel consumed (i.e. no-off-take) over the time period considered. The results are plotted in Figure 4.18, from which it can be seen that that the penalties merged to a single line.

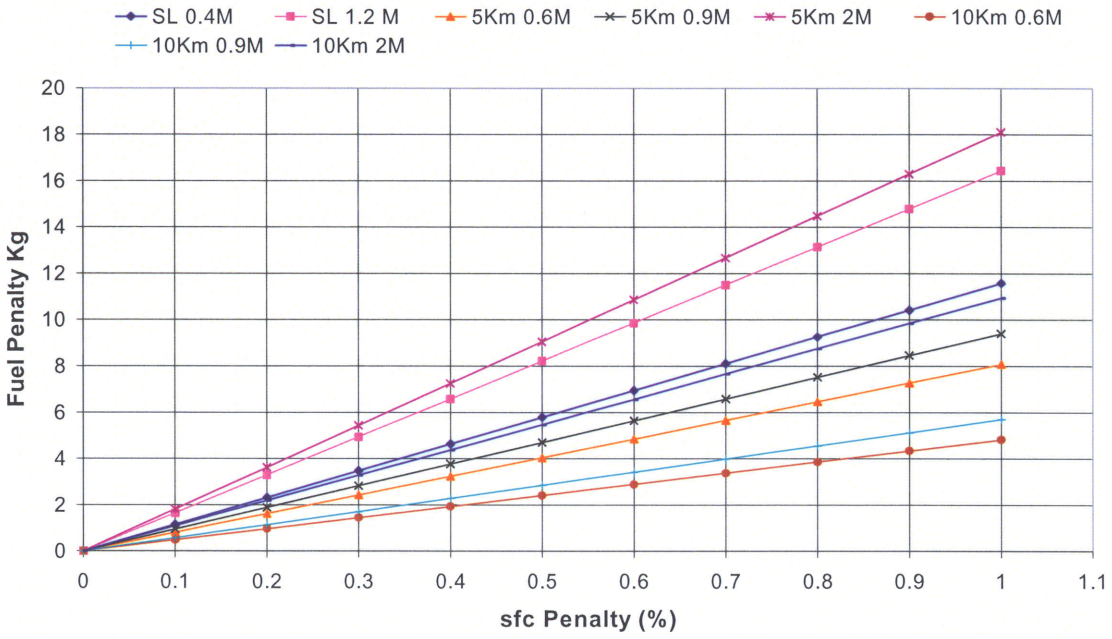


Figure 4.17: Fuel Penalty for 2 Minutes of Afterburner Operation

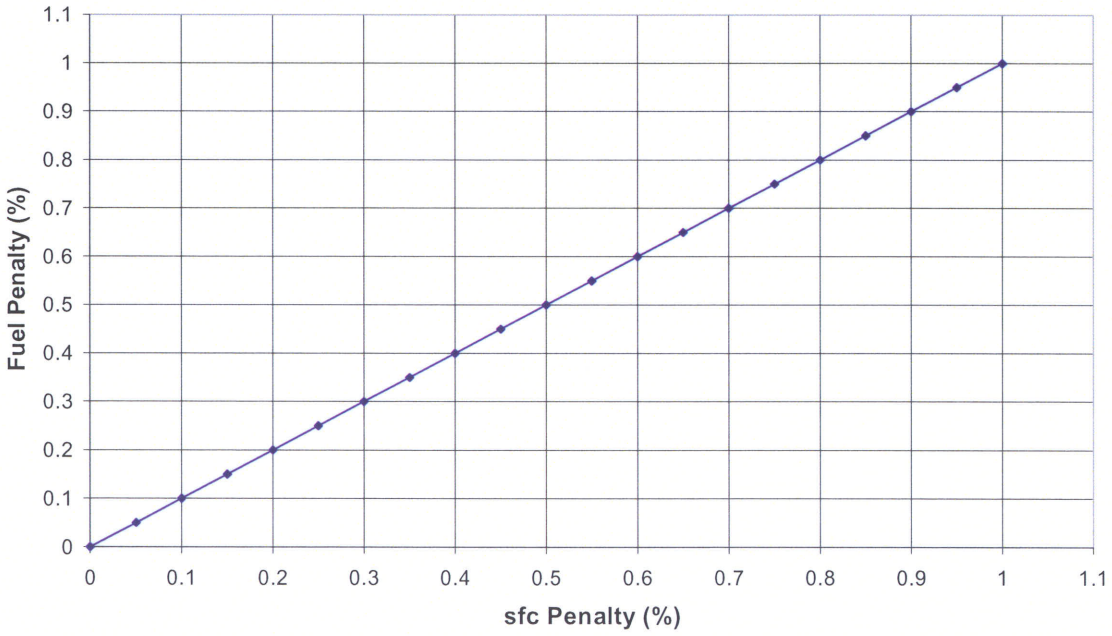


Figure 4.18: Percentage of Fuel Flow Penalty for 1 Minutes of Afterburner Operation

4.6.7.2 Increase in Fuel Weight Due to System Weight and Off-take Penalty (Cruise/Loiter Conditions)

From equation (19), the increase of fuel required by a system is given by:-

$$\Delta W_{FO} = (\Delta W_1 + \Delta Dr + \Delta f_S r/c) \left(e^{ct/r} - 1\right)$$

During cruise and/or loiter extra fuel is required:-

a) To carry the system's fixed weight

b) To compensate for the increase fuel flow when the afterburner is deployed

c) To carry the dead weight of the fuel up to the point where it is used

The method described in the previous paragraph can be used to calculate the fuel weight penalty due to an off-take at different flight conditions and periods of operation. Thus, the individual penalties for supplying secondary power to drive the afterburner pump can be added to give the total increment in fuel required to run the pump. This value can be then added to the fixed weight of the system. Equation (4.19) can be rearranged:-

$$\Delta W_{FO} = (\Delta W_1 + \Delta f) \left(e^{ct/r} - 1\right) \quad (4.22)$$

Where ΔW_{FO} =Increase in fuel during cruise and/or loiter
ΔW_1 = Weight of the system
Δf = Fuel weight increase due to system off-take
c = sfc during cruise and/or loiter
r = Lift/drag ratio during cruise and/or loiter
t = Time spent at cruise and/or loiter

Equation (22) can be written:

$$\frac{\Delta W_{FO}}{\Delta W_1 + \Delta f} = \left(e^{ct/r} - 1\right) \quad (4.23)$$

Hence the extra fuel needed during cruise and/or loiter can be expressed in a non dimensional form depending on the fixed weight of the system, the fuel penalty of the off-take when the system is in use, the time spent in cruise and the flight conditions.

For given flight conditions, i.e. speed and altitude, sfc is uniquely related to lift/drag ratio and therefore a family of lines can be produced by using equation (23), to express the fuel weight penalty as a ratio of the system's weight and fuel weight penalty due to the system's off-take. Such lines are plotted in Figure 4.20, while Figure 21, plots the c/r ratio for representative flight conditions and lift/drag ratios. According to references 76 and 78, in the case of military aircraft lift/drag ratio would vary between 4-9 during cruise and 6-9 during loiter. In general values of r between 6-7 would give very good results.

The total fuel weight penalty would be the sum of the extra fuel carried during cruise and the fuel used to compensate for the off-take penalty when the system is employed, thus:

$$\Delta W_{FO}\,(\text{Total}) = (\Delta W_{FO}\,(\text{cruise}) + \Delta f) \tag{4.24}$$

And the maximum take off penalty of the aircraft would be:

$$\Delta W_{\text{take-off}}\,(\text{Total}) = (\Delta W_{FO}\,(\text{Total}) + \Delta W_1) \tag{4.25}$$

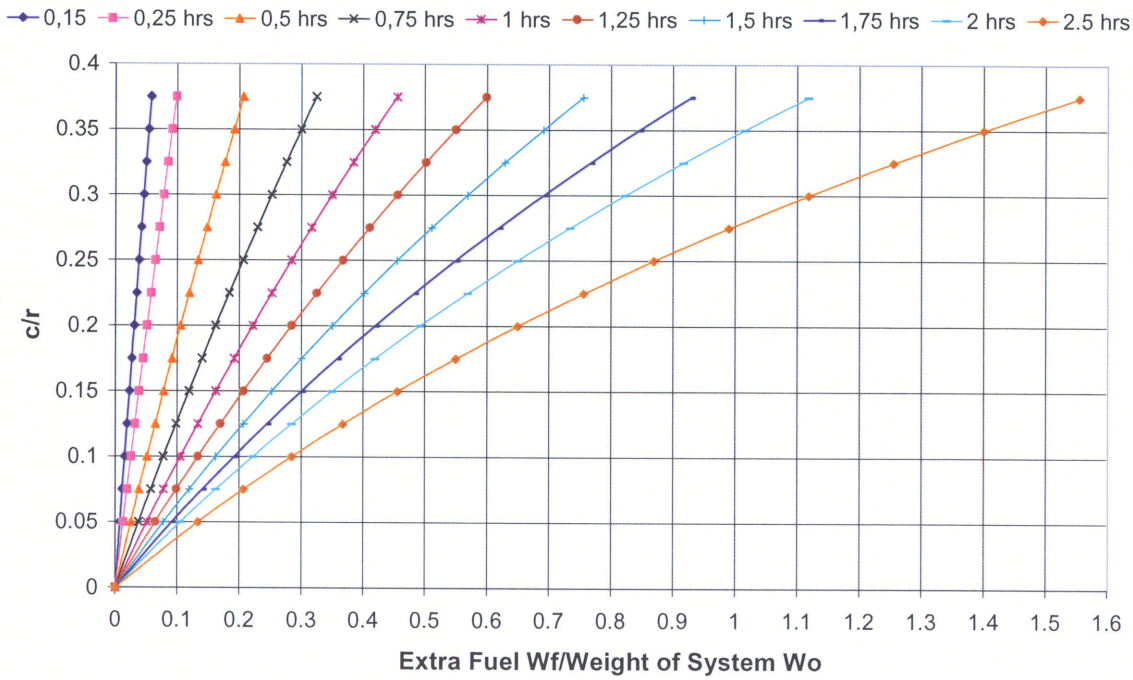


Figure 4.20: System Fuel Weight Penalty Ratio

Once the off-takes are known engine performance maps such as figure 4.9 and 4.10, can be used to find the sfc and thrust penalties at various flight conditions. Thrust penalties can be used with Figure 4.11 to determine the increase in TET required to recover the thrust lost.

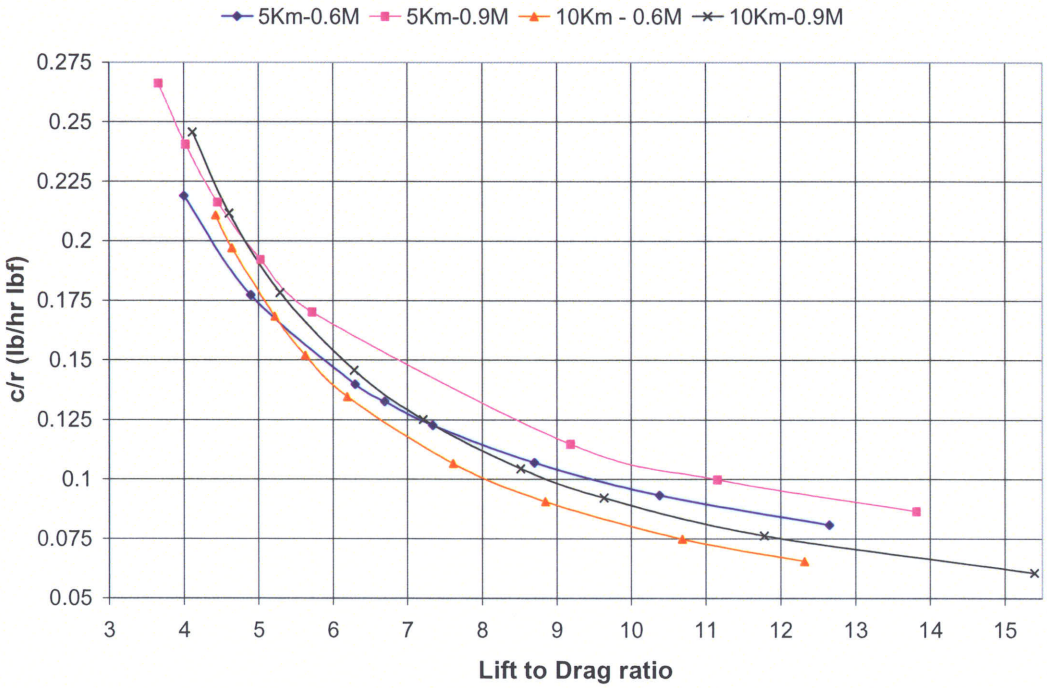


Figure 4.21: c/r Ratio for Typical Cruise and Loiter Conditions

Specific fuel consumption penalties to calculate the direct effects of off-takes on fuel usage at various flight conditions and periods of operation.

Knowing the flight conditions during cruise and loiter, Figure 4.21 can be used to find the sfc/(lift/drag) ratio (c/r). With c/r ratio, system weight and the previously found off-take penalty known, Figure 20, can be used to find the fuel weight penalty during cruise and or loiter. The total fuel penalty will be the sum of the off-take penalty and the cruise weight penalty.

Combining the effects on thrust, TET, sfc, system weight and finally fuel weight penalties, a preliminary comparison can be made between different systems.

4.6.8 Fuel Penalties Imposed by the Power Requirements of Driving the afterburner Pump

Table 4.1 summarises the increments in fuel used due to the sfc penalty imposed by the extraction of secondary power to drive the afterburner pump over a period of 11 minutes and at various flight conditions. A mechanical drive would require an extra 5.5 kg of fuel relative to a no off-take case. This penalty increases to 9.45 kg and 22.2 kg in the case of the electrical drive (65% efficiency) and bleed system respectively. Considering the use of an air turbine receiving air from the IP port and then expanding it to ambient the fuel penalty reduces to 18 kg.

Duration (minutes)	Mach No	Electrical System 65%	Mechanical System 95%	Bleed system Bleed from LP Port
0.5	SL-0.4 &1.2	1.7 kg	0.95 kg	3 kg
2	5000m-0.6	1.4 kg	0.8 kg	4 kg
2	5000m-0.9	2 kg	1 kg	5.5 kg
1	5000m-2.0	1.4 kg	0.8 kg	3.5 kg
2	10000m-0.6	1.2 kg	0.9 kg	2 kg
2	10000m-0.9	1 kg	0.6 kg	2 kg
1	10000m-2.0	0.75 kg	0.5 kg	2.2 kg
11	Total	9.45 kg	5.55 kg	22.2 kg

Table 4.1: Afterburner Off-Take Penalties

4.6.9 Fuel Penalties in Cruise and Loiter Due to the Weight of the Systems and the Extra Fuel Needed for Driving the Afterburner Pump

Table 4.2 summarises the increments in fuel used during cruise and/ or loiter due to the extra weight of the systems and the extra fuel needed to be carried relative to a no off-take case. Assuming that the aircraft would cruise at 10,000 meters and 0.9 Mach for 1.5 hours while having an aerodynamic efficiency (lift to drag ratio) of 6, the increment in fuel due to the mass of the system and the off-take penalty when the afterburner is in use would be 40 kg for the electrical drive. Considering the bleed system, the increment in fuel would be around 18 kg. Adding to these values the off-take penalties would yield a total of 49.5 kg of extra fuel for the electrical system and 39.7 kg of fuel in the case of the bleed system.

L/D (r)	Hrs	Electrical Sys Fuel Penalty (kg)	Bleed System LP Port Fuel Penalty (kg)	Electrical System Total Penalty (kg)	Bleed System Total Penalty (kg)
	1	24.48	6.8	34	29
	1.5	40.0	18.4	49	39.7
6	2	50.4	14	60	36
	2.5	64.8	18	74	40
	1	19.152	5.32	29	27
	1.5	28.8	8	38	32
7	2	40.32	11.2	50	33
	2.5	51.84	14.4	61	36

Table 4.2: Total Fuel Penalties

From the foregoing it can be seen that although the bleed system would impose higher sfc penalties than the electrical system, the total fuel penalty is more sensitive to the weight of the system leading, thus, to lower penalties in the case of the air turbine. However, as mentioned before the weight estimation in the case of the bleed system is overoptimistic and includes the weight of the turbine alone. Therefore the total penalty in the case of the turbine will become larger than the penalty of the electrical drive. Moreover, the introduction of ducts passing across the bypass stream would introduce pressure losses and thus further engine performance penalties. Unlike the afterburner off-take penalties, these penalties will be experienced over the entire mission and not only during the deployment of the afterburner.

Since it is more likely that the EJ-200 engine incorporates a single port, that being the IP port, it would be more convenient to consider a bleed system that receives air from the existing port. In that case the increment in fuel due to the effects of the off-takes would be around 18 kg. According to reference 81, however, the mass and size of the turbine would increase and therefore the total fuel penalty will be similar to the penalty imposed by the use of a turbine receiving air from the LP port and expanding it to ambient. Although the mass and size of the turbine would increase in the case of the turbine receiving air from the IP port the overall weight and complexity of the system would be significantly reduced. Assuming that the weight of the turbine would increase to around 25 kg and that the aircraft would cruise at the above-mentioned conditions for 1.5 hours, the total fuel penalty in the case of a turbine receiving air from the IP port would be around 37.14 kg. Once more this figure takes into account the weight of the turbines alone and excludes the additional weight of the ducts, valves, etc.

4.7 Previous Studies Considering Application of All Electric Accessory Drive Systems to Military Aircraft.

The only reference obtained, which specifically addresses the all-electric accessory drive systems implications on military engine design and performance, is by G B Toyne[63] of Rolls Royce, Bristol. The paper examined some of the implications, particularly those affecting the design and performance of the engines, of adopting the all-electric systems concept. The concept implies that all aircraft accessory power off-take is by mechanically driven electrical generators. Within the aircraft, electric motors drive air compressors for the environmental control system. Aircraft control surface actuators are electrically powered, either directly or through electrically driven hydraulic pumps. Although the paper focused on a subsonic transport aircraft, reference was made to a supersonic combat aircraft, though the analysis was qualitative rather than quantitative.

It was concluded that the concept is worthy of consideration for large subsonic transport aircraft, since it offers the prospect of simplification and weight reduction in the engines. For supersonic combat aircraft, however, the concept was not recommended, mainly because of the relatively higher levels of shaft power off-take and the additional handling problems that may be introduced. It should be noted that the paper did not considered the use of an air turbine for driving any of the accessories and the study focused on comparing the conventional mechanical system to an electrical drive system.

The paper argued that although some saving in engine weight could be expected if electrically driven accessories were adopted, the potential savings would be substantially less in the supersonic combat engine than in the case of the subsonic transport aircraft engine. According to the authors, the reasons that led to this decision were as follows:

a) Air Bleed system for the ECS. Unlike the civil case, for the combat aircraft there is not such a strong case for complex air bleed systems. There is therefore less to be gained by deleting the relatively simple air bleed systems used on military engines. Moreover, the study claimed that existing starting and handling bleeds would have to be retained.

b) Engine gearbox design. Since the study recommended that the afterburner fuel pump should continue to be engine driven, there was less scope for simplifying the engine gearbox.

Although the above concerns seem to be reasonable, the study was qualitative rather than quantitative and therefore some of the points raised are questionable. As far as the ECS is concerned the study is probably right on that in the case of a supersonic combat aircraft, the air bleed for ECS purposes is a lower percentage of the total accessory power load and therefore the saving in fuel consumption by introducing electrically driven compressors is in fact minor in comparison with the conventional system. Findings of previous studies [122-123] performed at Cranfield University support this decision. However, these findings merely mean that electrical generators cannot be used to drive the rest of the airframe and engine accessories and systems. Indeed Jones[70] argues that the All Electric (accessory) Engine concept, unlike the AEA and/or MEA concepts, does not assume that present bleed air power extraction is necessarily replaced.

With respect to the claim that air starting should be retained, although no actual demonstrations have been performed, calculations have shown that the generator capacity required per engine to drive the afterburner pump and the utility needs is more than sufficient when acting as a starter to start an engine under all conditions [58, 59, 60].

As far as handling bleed is concerned, although a number of engines incorporate this technique of controlling the upward movement of the compressor running line towards surge, alternative methods can be used instead. Indeed Janes [123], states that the front stages of the HP compressor in the EJ-200 engine are of variable geometry, while no reference is made to the extraction of handling bleed. Even Reference Toyne [62], which identifies handling as a potential problem, suggests that provided the necessary degree of variable geometry exists (e.g. variable inlet guide vanes), modern electronic control systems could be used to control the movement of the running line. Moreover, the movement of the running line can be effectively controlled via a variable geometry nozzle, which is a common feature of all engines incorporating afterburner.

Finally the study concluded that due to the relatively high power requirements of the afterburner pump, a mechanical drive should be retained for this pump. However, no results concerning engine performance penalties were presented in order to justify this conclusion.

In the contrary the findings of the present study suggest that the effects of system efficiency on engine performance were indeed minimal and the main, if not the only, drawback of an electrical system for driving the afterburner pump would be its weight.

Although Jones[70] expressed his concerns about the size and weight of the generator, especially when considering the short period of operation, no reference was made on the effects on engine performance.

4.8 Concluding Remark

In terms of engine performance implications, there is not such a strong case for not using an electrical drive for the operation of the afterburner pump. Although performance penalties would be higher than those imposed by the conventional mechanical system, the difference would be minimal and the performance of the engine will almost be unaffected. The opposite is true however when considering the use of a bleed system. In all cases, higher performance penalties were experienced when comparing the bleed systems to the mechanical and electrical drives. In terms of thrust loss and increase in TET, the difference was more noticeable at high altitude low speed operations where the reduced mass flow through the engine had a direct impact on the performance penalties imposed by the bleed systems.

Although adoption of an electrical drive may introduce handling problems when the engine accelerates, it is likely that these problems can be controlled with the existing variable geometry compressor and nozzle. However, further work is required in this area.

Chapter 5: Development of Aircraft Performance Model & Definition of Baseline Aircraft

The chapter deals with the development and use of an integrated aircraft-engine performance tool. To be able to assess the potential performance benefits of adopting all electric aircraft/engine technologies an aircraft performance model was developed and then integrated with an existing gas turbine performance program. The tool was validated against published data for similar class of aircraft; for the baseline configurations the deviation was less that 1%. The definitions of the baseline aircraft are also provided together with the specifications of the secondary power systems, i.e. power and performance requirements, etc. The performance of the baseline aircraft and engine are presented and studied in terms of mission profiles, aircraft aerodynamic efficiency and drag polar, engine sfc, turbine entry temperatures, etc. The effect of conventional secondary power systems on aircraft/engine performance and mass are compared against clean cases and all/more electric derivatives.

5.1 Introduction

To able to provide an insight into the detailed challenges and possible gains resulting from the adoption of the AEA/E concepts three baseline aircraft with conventional systems were selected. As mentioned before the baselines refer to the Rolls Royce Trent powered Airbus A330, A340 and A380 class of aircraft. Detailed explanations of the reasons that led to these selections were given in the first chapter of this work. In order to accurately and with complete consistency calculate the performance of the baseline aircraft a spreadsheet model was developed that simulates any given mission of an aircraft. The model was built in a manner that allows for different aircraft to be modelled as well as AEA/E versions to be studied by altering appropriate input data such as aircraft mass and geometric details, mission specifications as well as secondary power systems mass and off-take requirements. Turbomatch, an existing engine performance program, was used to develop and run representative models of the baseline engines. The models allowed complete freedom to vary secondary power off-takes as required for this study.

In addition to the engine performance data provided by Turbomatch, aerodynamic information and typical flight profiles were required. A number of references were reviewed and where necessary different semi-empirical methods were compared and/or

combined to derive the aerodynamic information required. Both induced drag and profile drag coefficients were derived and the integrated performance of the aircraft are summarised and presented in the form of payload diagrams.

5.2 Model Requirements

A set of requirements were developed and implemented into the aircraft engine performance model. These include the following:

- The model should calculate aircraft performance data such as lift and drag coefficients, distance for take-off, etc. from publicly available information on the geometry and mass of the aircraft

- The model should calculate the fuel usage, time elapsed and distance covered for the baseline and derivative aircraft performing a given mission, including contingency and reserve fuel as defined by the user

- The model should allow for changes in secondary power systems mass and off-takes

- The model should allow for the modelling of different aircraft and engines

- For specified maximum take-off weight, payload and fuel load the model should compute the range of the aircraft

5.3 Mission Profile

To qualify as airworthy all civil transport aircraft must comply with airworthiness regulations. To ensure a minimum standard of safety they must also meet operational requirements with regard to fuel planning. Such specific requirements exist in FAR 121[125] and JAR OPS 1[126]. The exact requirements depend on the operator, type of aircraft, route planned and the existence/availability of alternate airports. For the purpose of this study, the mission profiles and fuel planning specifications meet the requirements for international flights as defined by FAR 121, JAR OPS 1 and the recommendations provided by ICAO.

For the purpose of fuel planning the flight profile of the aircraft can be divided into several phases for both the main profile and the reserves profile.

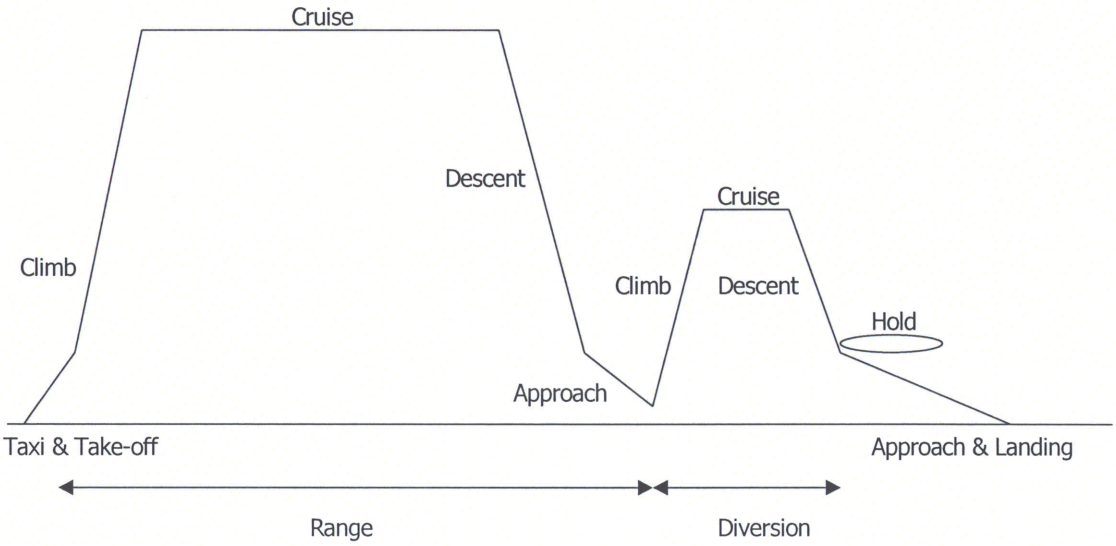


Figure 5.1: Typical Mission Profile

These segments are shown in Figure 5.1 and include:

- Start up and taxi-out
- Take-off and initial climb to 1500ft
- Climb from 1500ft to cruise altitude
- Cruise at selected speed and altitude including any stepped climb required
- Descent to 1500ft
- Approach and landing
- Taxi–in

Typical reserves include:

- Missed approach and climb to 1500ft
- Climb to diversion cruise altitude
- Cruise at speed for maximum endurance
- Descent to 1500ft
- Hold at 1500ft

- Approach and landing at alternate airport
- Contingency, estimated as a percentage of fuel used or time of flight in the main mission profile

Contingency fuel is carried to compensate for items such as:

- Deviation from the expected fuel consumption data
- Deviation from predicted meteorological conditions
- Deviation from planned routings and/or cruising levels/altitudes

The allowances for start up, taxi, take-off, landing and diversion distances will vary with each route to be flown by the aircraft. To enable a comparison between various aircraft a standard set of allowances forms part of the aircraft initial specification. Such a typical set of allowances and reserves is shown in Table 5.1.

	Domestic Flights	International Flights
Allowances		
Start and Taxi-out	9 minutes	12 minutes
Approach and Landing	6 minutes	6 minutes
Taxi-in	5 minutes	5 minutes
Reserves		
Block Fuel (Contingency)	45 minutes	Allowances of 5%
Diversion	200nm	200nm
Hold		30 minutes
Diversion Approach and Landing Overshoot	6 minutes	6 minutes
Assumed: ISA Conditions, Zero Wind		

Table 5.1: Typical Allowances and Reserves Used for Fuel Planning
(Reproduced from Reference 126)

Although reserve and contingency fuel are not to be consumed at normal operating conditions they must be carried at all times and therefore they have an impact on the weight, aerodynamic and performance characteristics of the aircraft. Adoption of all or more electric technologies will affect the mass of the aircraft and the performance of the propulsion system. Although the main effect would be at the cruise section, improved engine performance and mass characteristics would also require reduced reserves and contingency fuel. This reduction would have a snowball effect of the maximum take off weight of the aircraft and its performance at the main mission profile. For these reasons it is necessary to include these flight phases into the performance model.

5.4 Weight Breakdown

The mission profile shown in Figure 5.1 can also be used to define the weight breakdown of the aircraft at the various stations of interest. The simplest approach is to start with the Operating Empty Weight (OEW) of the aircraft and to work backwards through the mission profile adding the various weight elements. The OEW is the weight of the aircraft including structures, systems, furnishing and any additional equipment required for operation, i.e. safety equipment, blankets, trays, galleys, etc. The payload is added to the OEW to give the Zero Fuel Weight (ZFW). The weight of the aircraft after landing to the alternate airport equals the ZFW plus the contingency fuel. To this is added the weight of the holding fuel, alternate fuel and trip fuel to give the Break Release Weight (BRW) or Maximum Take-Off Weight (MTOW). The Ramp Weight equals the MTOW plus the start up and taxi fuel weight.

5.5 Payload-Range Diagram

The performance of a given aircraft can be summarised in the form of a payload – range diagram as shown in Figure 5.2. Normally three operating conditions are required to define the payload range diagram:

- Range for maximum take-off weight – maximum payload
- Range for maximum take-off weight – maximum fuel load
- Ferry range for maximum fuel load – zero payload

Range for maximum take-off weight – maximum payload: This point will be defined by the maximum take-off weight of the aircraft and either the maximum volumetric payload or the maximum zero fuel mass limited payload. In the first case the available volume would limit the maximum payload that includes the following:

- Mass of passengers in high density layout, assuming standard passenger weights; typically 75 kg per passenger
- Mass of passenger baggage, assuming standard baggage weight; typically 25 kg per passenger
- Remaining hold volume used for cargo

The maximum zero fuel mass limited payload is normally limited by structural requirements.

Range for maximum take-off weight – maximum fuel load: This point is defined as the range flown with maximum fuel and the payload required to maintain the aircraft's maximum take-off weight. The maximum fuel load would be limited by the capacity of the fuel tanks available. Most civil transport aircraft store fuel in the wing. Additional fuel can be stored in the wing box and the tail structures. The payload is calculated as:

Payload = Maximum Take Off Weight – Maximum Fuel – Operating Empty Mass

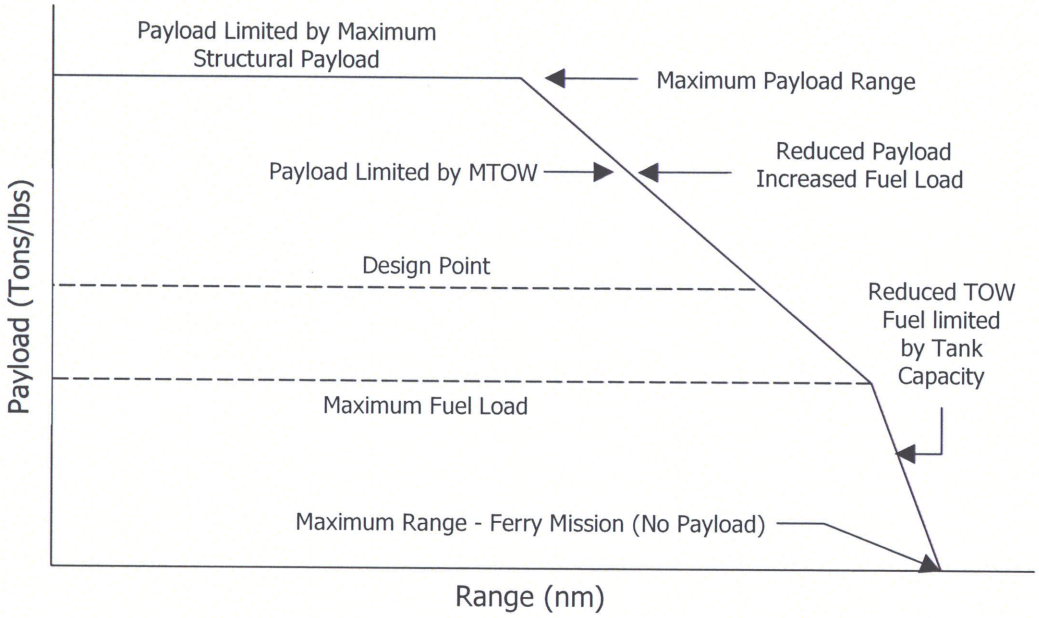


Figure 5.2: Typical Payload – Range Diagram

Ferry range for maximum fuel load – zero payload: This point corresponds to the range flown with maximum fuel and zero payload. In this case the weight of the aircraft is less than the maximum take-off weight. For this reason the lift to drag ratio (also known as aerodynamic efficiency) of the aircraft would be different from that in the two previous cases.

The aircraft specifications will define the design range and corresponding design payload. The design payload on a civil aircraft is usually less than the maximum payload and normally is placed between the first two points on the payload range diagram, Figure 5.2.

5.6 Model Structure

The model structure is based on a modular layout. This approach makes the spreadsheet model simpler to use and allows the modules to be easily altered and integrated for different design studies such as AEE/A versions and alternative baseline aircraft. The model consist of six different modules, shown below:

1. Input data
2. Mission profile module
3. Atmospheric module
4. Engine data
5. Aerodynamic module
6. Aircraft performance module

The modules were developed separately and then integrated into the main model that computes the aerodynamic characteristics of the aircraft, fuel, distance and time for the climb, cruise, hold and descent segments of the mission. As mentioned before allowances were made for the remaining segments that include start-up, taxi, take-off, landing and taxi out. The module also computes the take-off distance.

The computation occurs sequentially using publicly available data on the general arrangement of the aircraft, MTOW, payload-range diagram and/or mission profile. The computation sequence is shown in Figure 5.3. For a given range the computation of the cruise fuel and distance can only be performed if the time from the Top of Climb (TOC) to the Top of Descent (TOD) is known or vice versa i.e. fuel and time can be computed knowing the distance from the TOC to the TOD. At the start of the calculation neither the distance nor the time are known due to an uncertainty of the climb and descent distance/time. For this reason an estimate, based on flying Mach number and assuming that the distance from TOC to the TOD is the distance between the departure and the destination airport, is used to initialise the solution. The final result is compared to the specified mission and an iterative procedure is performed to determine the correct time spend at cruise.

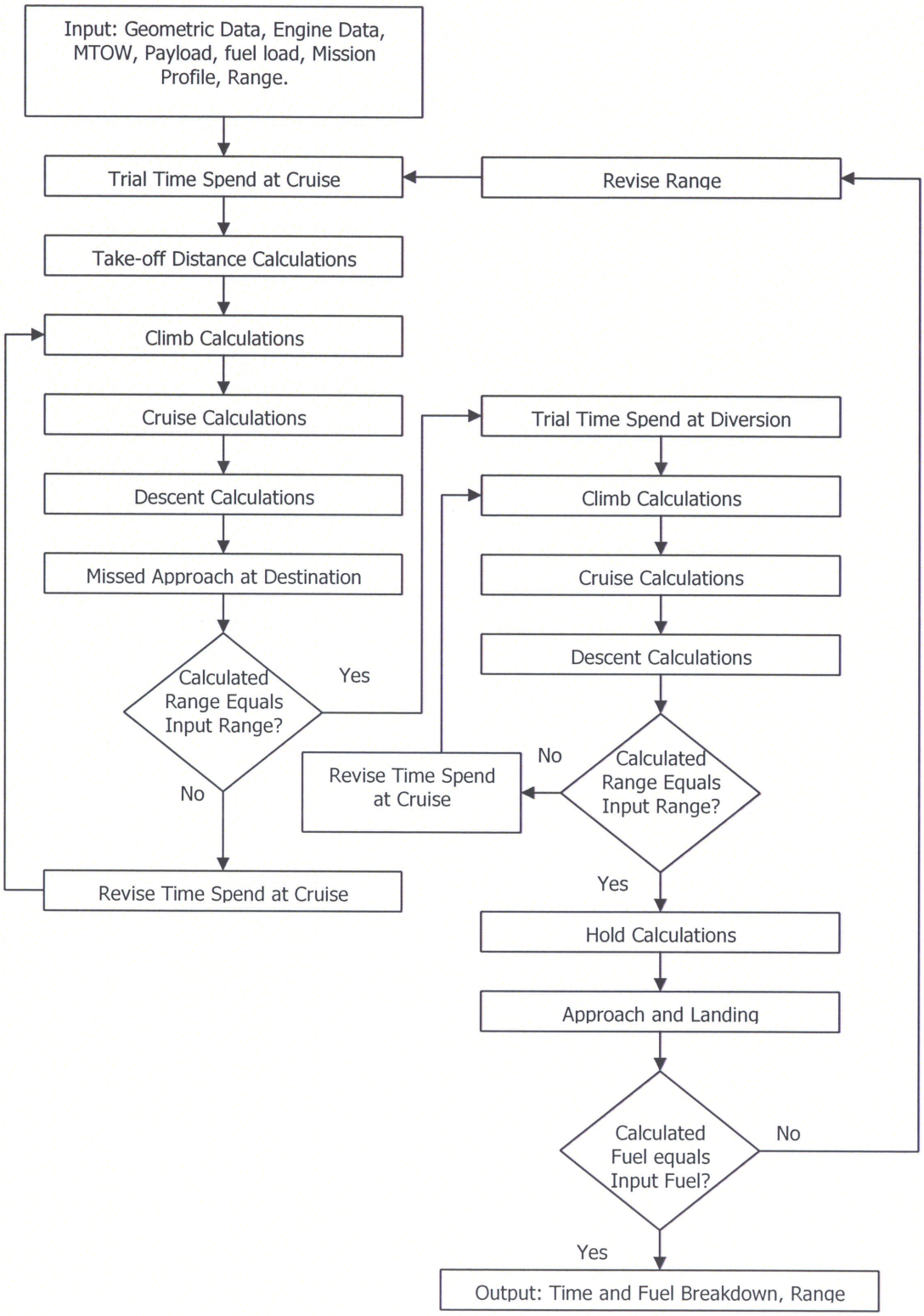


Figure 5.3: Simplified Flowchart of Calculations

As shown in Figure 5.3 the user specifies the MTOW, payload and fuel load as inputs. Although the range of the aircraft is also specified this is an initial trial value used to increase the accuracy of the model and to initialise the solution. In practice, the range is an unknown that is computed by an iterative procedure. Each iterative cycle determines the fuel required for a trial distance. Convergence is achieved when the total fuel load is consumed. In the model this is attained by subtracting the calculated total fuel (mission fuel plus reserves) and the payload from the MTOW, to give a trial OEW value. This value is then compared to the actual OEW of the aircraft and the difference is used to adjust the time spend at cruise and the distance covered, i.e. range. The computation is repeated until convergence within 0.1% of the OEW is achieved. Although the range of the aircraft is used as a trial value for the main mission profile, the user also specifies the diversion range and this is a condition that must be met to ensure the correct amount of reserves.

When the aircraft lands at an alternate airport following the main mission profile, the fuel load on board the aircraft would equal the en route contingency fuel. In this analysis the mission fuel is calculated is such a way that this is always true.

5.6.1 Input Data Module

The required input data comprise the basic information used to define the shape, geometry and required performance of the aircraft. Much of this information must be collected from textbooks such as Janes[128] and Jenkinson[127] as well as other available sources that provide information on the selected aircraft, i.e. aircraft manufacturer web site, etc. In the absence of some information, typical values derived from experience must be used. The data is used by the aerodynamic and aircraft performance modules to compute the aerodynamic characteristics and performance of the aircraft. To allow for a simple and easy to use model the number of input data was kept to a minimum and include:

- Weight Data:
    - MTOW
    - ZFW
    - OEW
    - Payload
    - Fuel Load

- Area Data:
    - Wing plan area
    - Tailplane plan area
    - Fin plan area

- Additional Geometric Data:
    - Wing, tailplane and fin aspect ratio
    - Wing, tailplane and fin tapper ratio
    - Wing, tailplane and fin thickness to chord ratio
    - Wing, tailplane and fin sweep angle

- Engine Data:
    - Installed engine diameter
    - Installed engine length

- Fuselage Data:
    - Fuselage diameter
    - Fuselage length

Most of this data can be found in the references mentioned above. In addition the module is used to calculate further geometric information, such as wetted areas mean and geometric aerodynamic chords and sweep angles at various wing cord stations required to calculate the aerodynamic characteristics of the aircraft.

5.6.2 Mission Profile Module

As mentioned in section 5.3 the flight profile is split in to several phases. The overall mission profile consists of the main mission profile and the reserves profile. The first is the range of the aircraft and is the figure that aircraft manufacturers refer to their payload range diagrams. The overall mission profile is specified by the user and is used by the aircraft performance module to compute the time, distance and fuel consumed at each segment. The mission profile is also used to model the operating conditions of the engines when modeled in Turbomatch. The user also specifies the climb schedule of the aircraft, cruise and speed altitude (including any stepped cruise required), alternate cruise altitude and speed for maximum range and finally descent schedule.

As far as climb from 1500ft to initial cruise altitude is concerned, this is split into the following parts:

1. 1500ft to 10000ft where the speed is restricted to 250 kt equivalent air speed (EAS)
2. Accelerate at 10000ft to 320 kt EAS
3. After this the speed is held constant up to the height at which the cruise Mach No is achieved. Thereafter the speed is kept at constant Mach No up to the initial cruise altitude.

The selected mission profile for the Airbus a340-500 class of aircraft is shown in Figure 5.4.

5.6.3 Atmospheric Module

Vital to any aircraft performance analysis is the need to know the atmospheric conditions at given altitudes and Mach numbers. For this reason the model calculates the international standard atmosphere characteristics (ISA) for the troposphere and lower atmosphere. While ISA is usually adopted the ability to alter the temperature to suit non-standard conditions is provided.

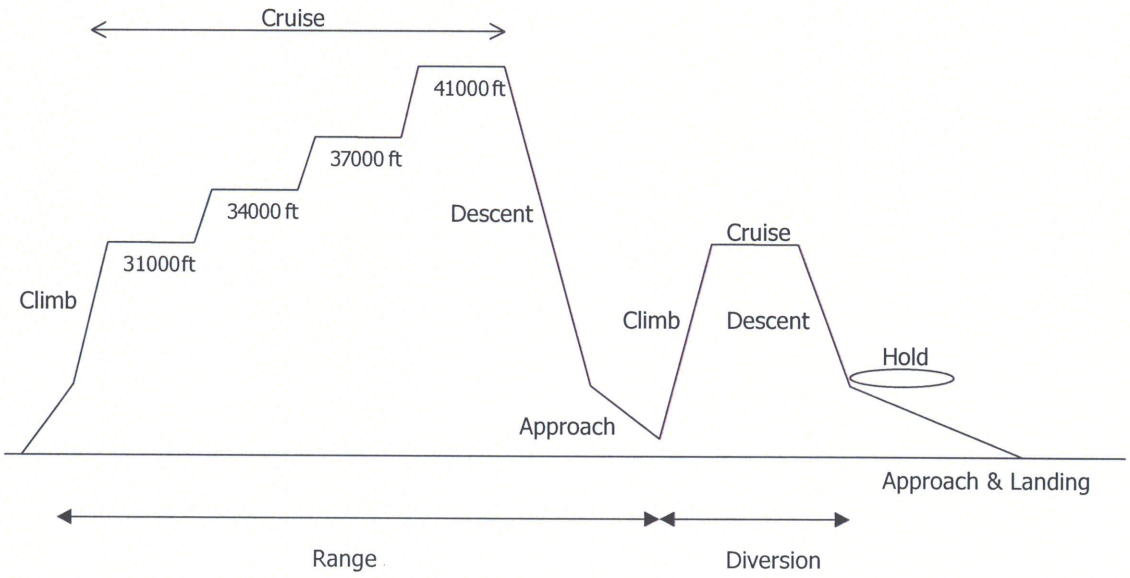
5.6.4 Engine Data Module

Engine data is essential in determining the performance of the aircraft. Typical data include maximum take-off thrust (used by the aircraft performance module to determine time and length required for take-off as well as fuel consumed), maximum climb thrust and sfc (used to determine rate of climb, horizontal distance covered, time to climb and fuel consumed), cruise performance (used to determine fuel consumed) descent performance (used to determine rate of descent, horizontal distance covered, time and fuel consumed). The data is obtained by Turbomatch and after developing and running a model that describes the selected engine.

5.6.5 Aerodynamic Module

The module is used to calculate the aerodynamic characteristics of the aircraft at appropriate flight conditions. Information from the input data, atmospheric and mission

are passed to the aerodynamic model where the drag characteristics of the aircraft are computed in the form of drag coefficients and drag polars profile modules.



Climb Schedule:
Climb to 10000ft at constant EAS of 250 kt
Accelerate at 10000ft to 320kt
Speed is kept constant (EAS=320 kt) up to the height (26500ft) that 0.82 Mach is reached

Figure 5.4: Mission Profile Used for Airbus A340-500 Class of Aircraft

5.6.5.1 Drag Components

The total drag of the aircraft can be considered as the sum of several drag components. In general, there are two distinct ways of describing the total drag. The first way of distinguishing drag components relies on the nature of the forces that are exerted on the body or the aircraft, i.e. either through the distribution of normal pressure or by the skin friction or shear stress at the surface. This method assumes the following drag components:

- Friction Drag or Skin Friction Drag: This is the viscous drag due to tangential forces acting on the surface of the body.

- Pressure Drag or Normal Pressure Drag: Pressure drag is the drag exerted on the body due to the normal pressure acting on the surface of the body.

As shear and normal forces make up the resultant force on the body the sum of friction and pressure drag is the total drag of the body. Pressure drag is further divided into the following components, also shown in Figure 5.4.

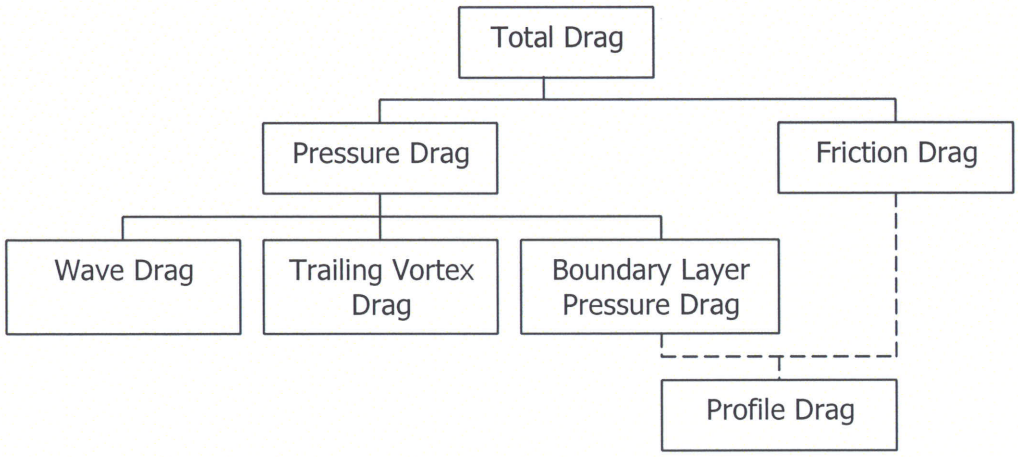


Figure 5.4 Drag Components Based on Nature of Force
(Reproduced from Reference 129)

- Wave Drag or Compressibility Drag: Is the drag associated with the formation of shock waves in high-speed flight. It has both lift and volume/shape components.

- Trailing Vortex Drag or Vortex Drag: This is the drag exerted on a finite wing due to the existence of trailing vortices when the wing is producing lift.

- Boundary Layer Normal Pressure Drag: This is the pressure drag due to the existence of boundary layer and depends on the shape of the body.

- Profile Drag: Is the drag arising due to the presence of boundary layer on a body when it is moving through a fluid. This is equal with the friction and boundary layer pressure drag.

- Interference Drag: According to ESDU 97016, aerodynamic interference between adjacent airframe components arises due to the "coalescing of the pressure fields

of the individual components and the confluence of their boundary layers in junction regions". As in the case of profile drag, interference drag is attributed to both pressure and friction drag.

The second method of distinguishing drag components relies on the fluid mechanics that cause drag. In this case, the drag components are divided into two parts, a lift dependent drag and a drag component that may be considered to be independent of lift and is known as zero lift drag, Figure 5.5. The zero lift drag is further divided into datum and spillage drag. A datum condition is defined in terms of a specified lift coefficient and mass flow into the engines. Any deviation from the datum mass flow into the engine will alter the drag characteristics of the aircraft and this is defined by the spillage drag. Deviations from the datum lift coefficient are accounted for by the lift dependent drag.

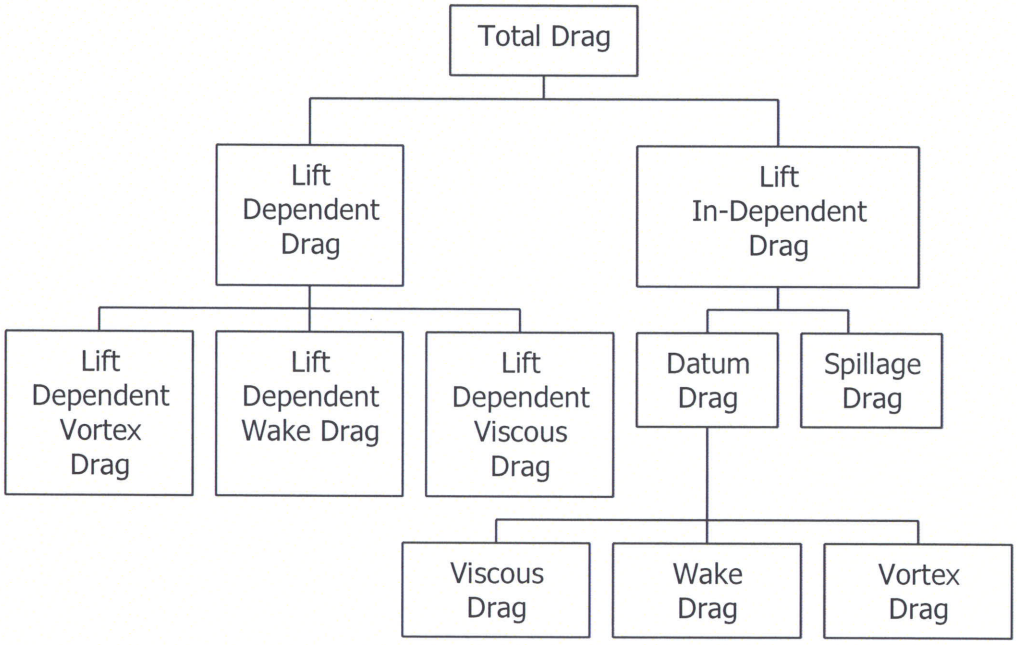


Figure 5.5: Drag Components Based on Fluid Mechanics Theory
Reproduce from Reference 130

For the purpose of calculating the aircraft drag the second approach of distinguishing drag components is usually adopted. Although the drag polar can be represented by an equation in several ways it is essential that the equation always include the two main components of drag, one dependent on lift and the other independent of lift. Therefore,

the total drag coefficient can be expressed as the sum of two terms, one constant and the other a function of the lift coefficient C_L as shown below:

$$C_D = C_{D_O} + C_{D_I} \tag{5.1}$$

where C_D is the total drag coefficient, C_{D_O} is the zero lift drag coefficient and C_{DI} is the lift dependent or induced drag coefficient. The profile drag shown in Figure 5.4 is largely lift independent drag. According to Bowes the profile drag is the sum of the minimum profile drag and a much smaller term that is lift dependent. Minimum profile drag exists due to friction, pressure, interference and roughness. Therefore, if the first method of classifying drag components is to be adopted the zero lift coefficient shown in the equation above will equal the minimum profile drag. The vortex drag, on the other hand, is the main component of the lift-induced drag. To this is added the remaining friction and pressure drag component of the profile drag. It can be shown that the lift dependent drag coefficient of the complete aircraft can be described by the following expression:

$$C_{D_I} = KC_L^2 \tag{5.2}$$

where K is the lift dependent factor. Using the above expression the following well-known drag expression is obtained:

$$C_D = C_{D_O} + KC_L^2 \tag{5.3}$$

It has been shown by experiments that equation 5.3 is a reasonably accurate expression for most aircraft except when the lift coefficient is either very small or so large that the wing is close to the stalling conditions. At such conditions the effect of wing camber will result in the minimum drag being slightly different to C_{D_O}, shown in Figure 5.6.

Despite its limitations this expression is known to provide reasonably accurate results and is widely accepted for the representation of the aircraft's drag polar. For the purpose of this study and in the absence of more information equation 5.3 is used in its general form to represent and compute the drag polar of the aircraft.

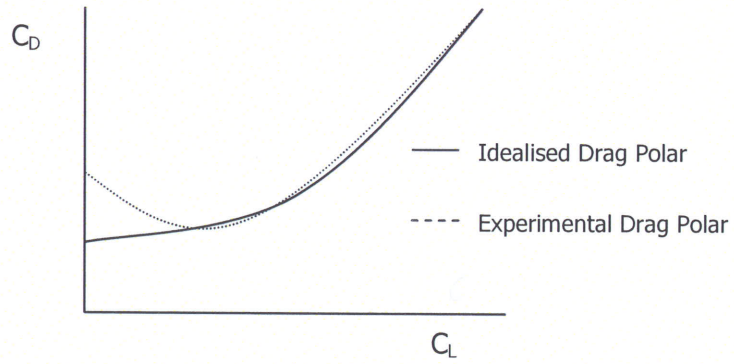


Figure 5.6 Idealised Drag Polar fitted to Real Drag Polar
Adjusted from Reference 130

5.6.5.2 Zero Lift Drag Coefficient

The zero lift drag coefficient is estimated using the component buildup method. The method estimates the subsonic profile drag of each component using a calculated flat-plate skin friction coefficient (Cf) and a component form factor (φ) that estimates the pressure drag due to viscous effects. Then the interference drag effects on the component are estimated as a factor "Q" and the total component drag is determined as the product of the wetted area Swet, Cf, φ and Q. The method also allows for the estimation of miscellaneous drag components arising from the deployment of flaps, extension of landing gear and the existence of trim drag. The general form of the profile drag buildup method is described by equation 5.4 below:

$$(C_{DO}) = \frac{\Sigma(Cf_c \, \varphi_c Q_c Swet_c)}{Sref} \qquad 5.4$$

where the subscript "c" denotes that those values are different for each component. The components that have been considered are the fuselage, outer wings, fins, tailplane, engine pylons and nacelles. The reference area Sref is the plan wing area. More information regarding the computation of profile drag coefficients for each component is given in Roskam[76, 77], Jenkinson[126] , Raymer [131] and Torenbeek [75].

5.6.5.3 Lift Induced Drag

Within the aerodynamic module lift dependent drag is calculated form equation 5.5 below:

$$C_{D_I} = \left[\left(\frac{C_1}{C_2 \pi AR}\right) + C_3 + C_4 C_{D_O}\right] C_L^2 \qquad 5.5$$

The lift dependent drag is computed by taking into account its three main components:

1. A component from the wing planform geometry
2. A component from the non-optimum wing twist
3. A component due to viscous effects.

Coefficients C_1 and C_2 account for the wing planform geometry and depend on the wing taper ratio and aspect ratio. Coefficient C_3 accounts for the non-optimum wing twist while coefficient C_4 is used to describe the viscous effects. More information on the computation of induced drag coefficient and the exact values used for the various coefficients is given in Reference 127.

5.6.6 Aircraft Performance Module

Information from the rest of the modules is passed to the aircraft performance module where detailed figures are produced and the overall performance of the aircraft is computed. Given the flight conditions from the mission profile module, aerodynamic characteristics are determined by the aerodynamic module and the information is used along with the engine data to derive all the necessary information required for the calculations to determine the performance of the aircraft at the various segments. The values derived are then used for the estimation of the overall performance. Typical outputs include: total fuel needed to complete a given mission, distance covered and mission duration, aerodynamic efficiency and detailed aerodynamic figures, engine sfc at particular segments, rate of climb and descent, etc. To be able to compute the required values the mission profile is divided into the sections described in paragraph 5.3 and shown in Figure 5.1

5.6.6.1 Take-off and Initial Climb Calculations

Take off distance is the distance required for take off with all the engines operating. This is the distance from rest to an initial altitude of 35 ft multiplied by 1.15. For the purpose of the study the take-off distance is divided into three sections:

1. Ground Roll
2. Transition to climb
3. Climb

Ground roll is further divided into two sections, the distance from rest to the point where the aircraft rotates and the distance from the point of rotation to lift off. During transition to climb the aircraft accelerates from the lift off speed to the climb speed, usually denoted as V_2. The final section of climb lasts until an altitude of 35 ft is reached. More details on the calculation of the take-off distance are provided in Jenkinson[126]. The fuel consumed is approximated by considering a fixed amount of time and the engine operating at maximum take off thrust. Take-off calculations with one engine inoperative have been omitted.

5.6.6.2 Climb Calculations

The general case of an accelerated climb is considered. As mentioned in paragraph 5.6.2 the aircraft climbs at a constant equivalent air speed of 250 knots until the altitude of 10000 ft is reached. Then the aircraft accelerates horizontally to 320 knots equivalent air speed and continues to climb until the cruise Mach number (0.82 Mach) is reached. The height at which the climb speed equals Mach 0.82 was determined to be 26500ft.

To allow for the computation of time, distance and fuel consumed, the climb is divided into intervals of 1500ft. As it is explained in Appendix B, climb calculations rely on the computation of the rate of climb, which at conditions of zero wind is equal to the change in height with respect to time, i.e.:

$$Rate\ of\ C\lim b = \frac{dh}{dt} = V_{VERTICAL} \tag{5.6}$$

The time, distance covered and weight of the aircraft (thus fuel consumed) at the end of each interval depend on the rate of climb. This in turn depends on the drag, thrust and

mass of the aircraft during the interval. The time to climb from altitude h_1 to an altitude h_2 can be calculated by the following expression.

$$t = \int_{h1}^{h2} \frac{1}{V_{Vertical}} dh \tag{5.7}$$

The integration of the above equation is not easy because of the interrelationship of the variables involved. For this reason an iterative procedure is adopted for each climb interval. The procedure relies on the estimation of a trial time and weight at the end of each interval, which are then used to compute the correct values.

When the aircraft climbs at constant equivalent air speed the true air speed increases as the density reduces. Further more, when the aircraft climbs at constant Mach number the true airspeed will reduce as the speed of sound reduces. In the Troposphere, however, constant Mach number will lead to constant true air speed. For these reasons appropriate acceleration factors are also included in the computation of the rate of climb.

The total time spend at climb is calculated by adding the time spend at each interval as follows:

$$Total\ \ time = T = \sum_{i=1}^{n} t_i \tag{5.8}$$

The horizontal distance covered in climb depends on the climb speed and the angle of climb, γ. For the reasons previously mentioned, γ will vary and needs to be computed for each interval. The total horizontal distance is then calculated as:

$$Total\ distance = \sum_{i=1}^{n} t_i V_i \cos\gamma_i \tag{5.9}$$

In a similar fashion, the total fuel consumed is computed by adding the fuel consumed at each interval.

5.6.6.3 Cruise Calculations

In the cruise phase the range flown is computed using the appropriate aircraft and engine parameters along with the fuel available for cruise. For an aircraft powered by turbojet or

turbofan engines the specific fuel consumption is defined as the fuel flow per thrust produced, i.e:

$$sfc = \frac{Q}{F} \tag{5.10}$$

where Q is the fuel flow and F is the thrust. In level flight, thrust equals drag and the lift produced equals the weight of the aircraft, so that:

$$F = D = EL = EW \tag{5.11}$$

Where D is the drag of the aircraft, L is the lift produced, W is the weight of the aircraft and E is the aerodynamic efficiency defined as the ratio of drag over lift. The Aerodynamic efficiency also equals:

$$E = \frac{\frac{1}{2}\rho S_{REF} C_D V^2}{\frac{1}{2}\rho S_{REF} C_L V^2} = \frac{C_D}{C_L} \tag{5.12}$$

Where ρ is the density, V is the flying speed, S_{REF} is the wing plan area and C_D, C_L are the drag and lift coefficients. The mass of fuel used per distance travelled is given by the following expression:

$$\frac{sfc\ F}{V} = \frac{sfc\ E\ W}{V} \tag{5.13}$$

As the aircraft travels a distance dR, fuel is consumed and the mass of the aircraft changes, so that:

$$dm = -\left(\frac{sfc\ E\ W}{V}\right) dR \tag{5.14}$$

The specific range r_a is defined as the distance travelled per unit mass fuel used. Therefore equation 5.14 can be written as follows:

$$r_a = -\frac{dR}{dm} = \frac{V}{sfc\ E\ W} \tag{5.15}$$

The total cruise range can be computed by integrating equation 5.15 and putting W = mg, as follows:

$$Range = -\int_{m1}^{m2} r_a \, dm = -\int_{m1}^{m2} \frac{V}{sfc \; E \; m \; g} \, dm \tag{5.16}$$

where m_1 and m_2 are the masses of the aircraft at the beginning and end of the cruise. The evaluation of the above integral requires the expression of variables V, sfc and E in terms of m during cruise. Usually the following approach is followed, assumptions regarding these variables are made to allow the equation to be solved. The implications of these assumptions on the flight mechanics of the aircraft are considered and approximations are made. In general, three flight schedules, which produce three set of assumptions for the variables, are considered and allow the integral to be solved. These are as follows:

1. Cruise at constant altitude with constant sfc and lift coefficient.

2. Cruise at constant airspeed, constant lift coefficient and sfc.

3. Cruise at constant altitude, constant airspeed and sfc.

In most cases the second flight schedule is assumed. The condition of constant lift coefficient and airspeed implies that the aerodynamic efficiency of the aircraft is constant. Moreover, the lift coefficient can be expressed as follows:

$$C_L = \frac{W}{q \; S_{REF}} = \frac{W}{\frac{1}{2} \rho \; V^2 S_{REF}} \tag{5.17}$$

Equation 5.17 implies that for constant lift coefficient and true airspeed the ratio of aircraft weight to air density must remain constant. Since fuel is consumed, the weight of the aircraft reduces and density must reduce, accordingly. This is possible if the aircraft is allowed to climb very slowly, so that the density reduces proportionally to the reduction in weight. This reduction in air density will also reduce the thrust delivered by the engine and therefore, it can be assumed that the aircraft will fly at constant true air speed. The above assumptions, however, underestimate the effect of air density on the profile drag of the aircraft. In practise, the aerodynamic efficiency of the aircraft will not remain constant and

it will vary with changes in altitude and aircraft weight. In addition sfc will vary with changes in altitude. In practise, air traffic control does not allow cruise-climb profiles and insists on airlines adopting a stepped climb procedure as shown in Figure 5.4. Assuming constant sfc, airspeed and lift coefficient equation 5.14 becomes:

$$Range = -\frac{V\ E}{sfc\ g}\int_{m1}^{m2}\frac{1}{m}dm = \frac{V\ E}{sfc\ g}\ln\left(\frac{m_1}{m_2}\right) \tag{5.18}$$

This is the well-known Brequet equation for range. In practise, an aircraft in cruise will not accelerate or climb. Normally, it will be flying at constant speed and altitude. Since fuel is consumed and the weight of the aircraft reduces, the power setting of the engine will reduce and sfc and aerodynamic efficiency will vary. Eshelby[103] and Mair[129] describe alternative solutions to the equation 5.16 corresponding to flight schedules 1 and 3. However, all three flight schedules have similar limitations with regard to the assumptions made and therefore will give approximate solutions.

According to Young[132], a better approach for estimating the range of the aircraft is to perform a numerical integration of the basic range equation 5.16. The technique is known as the integrated range method and the range is given by diving the mass change in 'n' intervals so that:

$$Range = \sum_{i=1}^{i=n}\frac{Vi}{Qi}\left(m_i - m_{i-1}\right) \tag{5.19}$$

Provided that the intervals are small enough, such an approach can account for constant speed and altitude as well as variations in sfc and drag. For the purpose of this study, a similar approach to the integrated range method is adopted. However, instead of dividing the mass of the aircraft into intervals and then calculating range time and fuel consumed, the cruise is divided into time intervals. The formulation is described in the following paragraphs.

For straight and level flight the decrease in aircraft weight with time due to fuel burn is given by the following expression:

$$\frac{dW}{dt} = -sfc\ E\ W \tag{5.20}$$

The total fuel consumed can be calculated by integrating equation 5.20 with respect to time:

$$Fuel\ Consumed = \int_{1}^{2} sfc\ E\ W\ dt \tag{5.21}$$

Dividing the cruise section into 'n' intervals the total fuel consumed can be calculated as follows:

$$Fuel\ Consumed = \sum_{i=1}^{n} sfc\ E\ W\ \left(t_i - t_{i-1}\right) \tag{5.22}$$

The range covered at each time interval is then given by:

$$dR = V_{CRUISE}\ dt \tag{5.23}$$

The total range is computed as the sum of the range covered at each interval.

To improve the overall efficiency of the aircraft, long-haul airliners are allowed to perform what is known as step climb cruise, shown in Figure 5.4. The mission profile module allows the user to specify different altitudes for each cruise section. Climb calculations are computed as explained before. The time spent at each cruise segment is calculated by the aircraft performance module through an iterative process for maximum specific fuel range, i.e appropriate sfc and aerodynamic efficiency for minimum fuel usage per distance covered.

Provided that the time spent at cruise is known, the fuel consumed and distance covered can be computed as explained before. However at the start of the calculation the time is not known due to the uncertainty of the climb and descent distances. For this reason the user specifies the time spent at cruise assuming that the total range of the aircraft would equal the distance covered at the cruise section and at appropriate flying conditions i.e. altitude and Mach number. This is a trial value that is used by the aircraft performance module to initialise the solution. The calculated total range after each iteration is compared to the input data and the calculations are repeated until convergence is achieved.

The calculations begin with the mass and flight conditions at the start of each time interval. The start weight corresponds to the weight at the end of climb. With the trial

value of total time spent at the cruise, individual time intervals are calculated and the total fuel consumed is computed at the end of the cruise. This is done using the modified integrated range method and 25 minutes time intervals or as required.

For a given altitude and flying Mach number the atmospheric module calculates the speed of sound, density, and true airspeed. The lift coefficient is evaluated for the weight at that particular station, i.e start and end of specified time interval. Equations 5.4 and 5.5. are used by the aerodynamic module to calculate the zero drag and lift induced drag coefficients of the aircraft. Based on the drag polar shown in equation 5.3, the total drag of the aircraft and hence the trust required by the engines is computed. Data from Turbomatch models are used to pass on the sfc of the engine at the required operating conditions. The fuel consumed at each time interval is then computed using equation 5.20 and mean values for sfc drag and aircraft mass. The total distance travelled and fuel consumed are computed using the modified integrated range method and equations 5.22 and 5.23. The final cumulative values of the range, weight and time are passed to the next flight segment. The remaining mission calculations (i.e. descent and approach) are then performed. The total range of the aircraft is calculated and the process continues until the calculated range equals the input range. The diversion profile is then computed in a similar way to the main mission profile. If the fuel consumed is not equal to the fuel available onboard the aircraft, changes are made to the range of the aircraft and the next calculation cycle is undertaken until convergence within 0.1% is achieved.

5.6.6.4 Descent Calculations

Descent calculations are similar to the climb calculation mentioned above. Therefore, the descent schedule is divided into a number of sections and iterations are performed to calculate the rate of descent, time elapsed and horizontal distance covered. Engine thrust and fuel flows at idle conditions are calculated for each section using Turbomatch. In the descent segment the thrust delivered by the engines is lower than the drag produced by the aircraft and therefore the aircraft is allowed to glide. The descent starts with a speed corresponding to cruise Mach number reducing to 250 knots at sea level.

5.6.6.5 Hold Calculations

The aircraft may be required to hold at its destination before landing. For international flights, a 30 minutes hold at 1500 ft over the alternate airport is required at the end of the

mission. To minimise the fuel carried, the aircraft should fly at conditions of minimum fuel consumption and maximum endurance. From equation 5.18 follows that endurance is a function of engine sfc and aerodynamic efficiency. Since the holding altitude is defined, minimum fuel consumption can be achieved when the aircraft operates at maximum aerodynamic efficiency, which in turn implies minimum drag. Figure 5.7 shows the derived variations of total drag, zero lift drag and lift induced drag with speed for the modelled A340-500 class of aircraft at typical hold conditions.

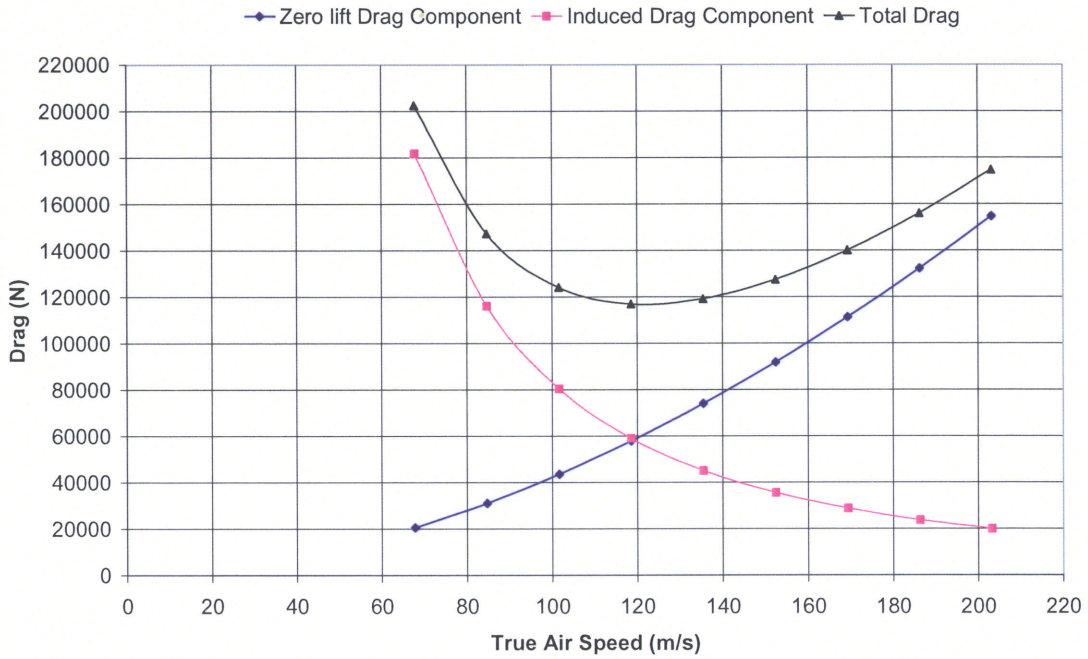


Figure 5.7: Variation of Drag with True Airspeed for the A340-500 class of Aircraft at Typical Holding Conditions

It can be seen that for given aircraft weight and flying altitude the zero lift drag component increases with speed and the lift induced drag component reduces with speed. The total drag, however, reduces with speed until the point where the zero lift drag equals the induced drag. Further increase of speed leads to a continuous drag increase. From the above it follows that minimum drag and aerodynamic efficiency is achieved when the following condition is satisfied:

$$\frac{1}{2}\rho\, S_{REF} C_{D_0} V^2 = \frac{1}{2}\rho\, S_{REF} C_{D_I} V^2 \Rightarrow C_{D_0} = C_{D_I} \tag{5.24}$$

The above condition can be used with equation 5.3 to compute the lift coefficient for minimum drag:

$$C_{D_O} = K\ C_L^2 \Rightarrow C_L = \sqrt{\frac{C_{D_O}}{K}} \tag{5.25}$$

maximum aerodynamic efficiency will then equal:

$$E_{MAX} = \frac{C_L}{C_{D_{MIN}}} = \frac{\sqrt{\frac{C_{D_O}}{C_{D_I}}}}{C_{D_O} + K\ \frac{C_{D_O}}{K}} = \frac{1}{2\sqrt{C_{D_I} K}} \tag{5.26}$$

The speed at which maximum aerodynamic efficiency occurs is given by:

$$W = \frac{1}{2}\rho\ S_{REF} C_L V^2 \Rightarrow V_{E_{MAX}} = \sqrt{\frac{2\ W}{\rho\ S_{REF}\sqrt{\frac{C_{D_O}}{K}}}} \tag{5.27}$$

With the aircraft weight, altitude, speed, time and aerodynamic efficiency known, the performance of the aircraft at the hold segment is computed using the integrated range method.

5.7 Aircraft Data

5.7.1 General Data

Data for the baseline aircraft have been collected from various references including the web site of the manufacturer. A general description of the aircraft is given in Table 5.2 below.

Since the estimated aircraft characteristics will not fully match those of the baseline aircraft, a distinction is made here and the performance models are referred to as the A330 class aircraft, A340 class aircraft and A380 class aircraft.

	Airbus A330-200	Airbus A340-500	Airbus A380-100
MTOW	233000 Kg	365000 Kg	560000 Kg
MZFW	165142 Kg	222000 Kg	356000 Kg
OEW	120200 Kg	170390 Kg	271000 Kg
Design No of pax	253	313	555
Design Range	6370 nm	8500 nm	7650 nm
Wing Area	363.10 m^2	437.3 m^2	817 m^2
Aspect Ratio	9.26	8.56	7.79
Number of Engines	2	4	4
Engine Type	RR Trent 700	RR Trent 500	RR Trent 900
Sea Level Static Thrust	310 (KN)	235.8 (KN)	294 (KN)
Bypass Ratio	6.5	8.5	9

Table 5.2: Baseline Aircraft Data

5.7.2 Aerodynamic Data

Based on the methodology outlined in the previous sections of this chapter the aerodynamic characteristics of the baseline aircraft were calculated. The derived drag polar at typical cruise conditions, i.e 0.82 Mach at 33000ft, are plotted against lift coefficient in Figure 5.8

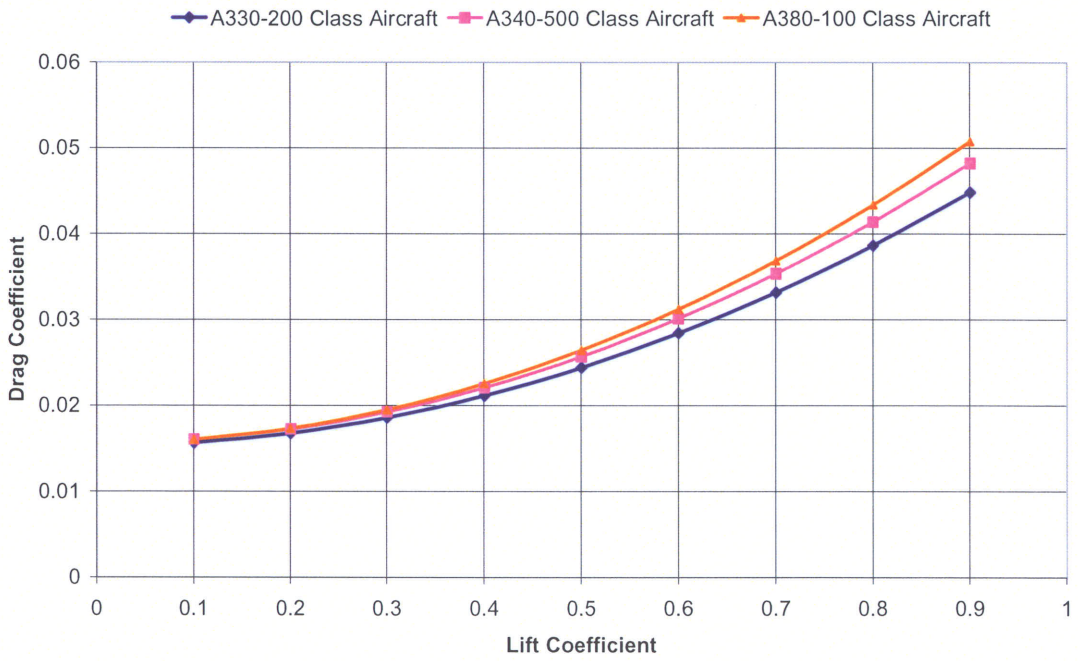


Figure 5.8: Derived Aerodynamic Data

The derived drag polar reveal that for fixed lift coefficient, the lowest drag coefficient corresponds to the A330-200 class aircraft. Moreover, the A330 model has the lower induced drag coefficient. As explained earlier, induced drag coefficient depends on wing aspect ratio and the above result is attributed to the fact that from the three aircraft modelled the A330-200 class aircraft has the highest aspect ratio and thus the lowest induced drag coefficient. The above observation is a clear indication of the limitations imposed by the 80-meter box requirement, discussed in Chapter 2, and its effect on the design and performance characteristics of large aircraft such as the A380. In the case of A380 it is safe to assume that the detrimental effects of reduced aspect ratio are partly offset by the increased wing loading of the aircraft, i.e. weight divided by wing plan area, which in this case has a triple effect. The first effect is of reduced lift coefficient and hence reduced lift induced drag. From equation 5.22, it follows that for a fixed flying speed and aircraft weight reduced wing loading requires a lower lift coefficient. The second effect is that of reduced zero lift drag, which in turn partly offsets the detrimental effects of increased wing area on total drag. Equation 5.4 shows that for a given aircraft, i.e. fixed geometry, the total drag coefficient reduces with wing area. The above assumption becomes evident when comparing the zero lift drag coefficients for the A340-500 and A380-100 models. The final effect is of reduced weight. It is well known that increased aspect ratio has a detrimental effect on aircraft weight, sometimes even higher than wing area, and usually its value is limited by weight considerations. The latter implies that even if the span of the wing were allowed to exceed the 80 meters constrain, it is quite probable that the aspect ratio of A380 would be smaller than that of A330 or even that of A340, due to the size and design mass of the aircraft.

The detrimental effects of reduced aspect ratio on a large aircraft, such as the A380, become even more important when considering the potential impact of adopting the AEA/E technologies. Previous studies demonstrated that implementation of all or more electric technologies could lead to a reduction in MTOW. These savings can manifest through improved engine performance and/or reduced equipment weight. Provided that the savings are large enough, fixed wing loading could lead to a reduction in wing area, which in turn would allow for increased aspect ratio without exceeding the 80 meter box constrain. The detrimental effects of aspect ratio on weight can then be offset by the initial weight reduction and the resulting wing area reduction.

As a first approximation it can be assumed that, fixed wing loading would lead to constant lift coefficient and therefore the area of the wing will not have to be optimised for the new

configuration. Although this is probably a pessimistic approach, it is sufficient for the purpose of evaluating the impact of adopting the AEA/E concepts on the design and performance of large aircraft. The above assumptions could lead to a dramatic reduction in aircraft total drag due to the effect of increased aspect ratio and wing area. It is for these reasons that consideration of large aircraft, as opposed to the aircraft used by previous studies described in the literature review, favours the adoption of all or more electric technologies. These issues are considered in Chapter 8.

5.7.2 Off-Takes/Secondary Power Systems Data

This section deals with the estimation of secondary power requirements and the extraction of off-takes from the engines. More information concerning the configuration and functions performed by the individual systems is given in the following chapter that deals with the limitations of conventional secondary power systems. For the purpose of estimating power off-takes power requirements the operation of the anti/de-icing system was omitted. Three main systems were considered and include,:

- Mass flow and power requirements for the operation of the environmental control and pressurisation systems
- Power requirements for the operation of the electrical system
- Power requirements for the operation of the hydraulic system

The Joint Aviation Authorities specify that, in normal operation the airflow into the cabin should be no less than 0.5 lb/min per passenger. Should one source fail the minimum airflow reduces to 0.4lb/min per passenger. However, this is the minimum airflow defined by the regulation authorities and it may not be adequate to hold cabin temperature at sufficient levels required to ensure passenger and crew comfort. Instead, Jones[58] mentions that 0.9 lb/min per passenger are usually supplied. The exact mass flow extracted from the engines will vary and depends on both the design of the environmental control system and the operator. For these reasons, a preliminary value of 0.9 lb/min per passenger was assumed. The exact mass flow requirements were defined by scaling published data on the customer bleed for the Airbus A300-600 and Airbus A330-200 aircraft. In the absence of detailed information it was assumed, that the data published refer to the design payload of the aircraft.

The same data were also used to define the power requirements for the operation of the environmental control system in the case of the AEA/E derivative aircraft and for sizing the electrical generators.

In addition to the bleed air required for the operation of the ECS discussed above, bleed air was extracted from the bypass stream. Because of the high temperature of the bleed air, a pre-cooler is normally used to reduce the delivery temperature of the air leaving the propulsion system. The preccoler is placed within the pylon and operates on either free stream or bypass air. The latter option is usually adopted to minimize mass and drag penalties. The value of the mass flow extracted from the bypass stream was half that extracted from the core of the engine.

In practice, bleed air requirements will also vary with operating conditions. For the purpose of the study, such variations were ignored and the bleed air requirements refer to typical cruise values that were assumed to remain constant throughout the mission profile. This is a safe assumption considering that, while bleed air requirements reduce with altitude, i.e. maximum bleed off takes occur at sea level operations, their effect on the performance of the aircraft are more pronounced at the cruise segment of the main mission profile.

For the three models, bleed air was extracted from the intermediate pressure port, located at the first stage of the high-pressure compressor and the high-pressure port located at the end of the high-pressure compressor. The location of the intermediate pressure port was defined from published engine drawings, available in Jaynes and the engine manufacturer website, as well as published data on the bleed air delivery pressure of RR RB211 engine operating at typical cruise conditions.

Bleed air was extracted from the high-pressure port at engine idle conditions. i.e. taxi and descent. For the remaining segments bleed air was extracted from the intermediate pressure port.

Shaft power off-takes were estimated by considering the power requirements for the electrical and hydraulic systems. A number of references were used in estimating shaft power off-takes. Published values on the Airbus A300-600, Airbus A330-200, Boeing 767 and Lockheed 1011 Tristar aircraft were used. However, allowances were made for the

number of passengers, size and range of the baseline aircraft. Normal and engine failure cases were also considered.

The exact power requirements were derived by combining the findings of three different approaches. First, the total shaft power off-takes were determined based on the data of existing aircraft. As mentioned before, the total power requirements were considered to be the sum of the electrical and hydraulic loads. Hydraulic power is mainly used by the aircraft's flight control system and normally is a function of aircraft weight. Electrical power requirements are usually considered to be a function of number of passengers. At the aircraft design stage, however, the maximum take-off weight of the aircraft can also be expressed as a function of number of passengers. Therefore, total shaft power off-takes were calculated by computing the power to aircraft mass ratio of existing aircraft, Table 5.5.

	A300-600	A330-200	A340-500	A380-100
Total Generator Capacity	2*90 KVA	2*100 KVA	4*75 KVA	4*150KVA
Cruise Loads	122 KVA	136 KVA	203 KVA	407 KVA
Cruise Off-Takes	150 KW	181 KW	271 KW	543 KW
Take-Off Loads	133 KVA	157 KVA	230KVA	460 KVA
Cruise Half Engines Failed	79 KVA	83 KVA	125 KVA	250 KVA
Take-Off Half Engines Failed	90 KVA	100 KVA	150 KVA	300 KVA

Table 5.4: Electrical Loads and Power Off-Takes

The second approach assumes that, at the cruise section the aircraft's generators operate at their design point and therefore the power requirements would equal to the power delivered by the generators at such operating conditions, Table 5.4. The latter is normally taken as a fixed percentage, 65%-75%, of the rated power. In terms of engine off-takes the efficiency of the generators was taken at 75% for A330 and A340 class aircraft and in the absence of the integrated drive generator 85% for the A380 class aircraft. The values derived where then used with the total shaft power off-takes, previously calculated, to compute the hydraulic loads, Table 5.5.

	A300-600	A330-200	A340-500	A380-100
MTOM	170500 Kg	230000 Kg	365000 Kg	560000 Kg
Off-Take / MTOM	0.001085 KW/Kg	0.001069 KW/Kg	0.001028 KW/kg	0.001107 KW/Kg
Cruise Off-Takes	185 KW	246 KW	375 KW	620 KW
Electrical Loads	122 KVA	136 KVA	203 KVA	407 KVA
Electrical Off-Takes at Cruise	150 KW	181 KW	271 KW	480 KW
Hydraulic Off-Takes at Cruise	35 KW	65 KW	104 KW	140 KW
Hydraulic Off-Takes/ MTOM	0.00021 KW/Kg	0.00028 KW/KG	0.00029 KW/Kg	0.00025 KW/Kg

Table 5.5: Aircraft Off-Takes

The estimated hydraulic power to MTOM ratio for the A380 class aircraft is lower than that of the A330 and A340 class aircraft. However, in the case of the A380-100, one of the conventional hydraulic channels is replaced with an electrical channel. Therefore the total power consumed by the aircraft's flight control system should be higher than the value shown in Table 5.5 and should include the power consumed by the electrical flight control channel. It is also noted that the passengers of the A340-500 at the design point are 313 while the passengers of the A380-100 at the design point are 555. As a result the rated KVA per passenger for the A380 is higher than that of the A340. The higher rating of the generators in the case of the A380-100 is attributed to the fact that some of the generated electrical power is also used to power the aircraft's flight control systems. Assuming that the A380 employs three conventional hydraulic channels and has the same or similar KVA per passenger with the A340, it can be shown that the ratio of hydraulic power to MTOM increases to 0.00035. This value is larger than that of the A330 and A340 and can be attribute to the larger moment of inertia and the shorter tail arm.

The third approach considered the power consumed by the electrical equipment of the aircraft at the cruise section, Table 5.6. The values were derived by scaling published data on existing aircraft [58-60]. In the case of the electrical system, the loads were divided into three types:

- Vital Loads

- Essential

- Commercial loads

	Load	Comments
External Lighting	10 KW	Essential Technical Loads
Internal Lighting	10 KW	Essential Technical Loads
Avionics	20 KW	Essential Technical Loads
Communication	5 KW	Essential Technical Loads
Navigation	5 KW	
Ventilation	10 KW	Essential Technical Loads
Fuel System	35 KW	Essential Load
Miscellaneous	5 KW	
Galleys	115 KW	Non- Essential Commercial Loads
In-flight Entertainment		
Total	220KW	
Total Essential	100 KW	Non Essential Commercial Loads Shed

Table 5.6 Estimated Electric Power Consumption at Cruise for A340-500 Class Aircraft

The estimated variation of off-takes for the A340-500 class aircraft throughout the mission profile is shown in Figure 5.9. The values were calculated by scaling the off-takes at the cruise section.

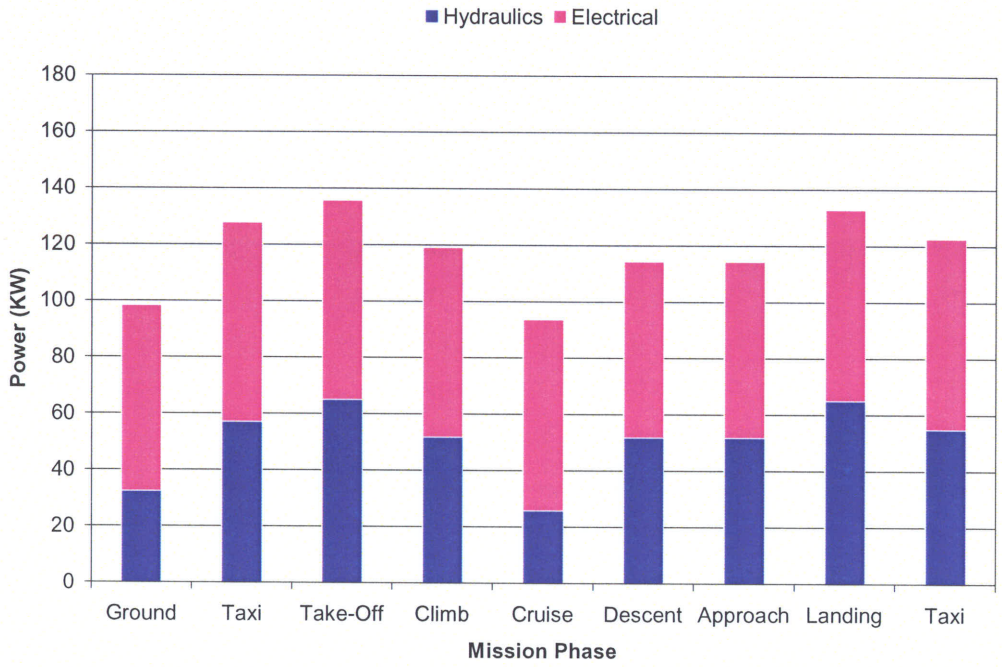


Figure 5.9: Estimated Variation of Shaft Power Off-Takes Throughout the Missions Profile for A340-500 Class Aircraft

5.7.3 Engine Data

Engine data were required for the modeling of the propulsion units with Turbomatch. Typical data were collected from Janes[127] and the engine manufacturer website. These include values for sea level static thrust, top of climb thrust (for the Rolls Royce Trent 500 engine), mass flow, bypass ratio and engine pressure ratio. Additional data, i.e. cooling flows, component efficiency, fan pressure ratio and turbine entry temperature, required for the modeling of the engine were either assumed or calculated.

The turbine entry temperature was calculated through an iterative procedure that involved the modeling of the engine with a trial value of turbine engine temperature. The thrust output from Turbomatch was compared with the value given by the manufacturer and appropriate adjustments were made.

The fan pressure ratio was calculated from the following relationship and keeping overall pressure ratio and thrust constant.

$$\frac{V_{BYPASS}}{V_{CORE}} = \eta_{FAN}\eta_{LPT} \tag{5.28}$$

This is the condition for maximum thrust and therefore minimum fuel consumption in the case of a separate exhaust turbofan engine. For fixed overall pressure ratio, bypass ratio, component efficiency, turbine entry temperature and thrust increase of fan pressure ratio increases the velocity of the bypass stream and reduces the velocity of the core. The characteristics of the engines modeled can be found in Janes[123] . It is also noted that the engines were modeled with the estimated off-takes at their design point.

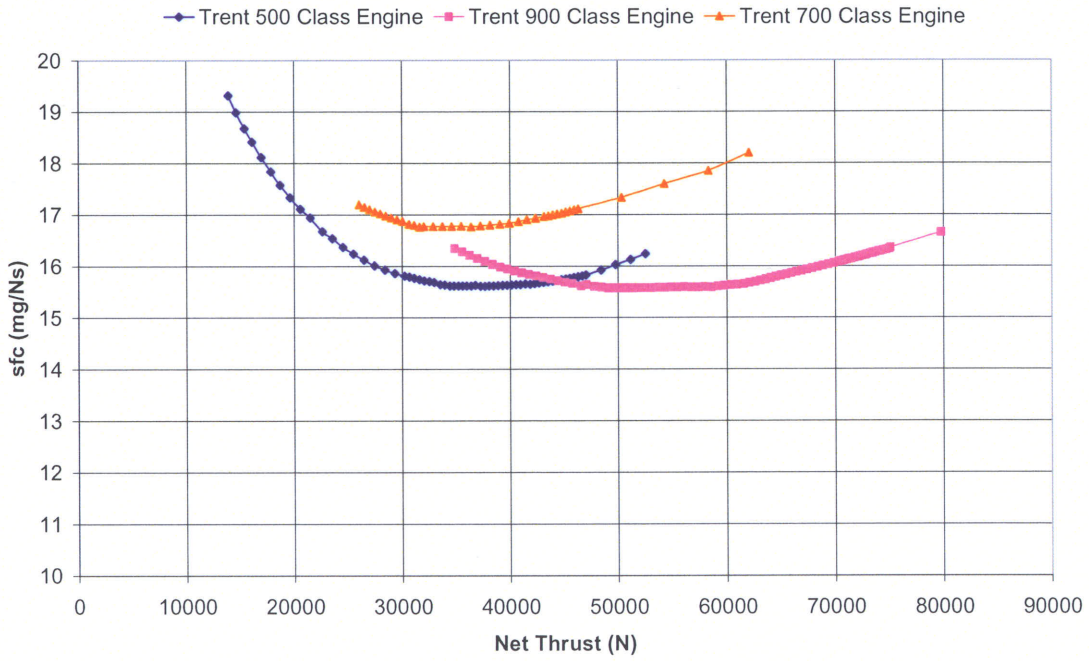


Figure 5.10: sfc Data for Baseline Engines at Typical Cruise Conditions

Typical engine performance data derived from Turbomatch are shown in Figure 5.10. The data include sfc variations with thrust at typical cruise conditions, i.e. 0.82 Mach at 33000ft.

It is noted that from the three engines modeled the Trent 700 class engine has the highest sfc. This result is attributed to the relatively low bypass ratio of the engine. The Trent 700 engine is used with the airbus A330-200 aircraft. The latter is a twin-engine aircraft and in the case of an engine failure the remaining engine must provide the thrust required for take-off and initial climb. Therefore, the engines are oversized to cater for such conditions. The detrimental effects of increased bypass ratio on aircraft mass and drag eliminate the benefits of improved sfc and therefore the engines installed on twin-engine aircraft usually have a lower bypass ratio than engines used with four engine aircraft.

In addition to the mass and drag penalties, extraction of bleed air for the operation of the environmental control system may also limit the use of high bypass ratio engines with twin-engine aircraft configurations. The design point of a high bypass ratio engine would be the thrust required at the top of climb. This is due to the fact that high bypass ratio engines have a greater thrust lapse with speed and altitude. Although bleed air requirements reduce with altitude the reduction in engine mass flow is greater and therefore the percentage of bleed air extracted from the core of the engine increases with altitude. In the case of high bypass ratio engines installed on a twin-engine aircraft, the detrimental effects of off-takes on thrust coupled with the take-off requirements may require the engine to be oversized, hence increasing the mass and drag penalties imposed by the propulsion system and the high bypass ratio in particular. The magnitude, and therefore importance, of these effects increases considering the operation of hot air anti-icing and the fact that high bypass ratio engines are known to be more sensitive to the extraction of secondary power and bleed air in particular. Finally, the above-mentioned effects are excaudate considering the difficulty of matching the engine for minimum sfc at cruise conditions.

Elimination of bleed air, however, and adoption of all or more electric technologies may offset some or all the limitations imposed by increased bypass ratio and may lead to a radical change in the design, configuration and performance of long-range twin-engine aircraft, such as the Airbus A330-200.

High bypass ratio engines can only be used with twin-engine aircraft if ways can be found to minimize the drag and mass penalties and also the effect of extracting bleed air on the design of the engine. Keeping the thrust to weight ratio of the aircraft constant and assuming that the long range of the aircraft would lead to sufficient fuel savings, the reduction in MTOW would lead to a reduction in engine thrust. This in turn, would lead to an additional reduction in MTOW and engine size. The wing area can also reduce to provide further mass and thrust savings. These reductions in MTOW and engine thrust coupled with constant and less severe shaft power off-takes may justify an increase in engine bypass ratio and therefore an overall reduction in fuel usage. These issues are considered in Chapter 8.

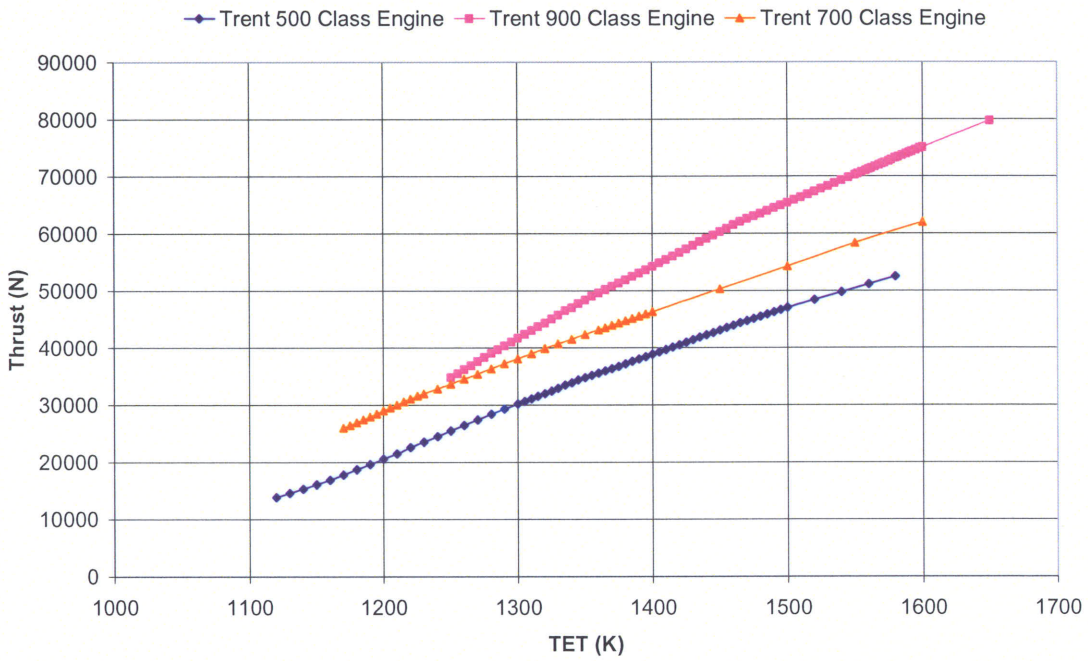


Figure 5.11: TET Data for Baseline Engines at Typical Cruise Conditions

5.8 Aircraft – Engine Performance Model Validation

The accuracy of the aircraft engine performance model was validated against published data. The results are shown in Figures 5.12 and 5.13. Figure 5.12 shows the details of the payload range diagram for the Airbus A340-500 while Figure 5.13 includes all three-baseline aircraft. The payload range diagram for the A340-500 class aircraft was derived by modeling the three operating conditions mentioned in section 5.5. In each case the

payload, fuel load and MTOW were adjusted as required based on information found in Janes [127].

In the case of the A330 class aircraft, the payload range diagram required the modeling of two operating conditions, i.e. maximum payload and ferry mission. In each case the MTOW was kept constant. In the case of the ferry mission the mass of the payload was replaced with fuel.

In the case of the Airbus A380-100 class aircraft only the design point was modeled. This was due to the absence of more information. It is noted that for the three baseline aircraft all subsequent calculations and results refer to the design point of the aircraft.

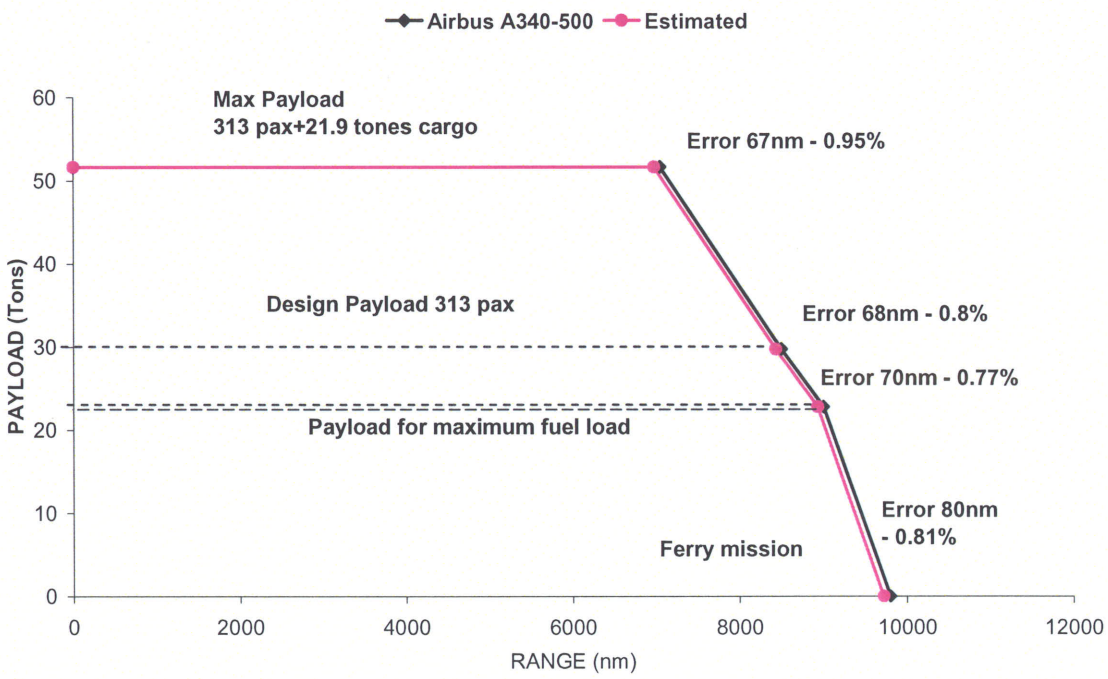


Figure 5.12: Airbus A340 Class Aircraft Payload Range Diagram

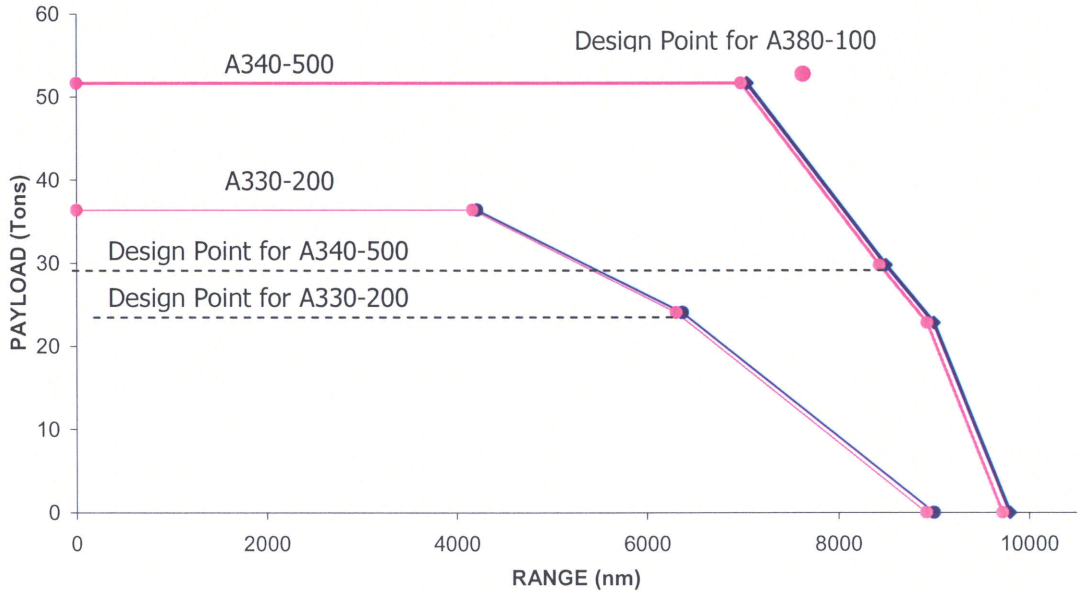


Figure 5.13: Payload Range Diagram Fro Baseline Aircraft

5.9 Results on Aircraft – Engine Performance

5.9.1 Cruise

The overall performance of the A340-500 class aircraft is shown in Figure 5.14. The performance of the aircraft is expressed in the form of aerodynamic efficiency while engine performance is expressed in terms of specific fuel consumption.

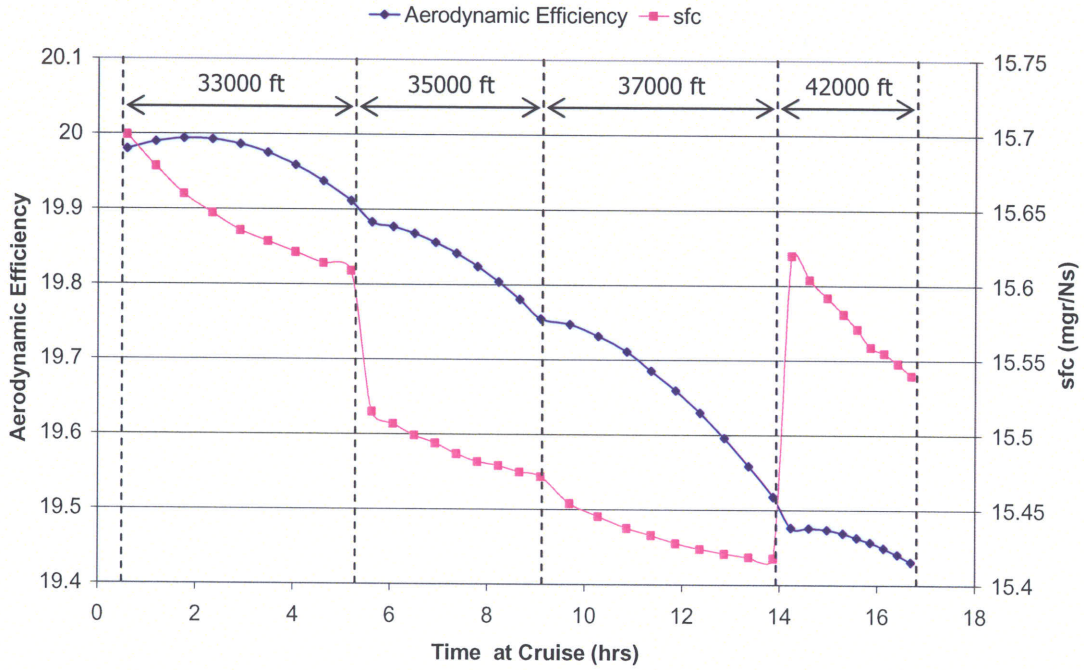


Figure 5.14: Aerodynamic Efficiency and sfc Variation for the A340-500 class Aircraft at Cruise

It is noted that, the aerodynamic efficiency of the aircraft reduces with time. The sfc on the other hand, reduces continually until the final stage of the cruise is reached, where the sfc peaks at 42000 ft. As the aircraft spends time at cruise, fuel is consumed and the aircraft becomes lighter. The lift produced by the wing is a function of wing area, flying speed and lift coefficient. Therefore, constant flying speed and wing area lead to reduced angle of attack and lift coefficient. From definition, maximum aerodynamic efficiency occurs at minimum drag and this in turn takes place when the induced drag coefficient is equal to the profile drag coefficient. Constant flying speed leads to constant zero lift drag coefficient and reduced lift drag coefficient. Thus, although overall drag reduces, the

weight of the aircraft falls quicker than drag, therefore reducing the aerodynamic efficiency of the aircraft. In other words at the start of the cruise, the aircraft is flying at lower speed than that required for minimum drag. At the later stages of the cruise, however, the aircraft flies at speed higher than that required for minimum drag. This is shown in Figures 5.15 and 5.16. The continuous reduction in aircraft mass and lift coefficient lead to an increase in aerodynamic efficiency until the lift coefficient equals the lift coefficient required for minimum drag. Further reduction in aircraft mass and lift coefficient lead to reduced aerodynamic efficiency.

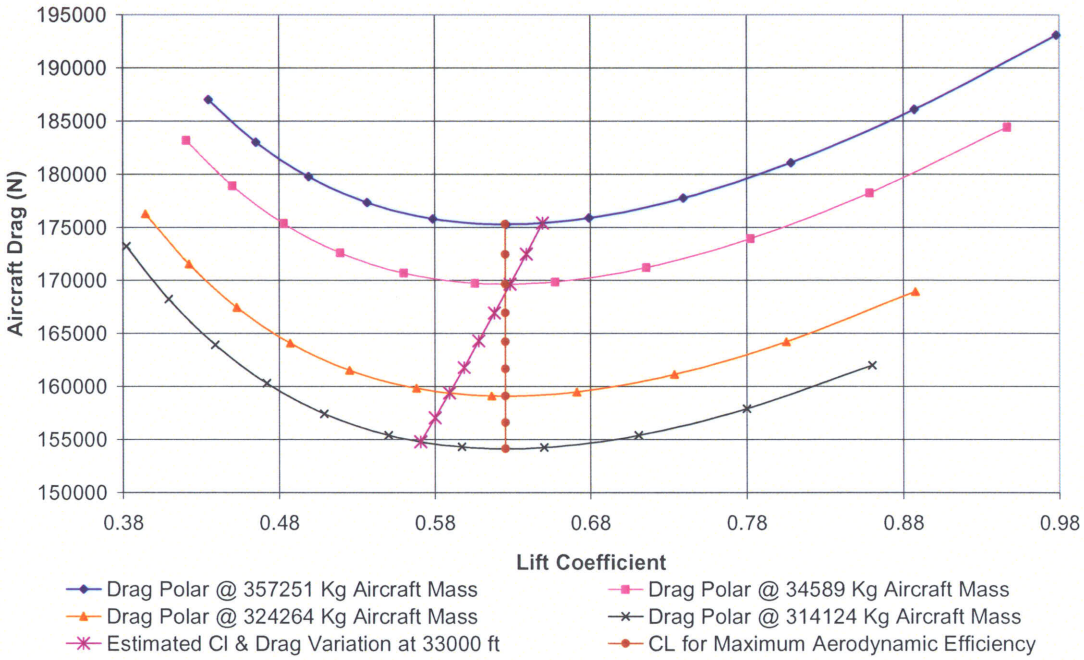


Figure 5.15: Variation of Lift Coefficient and Drag at 3300ft Cruise Conditions – Lift Coefficient for Maximum Aerodynamic Efficiency.

As explained earlier, if the aircraft were allowed to fly at constant Mach No and lift coefficient, it would climb continually maintaining in theory constant aerodynamic efficiency. This assumption is made when the well know Breguet equation is used. Such an approach however undermines the fact that aerodynamic efficiency varies and also that air traffic regulations prevent the aircraft from flying at constant speed and angle of attack.

When the aircraft climbs to a higher altitude at constant Mach No, both the flying speed and the density of the air reduce. This means that the lift produced by the wing reduces

and if the aircraft is to maintain its new altitude the angle of attack must increase. The latter causes a slight increase in aerodynamic efficiency with increase in altitude. It should be emphasized, however, that although the aerodynamic efficiency of the aircraft reduces, the overall drag produced by the airframe reduces as well and so does the fuel consumption per distance covered.

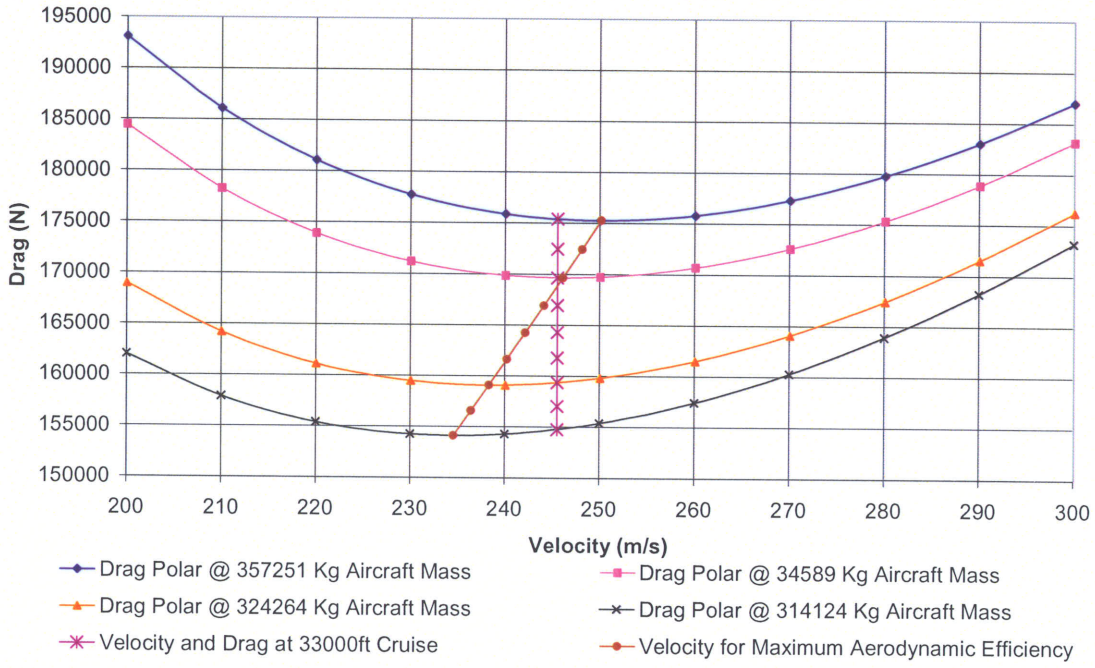


Figure 5.16: Velocity and Drag at 33000ft Cruise Conditions – Velocity for Maximum Aerodynamic Efficiency.

As far as engine sfc is concerned, when the aircraft flies at constant altitude the power setting of the engine is reduces -to account for the reduction in mass and drag- thus increasing the propulsive efficiency of the engine; in expense, however, of thermal efficiency. The dominant effect is the increased propulsive efficiency, which leads to a continuous reduction in engine sfc. When the aircraft climbs at a higher altitude, the thrust produced by the engine reduces, as a result of reduced mass flow and density, thus requiring an increase in TET. The increase in the ratio of TET to ambient temperature improves both thermal efficiency and specific thrust and therefore sfc falls with altitude. This effect is observed up to an altitude of 39000 ft and from then on any increase in TET leads to degraded engine performance and the dominant effect becomes the reduced propulsive efficiency, which in turn causes sfc to increase. The performance of the A330-

200 and A380-100 class aircraft follow the same trends as that of the A340-500 class aircraft. Because of their lower wing loading in order to achieve maximum aerodynamic efficiency, both aircraft were allowed to start the cruise at higher altitude. As a result the engine sfc at the two last stages of the cruise increased. The results are shown in Figure 5.17.

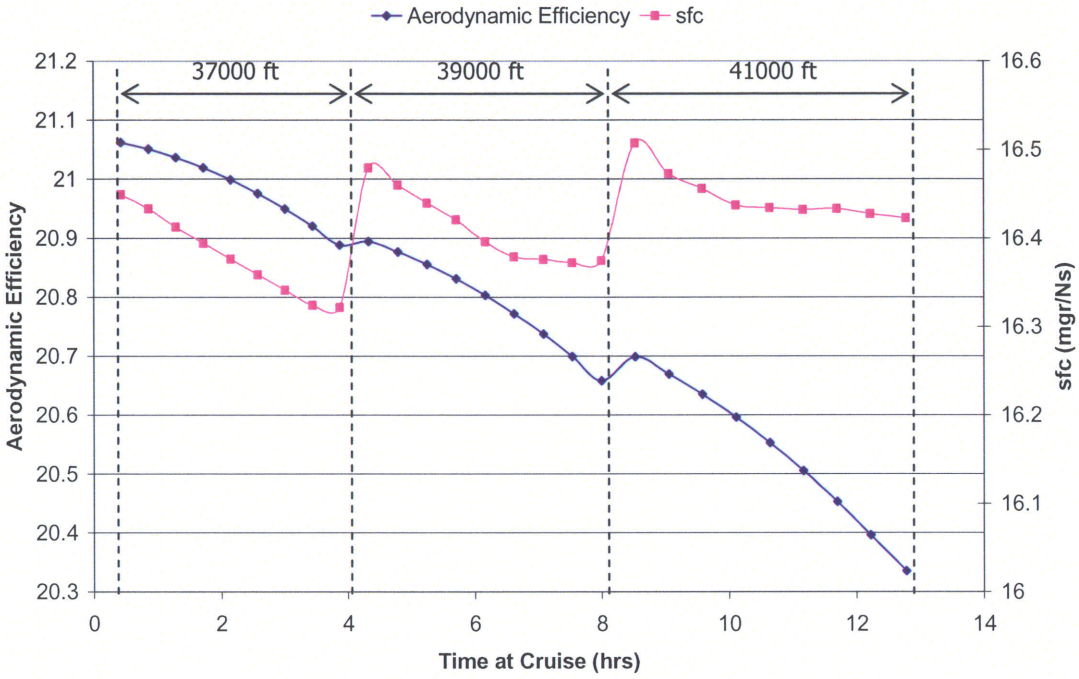


Figure 5.17: Aerodynamic Efficiency and sfc Variation for the A330-200 class Aircraft at Cruise

For the three baseline aircraft, variations in fuel flow, drag and TET with time are shown in Figures 5.19-5.21. From Figure 5.21 is noted that, when the aircraft climbs to a higher altitude the engine power setting and TET increase to offset the effect of reduced air density on thrust. Up to an altitude of 11000 meters the increase in TET required to recover the thrust reduction is partly offset by the reduction in ambient temperature. At higher altitudes, however, the ambient temperature remains constant, thus requiring a larger increase in engine TET. From Figures 5.19 and 5.20 is noted that, despite the degraded aerodynamic and engine efficiency both drag and fuel flow reduce with time. This happens because the dominant effect on both parameters is the reduction in aircraft mass. The latter implies that at the later stages of the cruise, the fuel consumed per distance covered reduces.

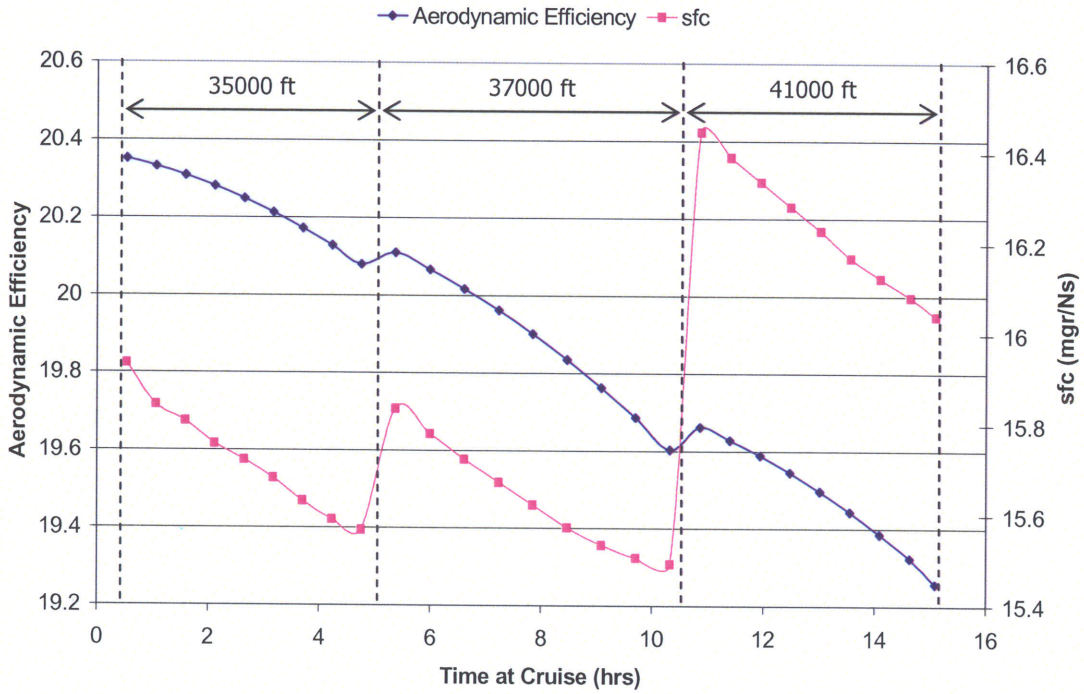


Figure 5.18: Aerodynamic Efficiency and sfc Variation for the A380-100 class Aircraft at Cruise

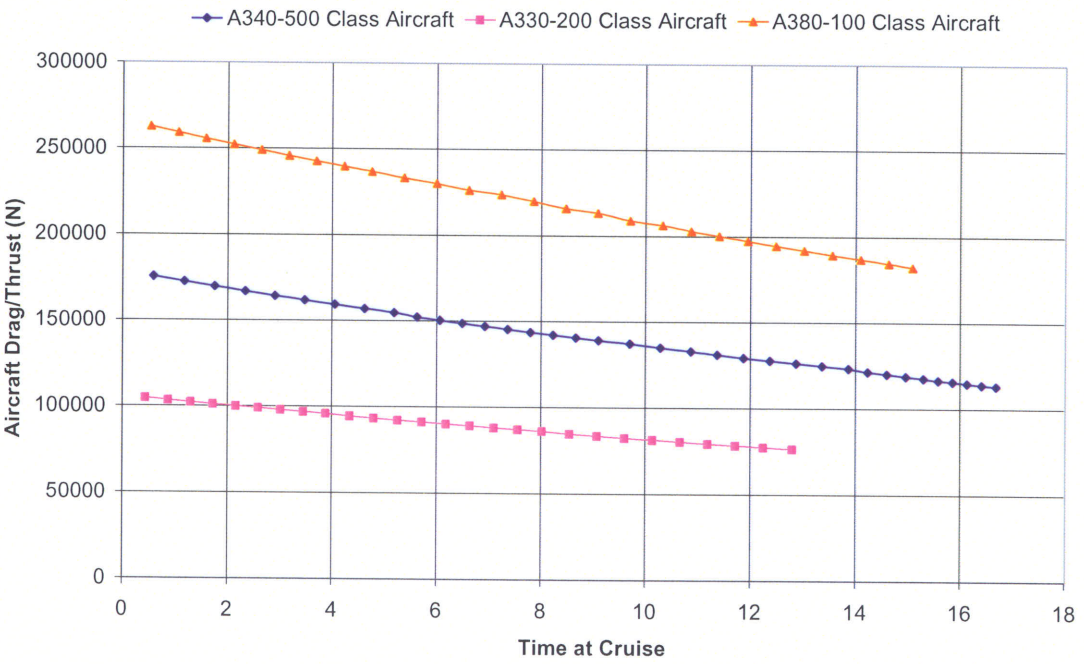


Figure 5.19: Drag/Thrust Variation at Cruise

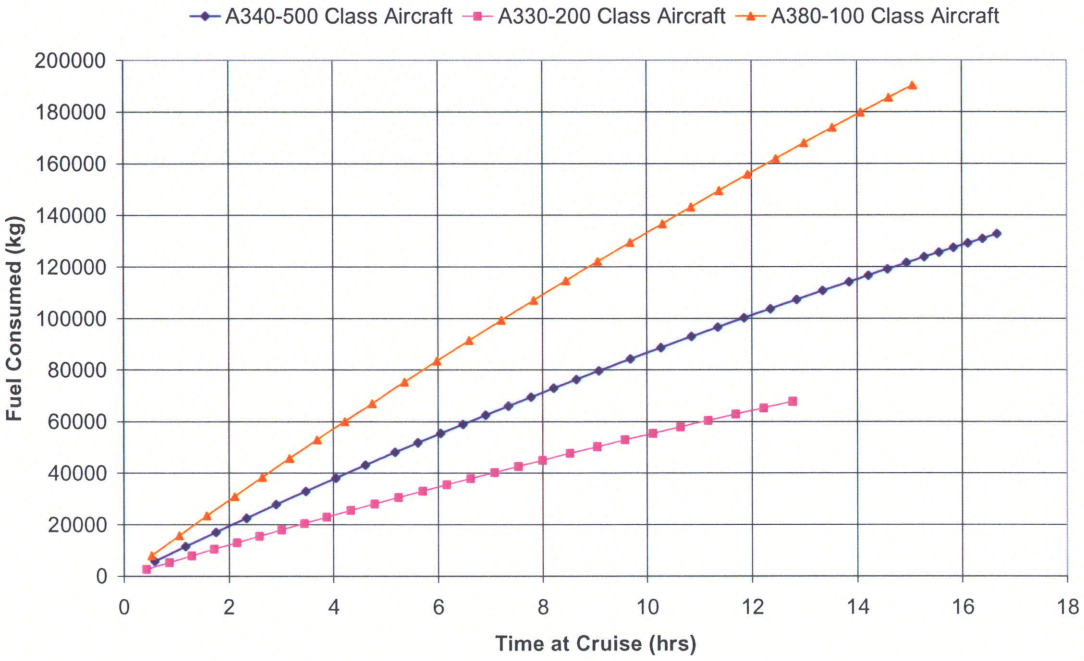


Figure 5.20: Fuel Consumed at Cruise

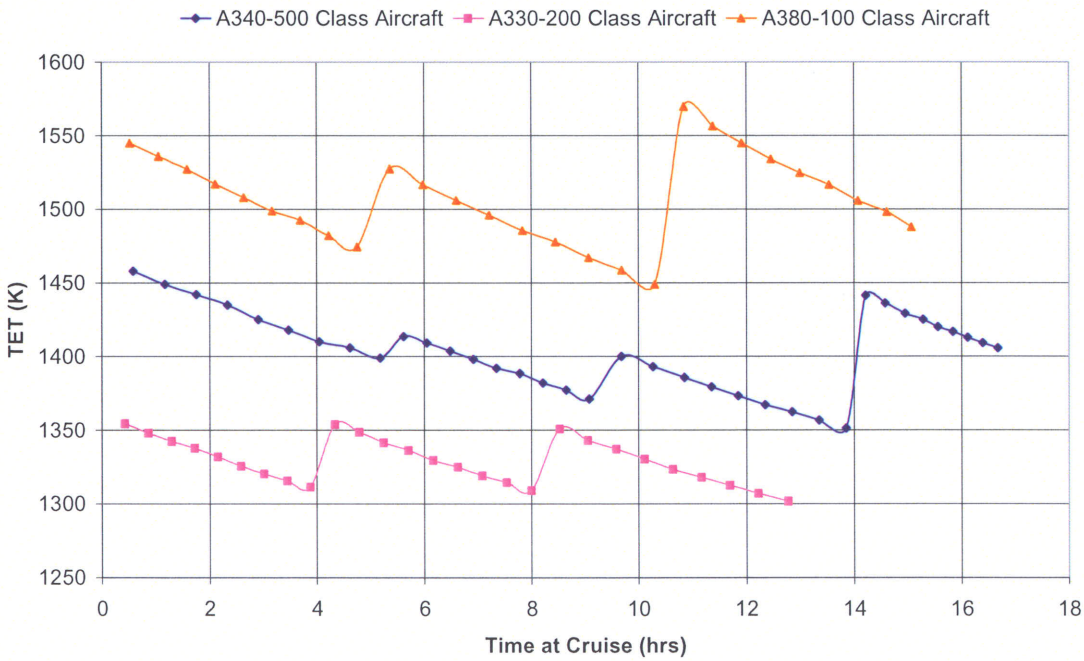


Figure 5.21: TET Variation at Cruise

5.9.2 Climb

The performance of the A340-500 class aircraft at climb is summarised in Figures 5.22-5.27. As mentioned before the general case of an accelerated climb is considered. The details of the climb schedule are given in previous sections. Variations of true air speed and Mach number with altitude are shown in Figure 5.22. As the aircraft climbs at constant equivalent air speed its true air speed and Mach number increase; this is due to the reduction in air density. Climb at constant Mach number, however, leads to reduced true air speed due to the reduction in ambient temperature.

The variation of engine thrust and sfc at various stations of the climb schedule is shown in Figure 5.23. It is noted that the thrust delivered by the engines reduces with speed and altitude. The sudden reduction in thrust at 3000 meters is due to the acceleration from 250 knots to 320 knots equivalent airspeed. The specific fuel consumption of the engine, on the other hand, increases with speed and altitude until the point of constant Mach number is reached. From then on, the reduction in ambient temperature causes sfc to reduce.

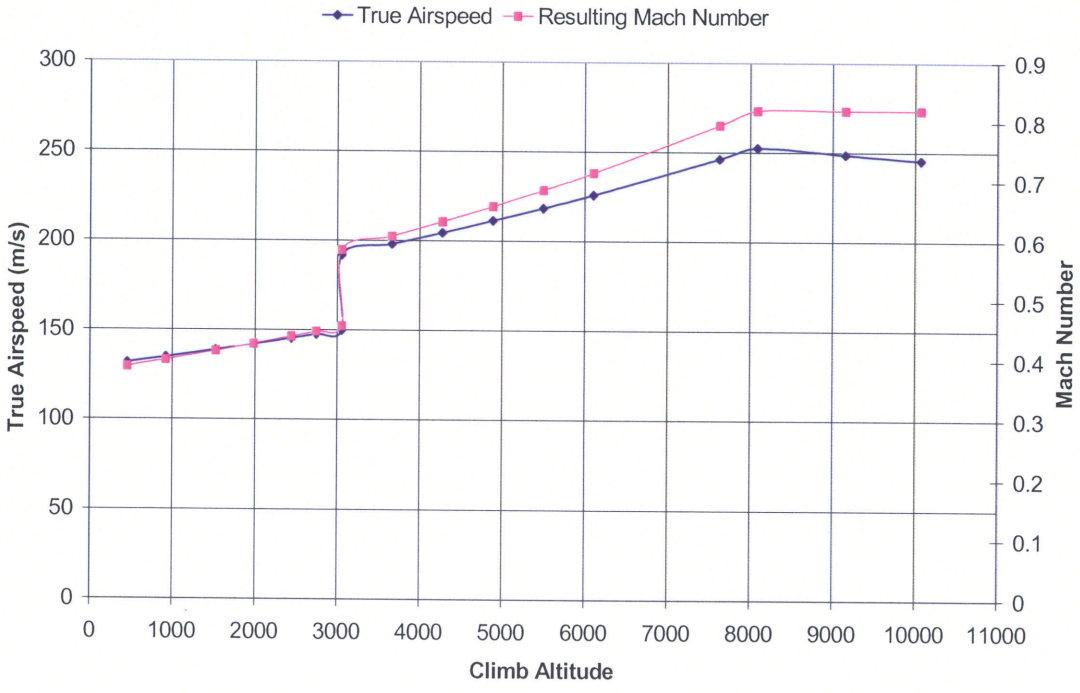


Figure 5.22: Variation of TAS and Mach Number at Climb

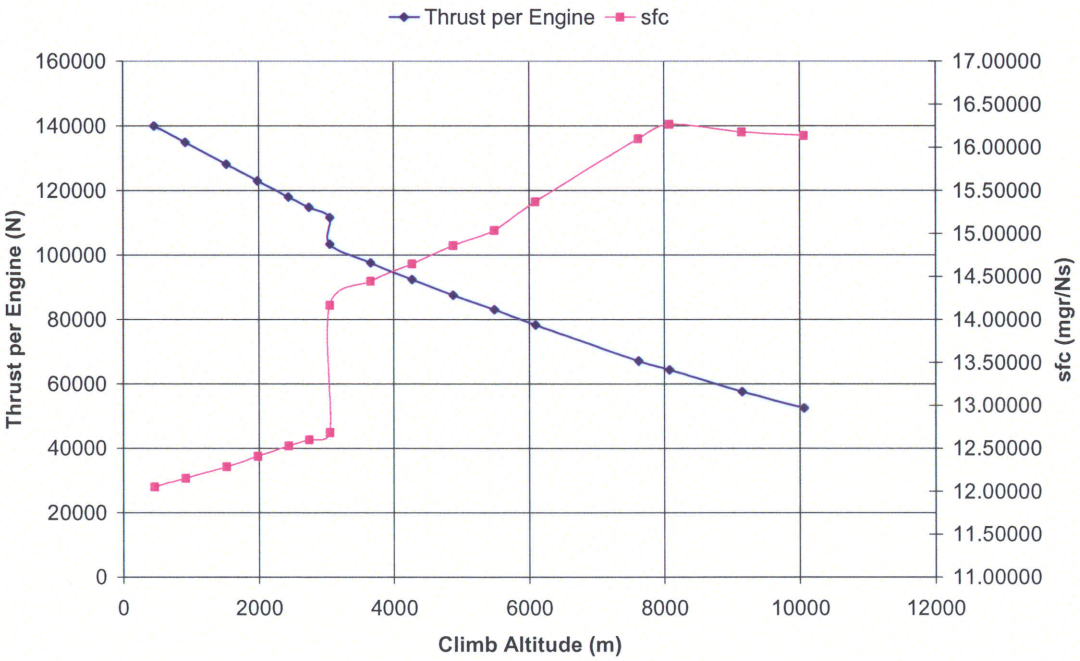


Figure 5.23: Engine Performance at Climb

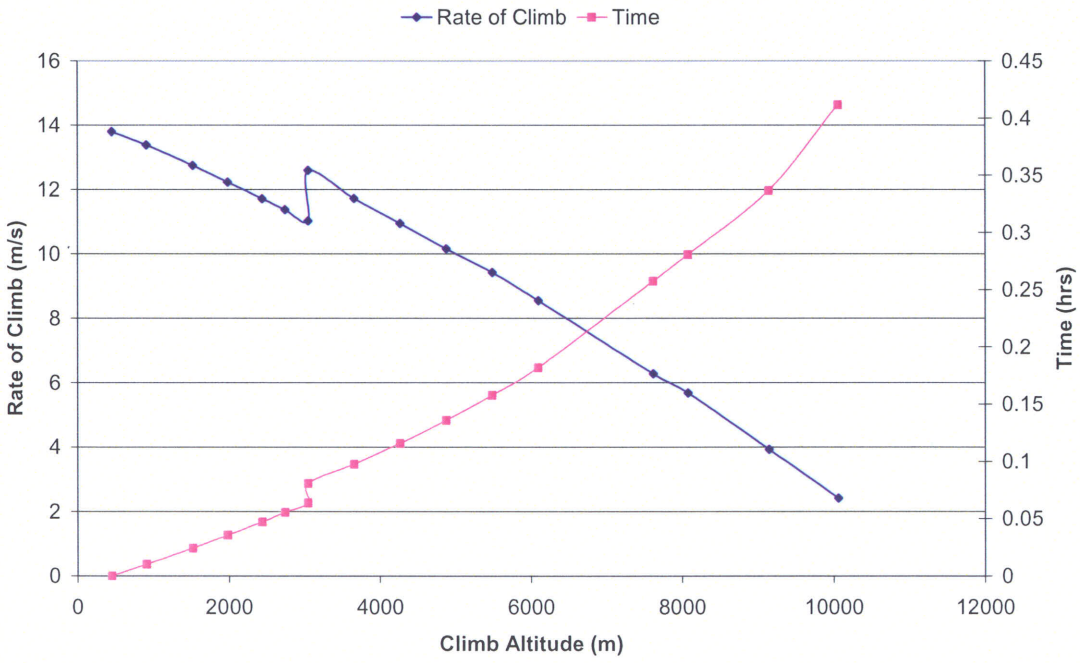


Figure 5.25: Rate of Climb and Time to reach Top of Climb

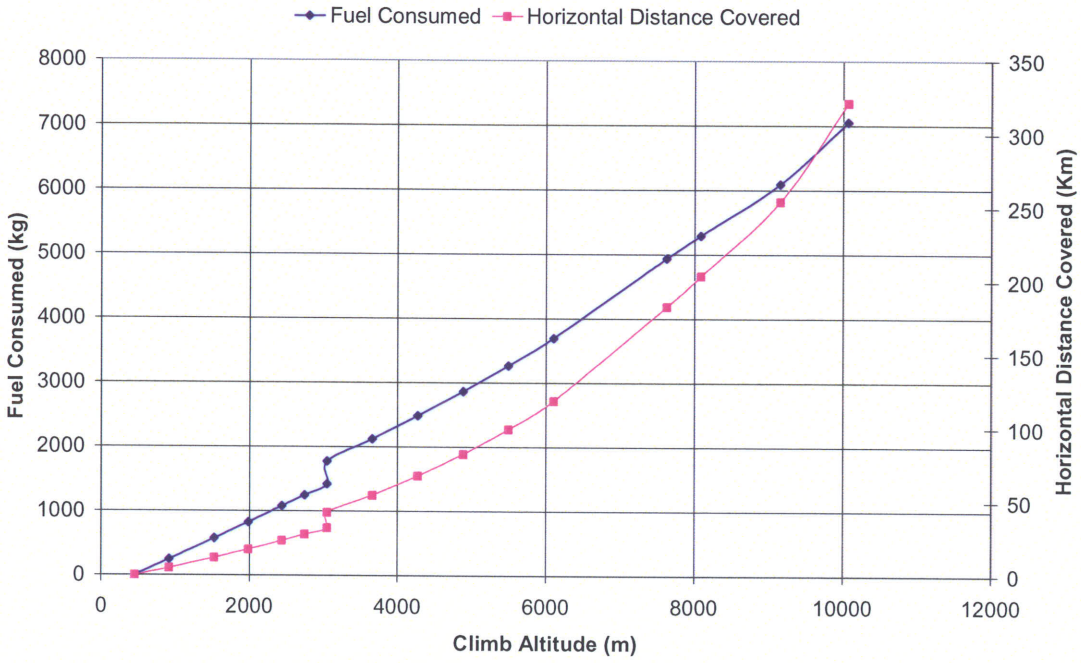


Figure 5.26: Horizontal Distance Covered and Fuel Consumed at Climb

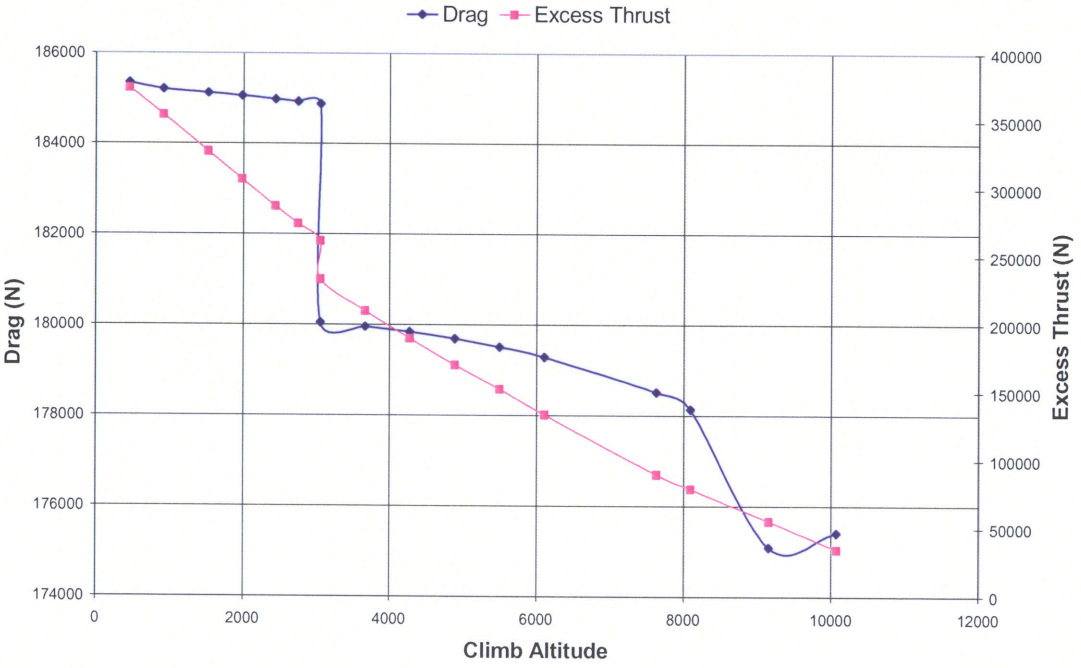


Figure 5.27: Drag and Excess Thrust at Climb

One of the main parameters normally used to assess the performance of the aircraft at the climb section is the rate of climb, which corresponds to the vertical distance covered per unit time. The rate of climb depends on the speed, thrust, drag and mass of the aircraft at the various stations of the climb, with speed having a dominant effect on all the parameters. As the aircraft climbs to a higher altitude its speed increases. The detrimental effect of speed on thrust is partly offset by the reduced mass and drag as well as the increase in aircraft's speed. In most cases, however, the main effect on the rate of climb is the reduction in excess thrust – defined as the thrust produced by the engines minus the drag of the aircraft.

At the early stages of the climb schedule, because of the relatively low speed the dominant effect on aircraft drag is the lift-induced drag. At 3000 meters, the increase in aircraft's speed causes the lift coefficient to reduce which in turn leads to a reduction in induced drag coefficient and total drag. This effect implies that at the early stages of the climb the aircraft flies at speeds lower than the minimum drag speed and hence any increase in flying speed would cause total drag to reduce. Following the acceleration at 3000 meters the rate of climb reduces continually. As the rate of climb reduces the vertical speed of the aircraft reduces as well and therefore the aircraft requires more time to climb to a given altitude, Figure 5.25.

The combined effects of increased engine sfc, reduced thrust and rate of climb, lead to a continuous increase in the fuel consumed per altitude climbed, Figure 5.26. Finally, because the aircraft requires more time to climb at a given altitude and the rate of climb reduces the horizontal distance covered increases with altitude, Figure 5.26.

5.10 Effects of Conventional Power Systems and Performance of All/More Electric Derivatives

The last section of this chapter deals with the effects of conventional secondary power requirements and the impact of adopting all or more electric technologies on the performance of the aircraft. For this reason the performance of the three baseline aircraft is compare to the performance of the same aircraft operating with clean engines and all electric derivatives. In the case of clean engines, extraction of secondary power has been eliminated and the engines were modelled and run with no off-takes. Adoption of all electric technologies, on the other hand, assume elimination of bleed air and the use of electrically driven compressors for the supply and operation of the environmental control system.

Flow rates and delivery pressures, required for estimating the power requirements for the compressors, were rescaled from previous studies and the Airbus A300-600 Standard Specification[5, 6]. Inlet flow conditions have been calculated at predefined cruise conditions (i.e. altitude and Mach No) and at ISA temperatures. In accordance with previous studies [7-15, 22-24], the pressure ratio of the compressor at cruise was set at 3.2.

In the case of the A340-500 class aircraft the power extracted from each engine for the operation of the electrically driven compressors was compared against the bleed air power extracted in the case of conventional secondary systems. The results are shown in Figure 5.28. The power extracted in the form of bleed air has been converted to shaft power consumed by the engine's compressors to supply the bleed air at the pressure required.

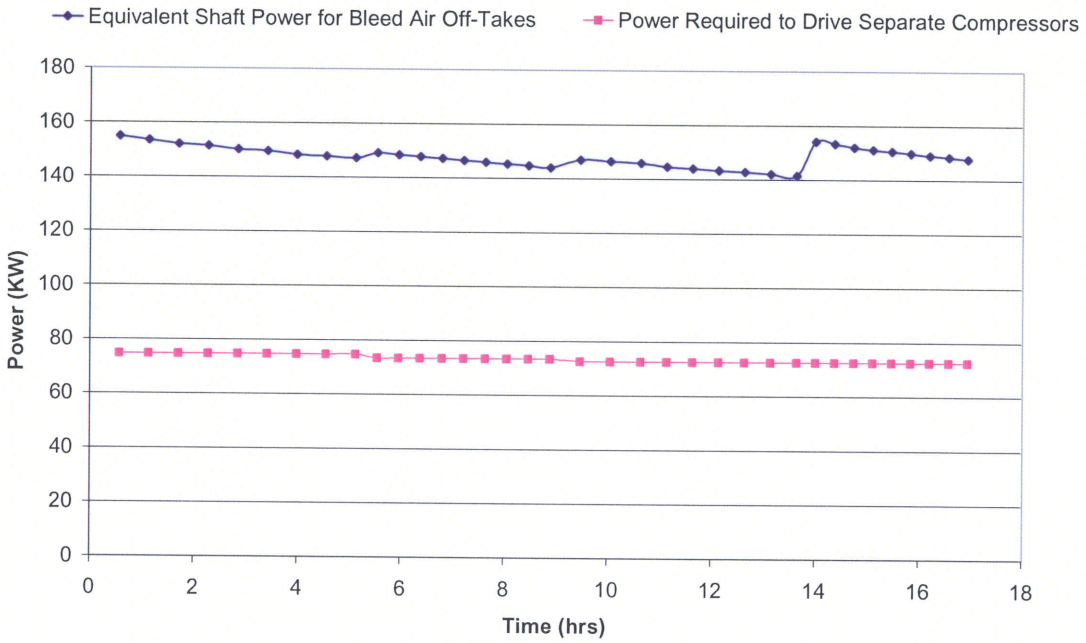


Figure 5.28: Power Required for the Operation of the ECS for A340-500 Class Aircraft AEA Version Vs Conventional SPS

In the case of bleed air extraction the power varies with altitude and power setting. Therefore, at each stage of the cruise the power extracted reduces with time. When the aircraft climbs to a higher altitude the engine pressure ratio increases and this causes an increase to the bleed pressure and hence power extracted. The pressure ratio of the

separate compressors, however, remains constant throughout the cruise phase leading to constant power extraction.

It is noted that the power required by the separate compressors was almost half the power extracted from the engines in the case of the conventional bleed air system. The findings emphasize the inefficiencies of the current bleed system and the fact that the location of the intermediate pressure port is a compromise and is dictated by both, the temperature and pressure requirements of the existing bleed air thermal anti-icing system and the requirements of the cabin pressurization system at the later stages of cruise. In other words, the temperature and pressure levels of the bleed air during most stages of the cruise section are considerably higher than those needed.

Since the AEE concept makes no assumption on how the power is distributed around the aircraft, initially it has been assumed that hydraulic and electric baseline loads will remain the same. As a direct result of the above assumption, the generators for the AEE version have been sized with the existing electric loads in mind plus the additional loads for the ECS compressors and the hydraulic system. As far as the performance of the propulsion system is concerned, and since the engines do not discriminate between hydraulic and electric power, the total amount of power extracted from the engines for the electric and hydraulic systems will remain almost unaffected, with a small increase incurring as a result of the generator efficiency. Thus, the main impact on engine performance is the replacement of bleeds with shaft power extraction. The power requirements per engine for the A340-500 class aircraft in the case of the all-electric version are shown in Figure 5.29. It is noted that the majority of the power is consumed by the environmental control system.

The impact, however, on the capacity size and mass of the generators for the AEE version is indeed significant and the capacity of each generator has been increased by 125 KVA. This figure takes into account the possibility of engine failures but allows for power management during such emergency situations. These issues along with the performance of the AEE version are discussed in the latter stages of this report.

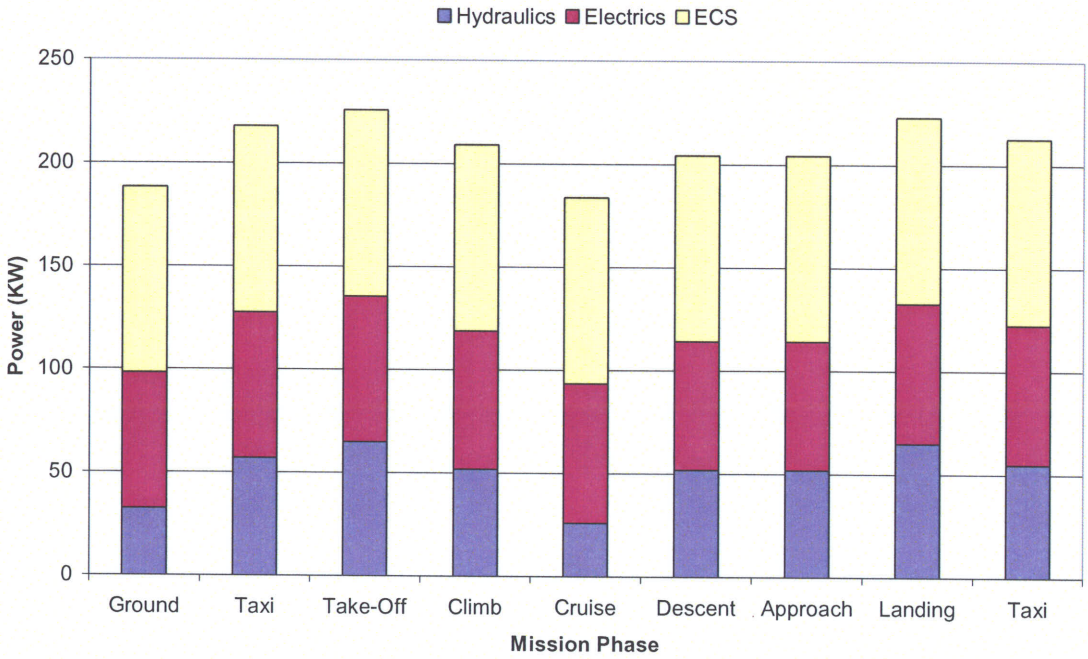


Figure 5.29: Engine Off-takes and Power Requirements at Various Mission Phases for A340-500 Class Aircraft - AEA Version

The effects of conventional and all electric systems on the performance of the aircraft and engine were studied at the various mission phases, with the emphasis placed on the climb and cruise phases. The analysis considered the effects on engine sfc, TET and thrust output. The performance of the aircraft was analysed in terms of fuel changes and their effect on aerodynamic efficiency. In the cases of fuel changes two operating conditions were considered. The first assumes adoption of all electric technologies at the cruise section only, while the second examines the effect of adopting all electric technologies for the whole mission profile, including the climb and descent sections.

5.10.1 Effect of Conventional and AEA Secondary Power Requirements on Engine Performance

The effects of secondary power requirements at the cruise section on the performance of the A340-500 class aircraft are shown in Figures 5.30 and 5.31 In addition to the conventional and all electric cases, the case of a three-port engine is also shown. The latter case assumes the use of a third port, where the bleed air extracted has similar pressure to the air delivered by the separate compressors.

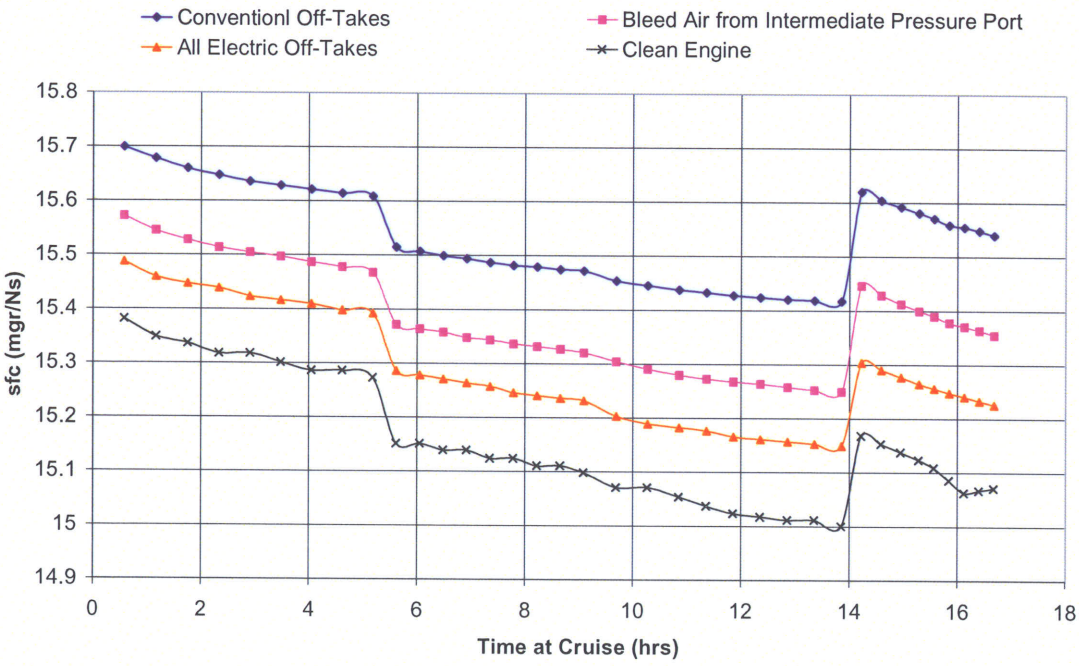


Figure 5.30: Effects of Off-Takes on sfc – A340–500 Class Aircraft

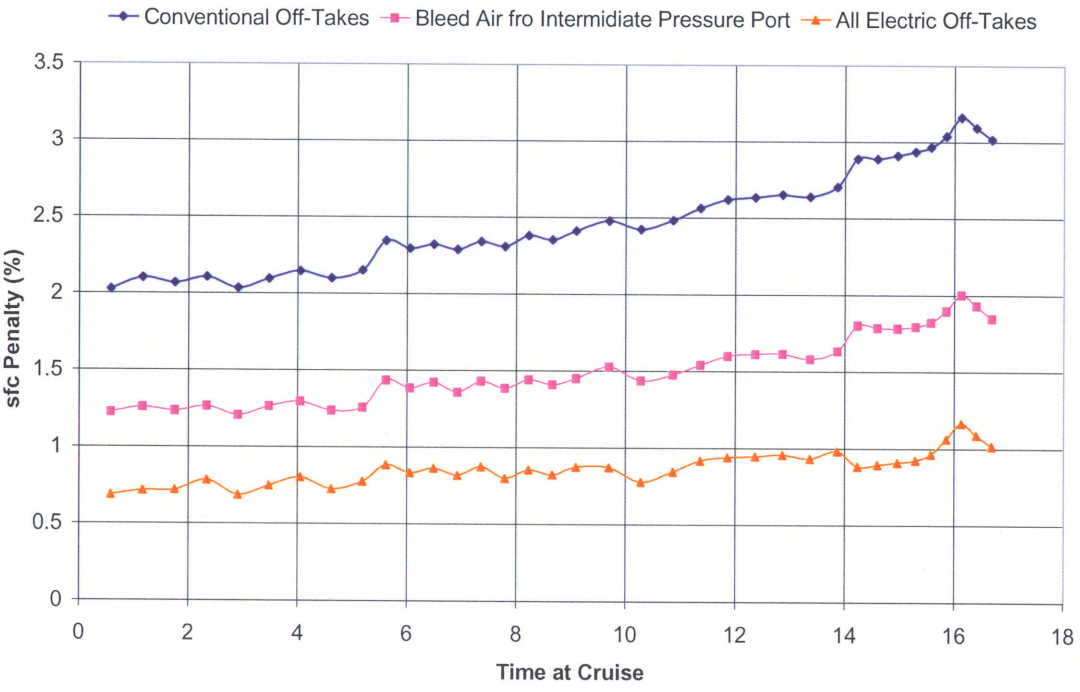


Figure 5.31: Effects of Off-Takes on Engine sfc – A340-500 Class Aircraft

In the case of the low pressure port, the pressure and temperature of the bleed air are those needed for the operation of the environmental control system at cruise conditions. More information regarding the configuration of this system is given in the next chapter of the thesis. The introduction of a third port along the intermediate pressure compressor and the lower temperature of the bleed air, eliminates the need to use the pre-cooler at the cruise phase. As a result, bleed air from the bypass stream can be eliminated. The use of a third bleed port could improve the performance of the engine and as shown in Figure 5.33 sfc penalties could be half those incurred in the case of the conventional bleed air system.

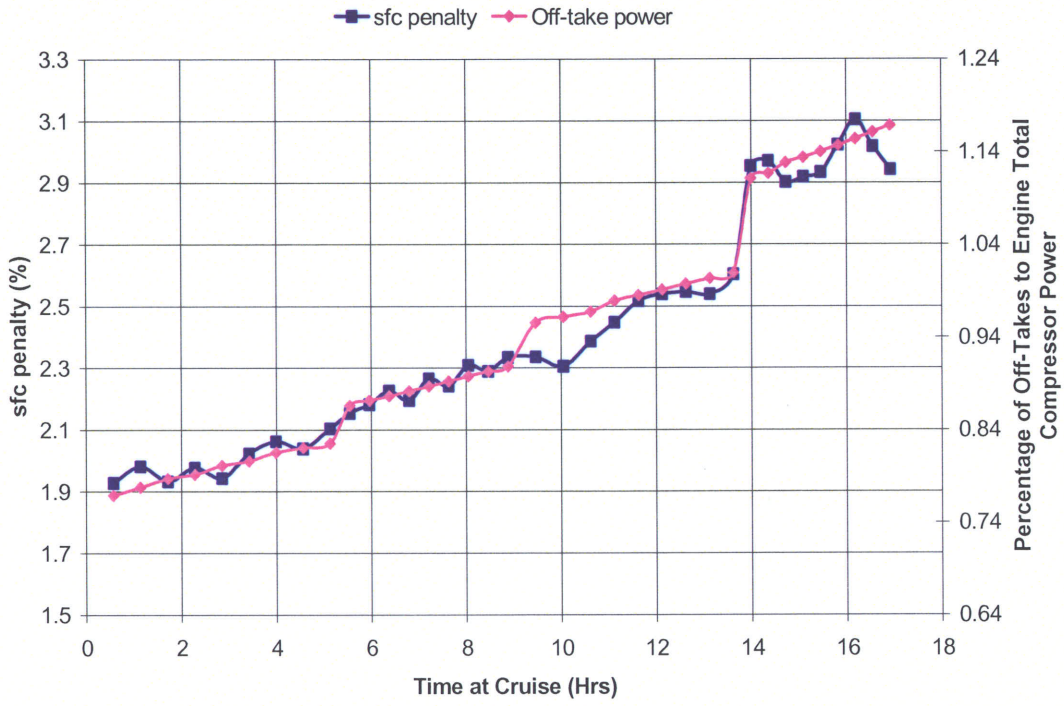


Figure 5.32: Variation of sfc Penalties and Off-takes

Although the equivalent power extracted in the case of the intermediate port is similar to the electrical power off-takes, specific fuel consumption penalties are still higher. Specific fuel consumption penalties in the case of bleed air can be expressed by equation 3.40, derived in Chapter 3 and shown below:

$$\frac{sfc_B}{sfc} = \frac{\left(\frac{T_4}{(1-\beta)} - T_2\left(1+\left(\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right)\right)}{\left(T_4 - T_3\left(1+\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right) * \frac{1}{\sqrt{1-\beta}}} \tag{3.40}$$

Similarly sfc penalties in the case of shaft power extraction are given by:

$$\frac{sfc_{ShaftPower}}{sfc} = \frac{\left((T_4 * (1+\beta)) - T_2\left(1+\left(\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right)\right)}{\left(T_4 - T_2\left(1+\frac{PR^{\frac{\gamma-1}{\gamma}}-1}{\eta_C}\right)\right) * \left(\sqrt{1+\beta}\right)} \tag{3.44}$$

In both cases T_4, T_2 and PR refer to the conditions of the clean engine and are fixed. Dividing 3.44 with 3.40 gives:

$$\frac{sfc_{Shaft\ Power}}{sfc_B} = \frac{\sqrt{1-\beta^2}}{1}$$

For values applicable to this analysis this ratio is always lower than unity and hence the sfc penalties in the case of shaft power extraction are lower than equivalent bleed air penalties. It is interesting to note, however, that the ratio approaches unity as the percentage of off-takes reduces. The latter implies that in terms of engine performance adoption of all electric technologies favour aircraft with high amounts of off-takes.

Figure 5.21 shows that sfc penalties increase with time. As the aircraft spends time at cruise, fuel is consumed and the power setting of the engine is reduced. Although the delivery pressure of the bleed air is reduced along with the power setting of the engine, bleed air mass flows remain constant and therefore the power extracted constitutes a larger percentage of the total power re-circulated within the engine. The effect is more noticeable when the aircraft climbs to a higher altitude and at the later stages of the cruise where the reduction in engine power setting is larger. The latter effect is also

shown if Figure 5.33, which plots the effects of conventional and all electric off-takes on the turbine engine temperature of the engine. The effect is more noticeable in the case of all electric off-takes. At the early stages of the cruise, the increaser in turbine entry temperature is minimal increasing, however, with time and altitude. In terms of turbine engine temperature penalties, the findings are in agreement with those of previous studies and the results presented in Chapter 3.

Conventional off-takes and extraction of bleed air lead to significantly higher turbine entry temperatures penalties than equivalent shaft power penalties. Equations 3.39 and 3.42 derived in Chapter 3 and shown below, can be used to model the effect of bleed air and shaft power off-takes respectively on turbine entry temperature. Therefore for bleed air off-takes:

$$\varphi = \frac{T_{4B}}{T_4} = \frac{1}{(1-\beta)} \tag{3.39}$$

and for shaft power off-takes:

$$\varphi_{Power} = \frac{T_{4B}}{T_4} = 1 + \beta \tag{3.42}$$

dividing 3.42 with 3.39 leads to:

$$\frac{T_{4\,Shaft\ Power}}{T_{4\,Bleed}} = \left(1 + \beta_{Shaft\ Power}\right)\left(1 - \beta_{Bleed}\right)$$

Once again, T_4 refers to the turbine entry temperature of the clean engine and is fixed. Assuming that the difference of $\beta_{\text{Shaft Power}}$ and β_{Bleed} is, then the above equation becomes:

$$\frac{T_{4\,Shaft\ Power}}{T_{4\,Bleed}} = 1 - \beta^2$$

Similar to the case of sfc the above ratio is less than unity and reduces with increased off-takes.

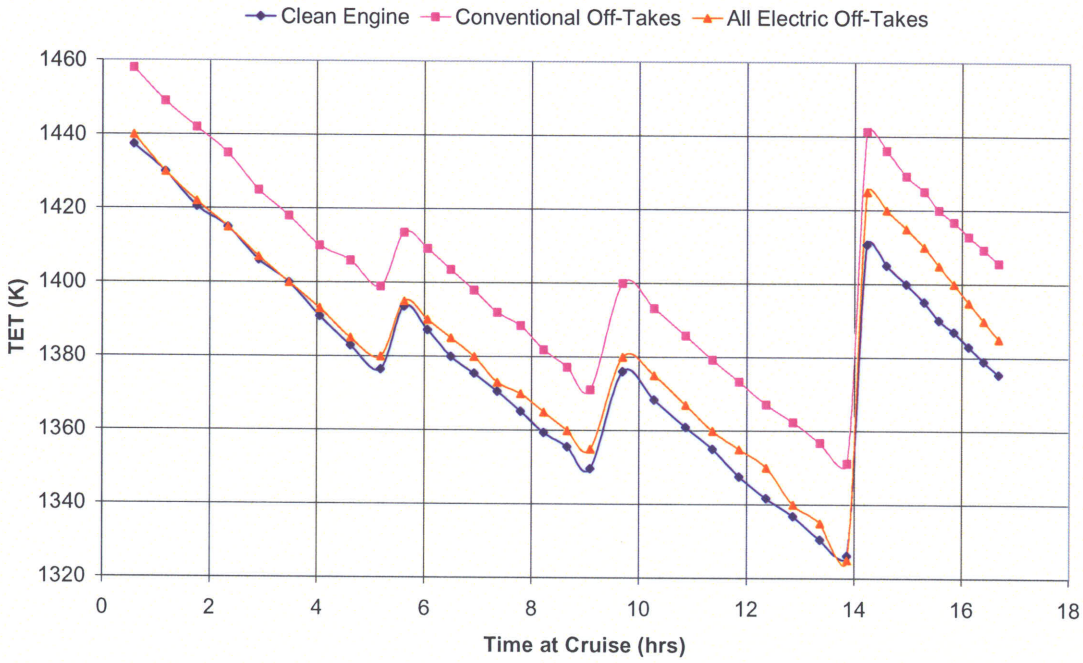


Figure 5.33: Effects of Off-Takes on TET

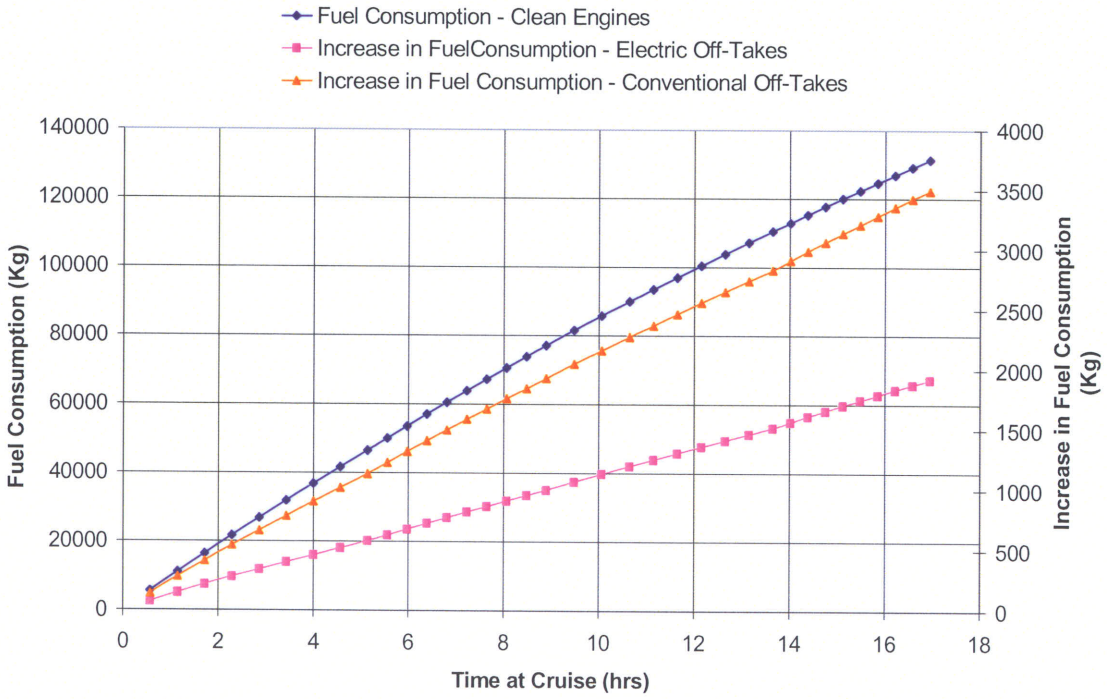


Figure 5.34: Effect of Off-Takes on Fuel Consumption at Cruise

The effect of conventional and all electric off-takes on the fuel consumed at the cruise section is shown in Figure 5.34. The results assume that the aircraft operates at constant maximum take off weight and the increase in fuel consumption in the case of conventional and all electric off-takes is attributed to the deteriorated performance of the engine. It is noted that relative to the clean case, extraction of conventional off-takes require 3500 kilograms of extra fuel. Assuming that each passenger weights 100 kilograms – including the mass of passenger baggage, the extra fuel required equals to 35 more passengers. In the case of the al electric off-takes the fuel penalty is reduced to 1900 kg, which corresponds to 19 more passenger.

5.10.2 Impact of All Electric Technologies on Engine sfc and Aircraft Fuel Usage

The impact of all electric technologies on the performance of the baseline engines is shown in Figure 5.35. It is noted that from the three engines modelled the largest sfc savings correspond to the Trent 900 class engine which is used with the A380-100 class aircraft.

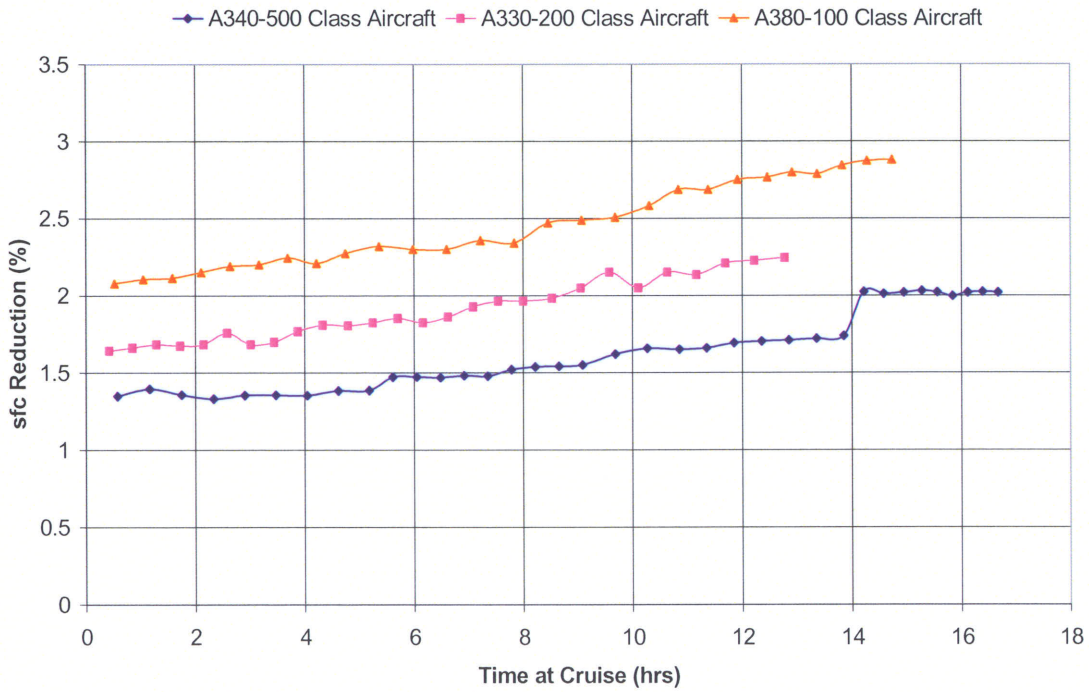


Figure 5.35: sfc Reduction

This result is the combined effect of two factors; because of the high bypass ratio of the engine and the increased number of passengers the bleed air requirements constitute a larger percentage of the mass flow through the core. The second effect is of increased engine pressure ratio. From the three engines modelled, the Trent 900 class engine has the highest-pressure ratio. Although the turbine entry temperature of the engine is higher than that of the two remaining engines, the increased pressure ratio predominates and coupled with the increased bypass-ratio leads to higher sfc penalties with extraction of off-takes. As a result elimination of bleed air has a larger effect and leads to increased sfc reductions. The Trent 700 class engine also experiences larger savings relative to the Trent 500 class engine. In this case, the result is the combined effect of three factors. Because of the design characteristics of the aircraft, at the cruise section of the mission profile the power extracted form the engines constitutes a large percentage of the power re-circulated in the core. As in the case of the Trent 900 class engine, the second effect is of increased pressure ratio. Both, at the design point and the cruise section the pressure ratio of the engine is larger than that of the Trent 500 class engine. The above effect coupled with the low turbine entry temperature lead to increased sfc penalties with extraction of off-takes. Although the T700 class engine has a lower bypass ratio engine, the above effects predominate causing sfc penalties to reduce with the elimination of bleed air and the adoption of electric power off-takes.

Despite the limitations discussed in the previous sections of this chapter the Brequet equation is a useful tool for analysing the performance characteristics of different aircraft. Equation 5.16, shown below, can be used to study the design efficiency of an aircraft, in terms of propulsion, aerodynamic and finally structural efficiency.

$$Range = \frac{V\ E}{sfc\ g} \ln\left(\frac{m_1}{m_2}\right)$$

It can be assumed that the mass m_2 at the end of the cruise is equal to the zero mass fuel of the aircraft and consists of the mass of the structure, propulsion system, secondary power systems and payload. Similarly the mass m_1 at the start of the cruise can be taken as the maximum take of mass of the aircraft. The mass of the fuel is then equal to:

$$m_f = m_1 - m_2$$

and the mass ratio is:

$$\frac{m_1}{m_2} = 1 + \frac{m_f}{m_2}$$

The above equation implies that as the range of the aircraft increases so does the ratio of fuel mass to zero fuel mass. As a result both the fuel mass and the take-off mass of the aircraft increase with payload and range. This is also true when considering the design characteristics of the baseline aircraft, shown in Figure 5.36. The characteristics are plotted against the product of the range and the payload, which is a parameter normally used at the deign stage of the aircraft. For comparison the characteristics of the Airbus A300-600 aircraft are also included.

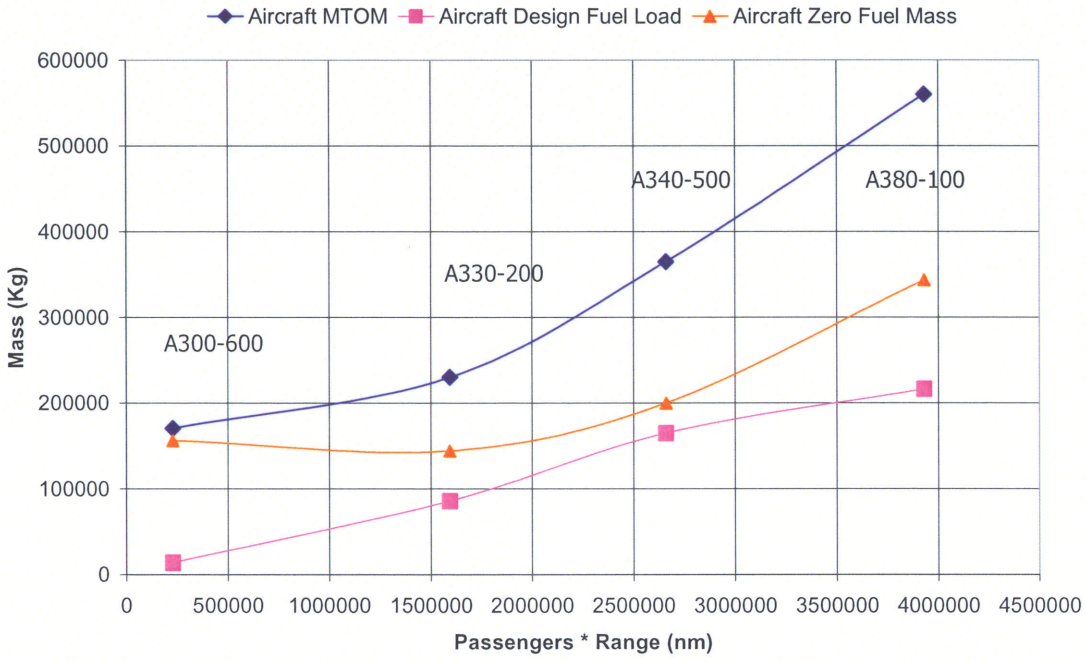


Figure 5.36: Aircraft Design Characteristics

Figure 5.36 shows that both the zero fuel and maximum take-off mass of the aircraft increase rapidly and non-linearly with payload and range. The increase in maximum take off mass leads to an increase in aircraft drag at the cruise phase. Secondary power requirements, however, increase linearly with passengers and hence the proportion of off-takes, and bleed air in particular, to the thrust delivered by the engines is higher in the case of the A330-200 and A380-100 class aircraft. This effect is shown in Figures 5.37 and 5.38.

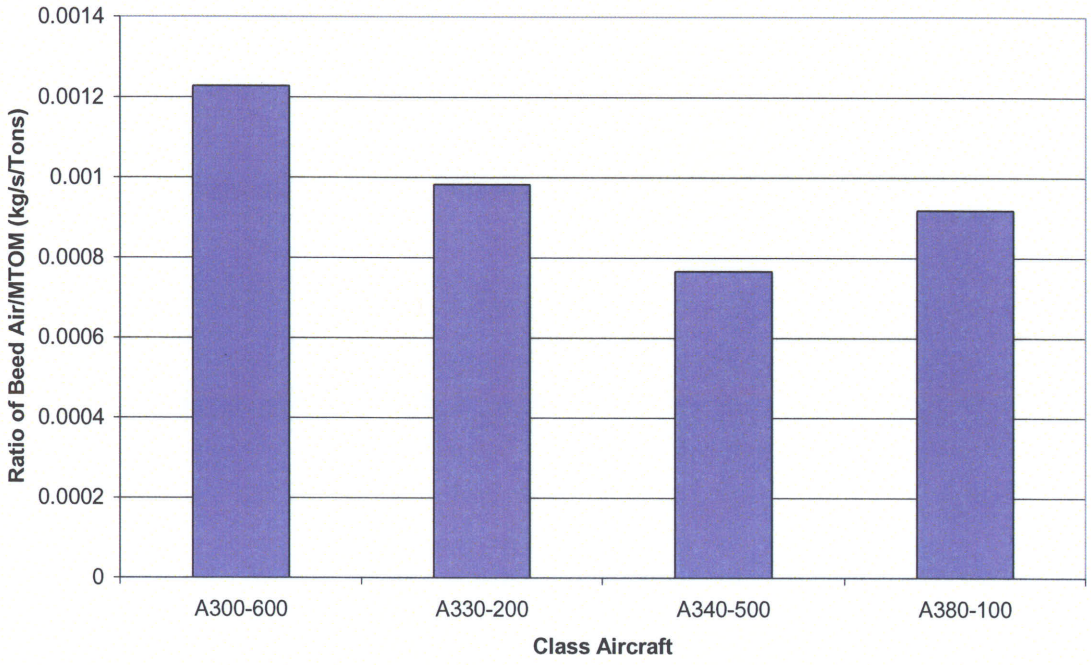


Figure 5.37: Ratio of Bleed Air to Aircraft MTOM

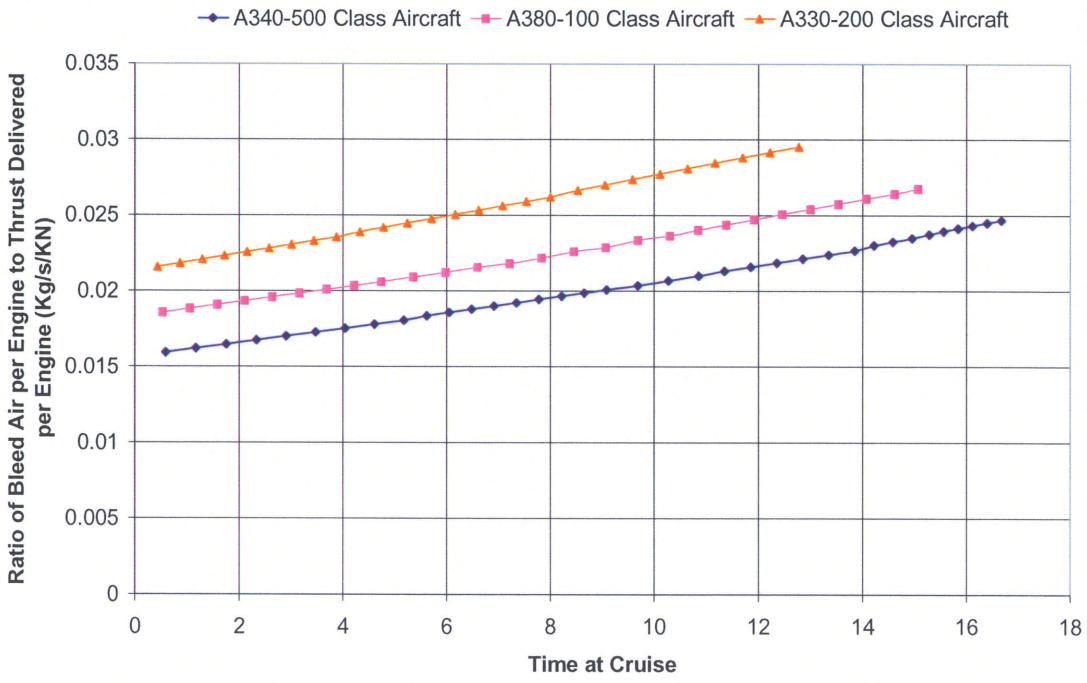


Figure 5.38: Ratio of Bleed Air to Engine Thrust at Cruise

The better aerodynamic efficiency of the A330-200 aircraft cause the ratio of bleed air to engine thrust to increase at the cruise section. Although, this ratio is higher for the Trent 700 class engine, the increased bypass ratio of the Trent 900 class engine leads to higher sfc savings with the elimination of bleed air.

The impact of all electric technologies and elimination of bleed air on fuel consumption is shown in Figures 5.39 – 5.43. The results shown in Figure 5.39 assume elimination of bleed air at the cruise section only. The findings of previous studies for an Airbus A300-600 class aircraft undertaken by Canfield College of Aeronautics during the mid 1980's are also included.

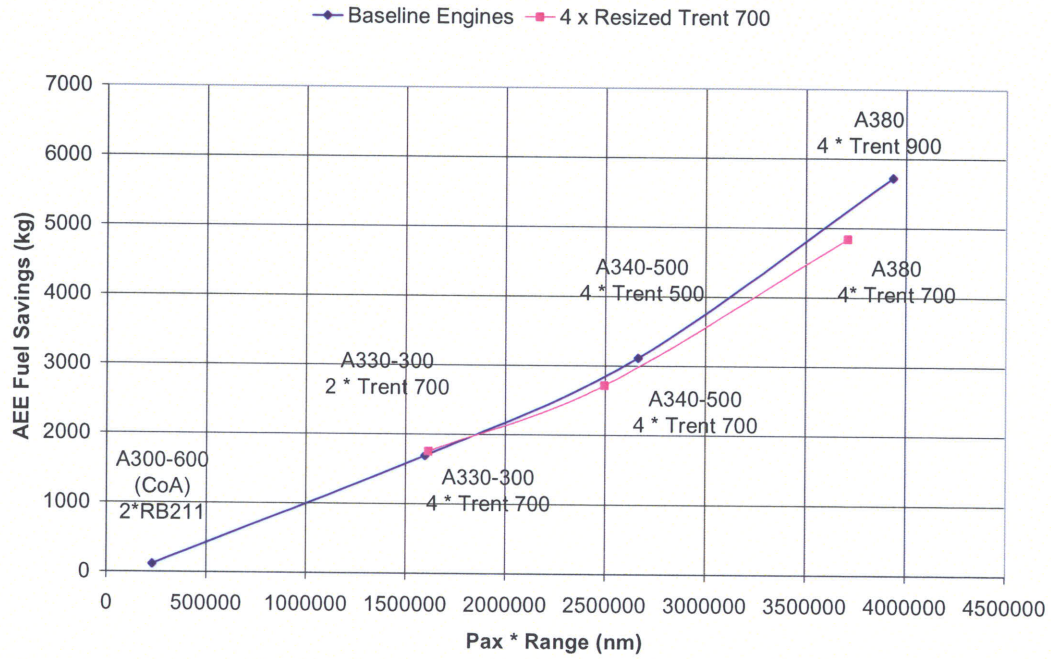


Figure 5.39: Fuel Savings with Elimination of Bleed air at Cruise

To be able to study the effect of engine bypass ratio on fuel savings, the baseline aircraft were also modelled with hypothetical resized Trent 700 engines. The pressure ratio, bypass ratio and turbine entry temperature of the Trent 700 class engine was kept constant and the mass flow increased accordingly to achieve the thrust required at sea level static conditions for the A340-500 and A380-100 class aircraft. The low bypass ratio of the Trent 700 hundred engine leads to an increase in sfc and therefore the range of the A340-500 and A380-100 class aircraft reduces.

Fuel savings in both cases, i.e. baseline and resized Trent 700 class engine- follow the same trend increasing with payload and range. The resulting increase in fuel consumption is the net effect of two factors. In the case of the A330-200 class aircraft the increased sfc savings are off-set by the shorter range of the aircraft. The opposite is true in the case of the A340-500 class aircraft, where the lower sfc savings are counteracted by the increased range of the aircraft. Finally, in the case of the A380-100 class aircraft the combined effect of high sfc savings and long range lead to a rapid and non-linear increase in fuel savings.

It is also noted that the effect of engine bypass ratio is relatively small in the case of the A340-500 class aircraft and becomes more noticeable in the case of the A380-100 class aircraft. This is despite the fact that in both cases the lower bypass ratio causes the same reduction in the percentage of bleed air relative to the mass flow into the core of the engine. The Trent 500 engine operates at higher turbine entry temperature and lower compressor pressure ratio than the Trent 700 engine. The Trent 900 class engine, on the other hand, has a similar compressor pressure but a higher turbine entry temperature than the Trent 700 engine. As shown in Chapter 2, increased pressure ratio has a detrimental effect, whilst high turbine entry temperature has a positive effect on the performance of the engine with extraction of off-takes. In the case of the A340-500 engine, the difference in these two parameters partly offset the effect of reduced bypass ratio and percentage of bleed causing a relative small reduction in fuel savings with reduced bypass ratio.

The percentage fuel savings relative to the baseline fuel load are shown in Figure 5.40. As shown above, aircraft with shorter cruise require a lower ratio of fuel mass to maximum take off mass. As a result the percentage savings in the case of the A330-200 class aircraft are higher than those of the A340-500 class aircraft.

Fuel savings resulting from the elimination of bleed throughout the mission profile are shown in figure 5.41. Once again, savings increase with range and payload. In the case of the A380-100 class aircraft the total reduction of fuel is more than 7500 kilogram reducing to 4000 kg and 2000kg for the A340-500 and A330-200 class aircraft respectively.

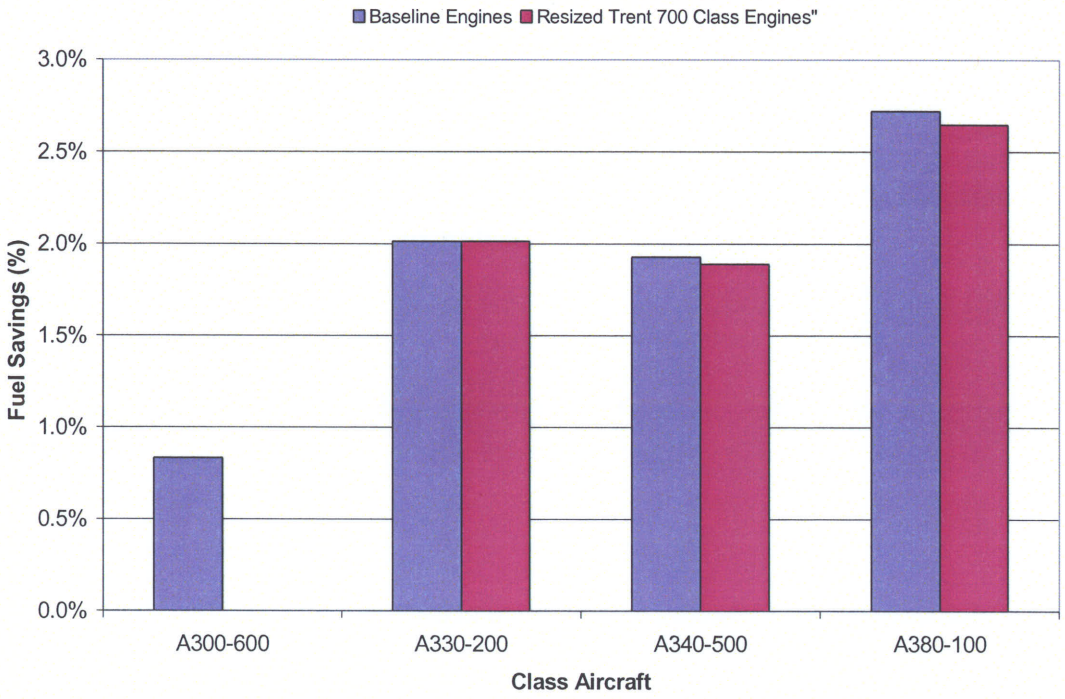


Figure 5.40: Percentage Fuel Savings with Elimination of Bleed Air at Cruise



Figure 5.41: Fuel Reduction – Elimination of Bleed Air at Cruise & Whole Mission Profile

5.10.3 Effects of All Electric Off-takes on Climb Performance

Although in terms of fuel savings the most important phase of the mission profile is the cruise, the climb performance is equally important in terms of time required to reach the top of climb conditions. For this reason this section of the mission profile is also considered in detail. From the three baseline aircraft the A340-500 class aircraft is affected more by a change in the extraction of secondary power. As in the case of the cruise, elimination of bleed air can improve the performance of the engine. At the cruise section, the more important effects are those of reduced sfc and turbine entry temperature. At the climb section, elimination of bleed air would also reduce the turbine entry temperature for a given thrust. If the turbine entry temperature is restored, the thrust output of the engine will increase along with the rate of climb. In the case of the A330-200 class aircraft the engines are oversized to cater for the one engine inoperative conditions. As a result the thrust output of the engine can be increased if needed without affecting the integrity of the engine. In the case of the A380-100 the increase in engine thrust would lead to increased noise, which as mentioned in Chapter 2 already limits the design characteristics of the engine. Therefore it is assumed that the A340-500 class aircraft will benefit more from the increase in engine thrust. The results are shown in Figures 5.42-5.44.

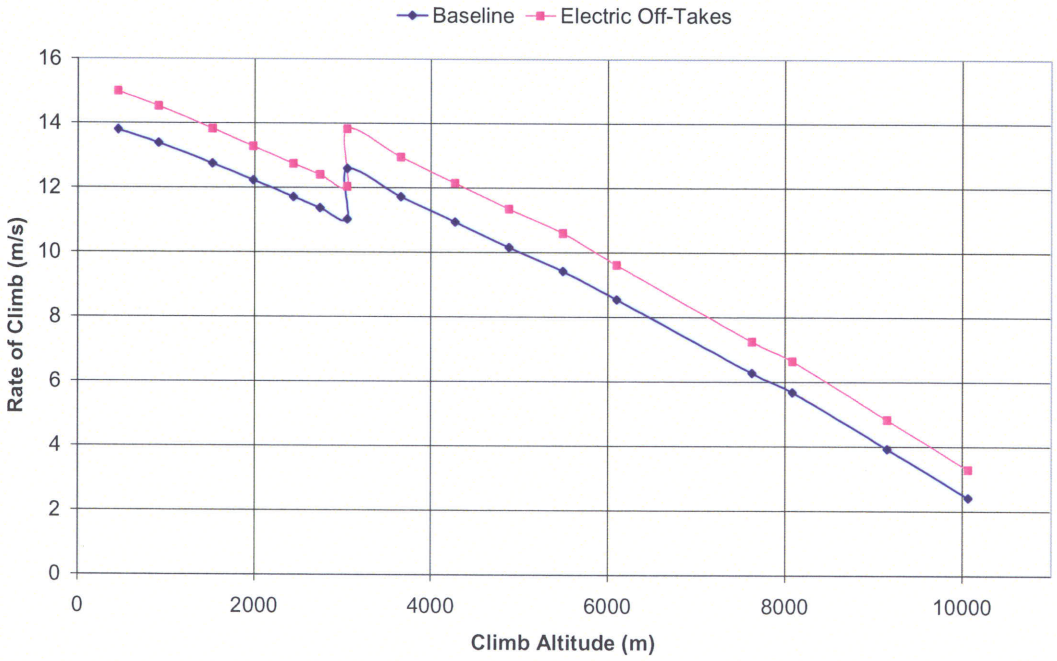


Figure 5.42: Effect of Electric Off-Takes on Rate of Climb

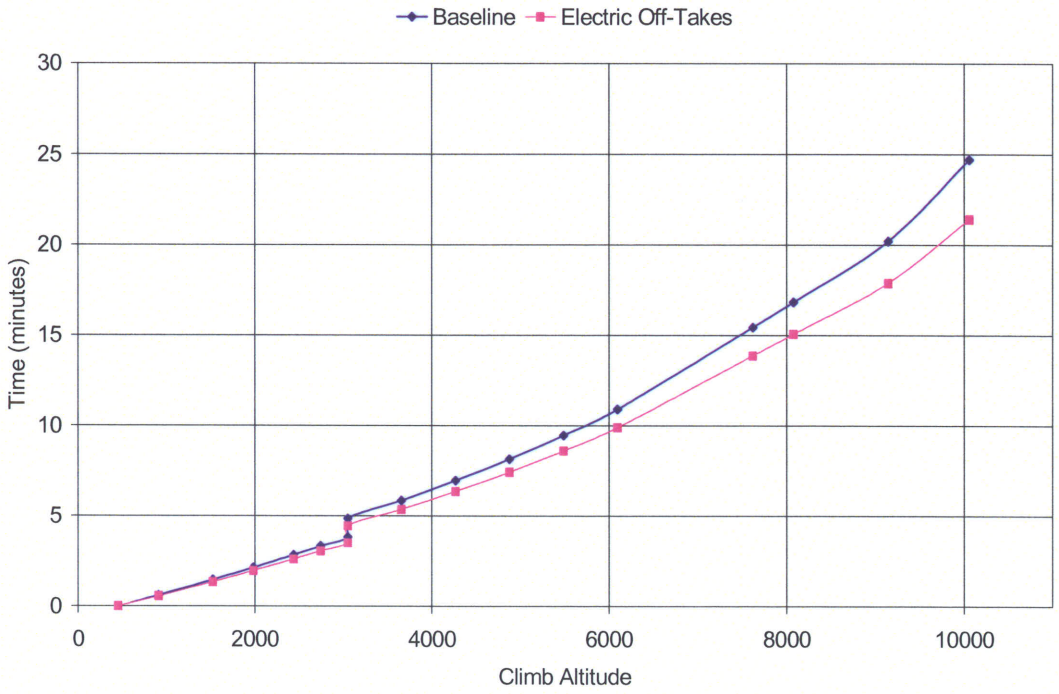


Figure 5.43: Effect of Electric Off-Takes on Time to Reach the Top of Climb

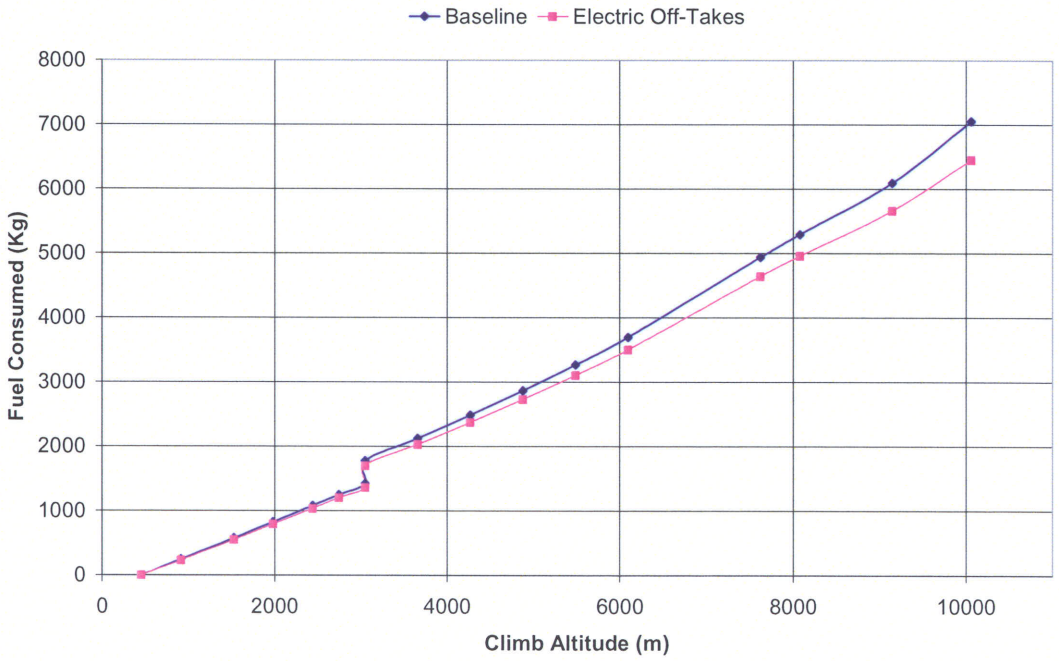


Figure 5.44: Effect of Electric Off-Takes on Fuel Consumed at Climb

Chapter 6: Definition and Limitations of Conventional Secondary Power Systems

The chapter considers the operation and layout of typical secondary power systems including environmental control and pressurization, anti-icing, hydraulic and electrical systems. The limitations and implications of conventional systems are presented and discussed; alternative system configurations are also considered.

6.1 Introduction

The chapter considers the generation, distribution and utilisation of secondary power on present aircraft. In the case of civil transport aircraft, the definition of secondary power is as that not used for propulsion, this being primary power. Thus, secondary power systems are the systems that distribute power throughout the airframe. On military aircraft, however, the term also includes the systems used to extract power from the engines and transfer it to airframe and engine accessories.

For the purpose of this study, three types of secondary power systems are considered and presented separately. In each case, a basic description of the system with present and past uses is given. Consideration is also given to the sources of secondary power.

6.2 Secondary Power Sources

6.2.1 Main Engines

The most obvious source of secondary power on an aircraft is the main propulsion unit. The engine can be used as a source of secondary as well as primary power. However, as explained before, provision of secondary power increases the mass, complexity and cost of the powerplant and in some cases it may compromise the design and performance for its primary role.

Turbofan engines are capable of supplying secondary power in two main forms:

- Shaft power
- Pneumatic power

A so-called 'power off-take shaft', driven by the engine's high-pressure shaft, drives a gearbox on which secondary power system components and engine accessories are

mounted. Three different tappings for compressed or bleed air are also provided at different stages along the engine's compressors.

As in the case of turbofan engines, turbojets are usually able to supply both shaft power and bleed air. Piston engines, however, are unable to supply directly compressed air and therefore all secondary power must be extracted in the form of shaft power. Turboprops, on the other hand, lie between piston and turbofan engines in that, although some units are capable of supplying some bleed air, amounts are limited and most power is extracted in the form of shaft power. Unlike turbofans, turboprop engines already have a large gearbox through which primary power is transmitted to the propulsor. This provides a second point from where shaft power can be extracted.

6.2.2 Auxiliary Power Units (APUs)

APU's are used whilst on the ground and when the main powerplants are not running. In such conditions, significant amounts of secondary power may be required. In order to be self sufficient in producing this power most large aircraft are fitted with an APU. Such units can supply secondary power in both shaft power and compressed air forms, depending on the requirements of the aircraft. The manner in which the power is extracted can vary considerably and depends on the configuration of the APU. The principle of operation of the two most common options is described in Chapter 4. Although in most cases APU's are used for the supply of secondary power on the ground, in some cases they are able to supply a proportion of secondary power required during flight. The provision of secondary power during flight, however, has implications on the design of the inlet and the exhaust and increases the complexity of the system.

6.2.3 Additional Sources of Secondary Power

On the ground, secondary power can be provided to the aircraft from an external power unit that can be either stationary or mobile, i.e. ground cart. Such units can supply both bleed air and/or electrical power.

In the air, limited amounts of electrical power can be supplied by packs of batteries. In the case of civil transport aircraft and with all engines inoperative, such packs are usually capable of powering vital electrical equipment for around 30 minutes while providing for a number of attempts to start the APU.

A Ram Air Turbine (RAT) device that is used to supply electrical or hydraulic power in the case of failure of all other secondary power sources. This device is deployed during emergencies to extract power from the free stream air.

Finally, in the case of high performance-low stability military aircraft, to which hydraulic power is essential at all times to maintain control, a source of power is necessary for the period between losing all engine supplied secondary power and its restoration or its provision by other slower acting systems, i.e. a RAT. These sources are usually called Emergency Power Units.

6.3 Pneumatic Power Systems

In the case of large civil transport aircraft, the largest amount of secondary power is provided in the form of high temperature pressurised air. Pneumatic power can be used by the following systems:

- Environmental control and pressurisation system
- Engine starting system
- Ice protection system
- Additional pneumatic functions

6.3.1 Environmental Control and Pressurisation System

To maintain the pressure, temperature and oxygen levels required within the cabin a continuous supply of high-pressure air is necessary. Cabin pressurisation, alone, requires moderate mass flows and pressures. However, the need to supply fresh air at the required temperature levels increases the pressure and mass flow demands placed on the environmental control and bleed air system. Cabin and equipment temperature control also implies a significant expenditure of energy to supply the cooling required.

To determine the heat removal or addition required from the cabin a heat balance is performed for different operating conditions of the aircraft. Such an analysis considers the various heat sources and sinks available. Typical components considered are presented below.

Metabolic Heat Output of Passengers and Crew Members: Humans reject heat in two distinct ways. The first way is by conduction, convection and radiation due to the higher temperature of the body relative to the surroundings. This is called 'sensible heat'. Heat is also rejected by evaporation of water, often referred to as 'latent heat'. Provided that the water vapour produced is removed from the cabin without condensation, latent heat does not affect the cabin heat balance. The overall heat rejection is related to the level of activity while the temperature of the surroundings affects the magnitude of each heat output. At low temperatures sensible heat predominates but as the temperature increases the ability to reject heat in this way reduces and more heat must be lost through evaporation.

Solar Radiation: Heat is transmitted to and from the cabin through radiation. Solar radiation increases with altitude and also depends on weather conditions. The heat input to the cabin will depend on the amount of radiation reflected and that transmitted by the surface into the cabin. The latter depends on many factors such as the material, paint and colour of the skin. Radiation through transparent surfaces is often a problem and its effect may become significant in the case of military aircraft.

Conduction Through Cabin Walls: Heat may enter or leave the cabin through the walls of the fuselage. The temperature within the cabin and ambient conditions would determine the heat rate transmitted to or from the cabin. At high altitude low Mach number operations heat is usually transmitted from the cabin to the atmosphere. The opposite is true in the case of low altitude and or high Mach number operations. Although the effective conductivity of the walls depends on the material and the configuration of the fuselage this can be modified by introducing an air duct between the inner and outer surfaces. Such an approach will limit the heat transfer effects and will result in a more uniform temperature distribution within the cabin.

Heat must be transferred from these sources to a heat sink and rejected from the aircraft. Available heat sinks include the outside air, internal fuel and oil. The outside air can be used either directly in the form of ram air or/and indirectly in the form of bleed air from the engines and the APU.

Ram air-cooling relies on the heat transfer to the air flowing round the aircraft. It is achieved by scooping air from the aircraft boundary layer. The air is forced into a scoop, through a heat exchanger and then rejected to the outside by the forward motion of the aircraft, Figure 6.1. Despite its simplicity the system posses a number of disadvantages including increase in aircraft drag and limited effectiveness at high

altitude low speed operation and while on the ground. Effectiveness can be improved through the use of a jet pump or cooling fan.

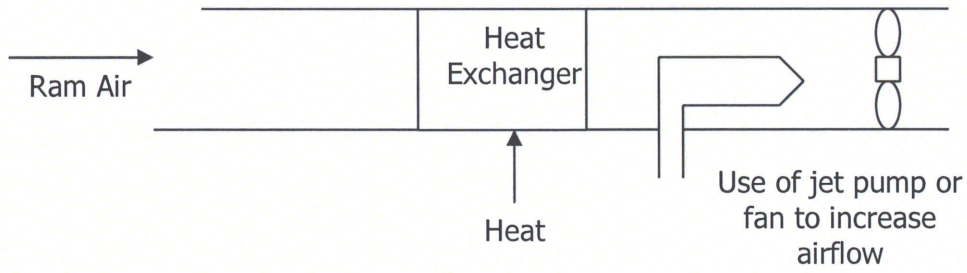


Figure 6.1: Simplified Ram Air Cooling System[79]

Fuel cooling systems are typically used to cool the hydraulic fluids and oil from the engine's accessories. Despite the high heat capacity of fuel, its application as cooling mean is limited because of the variation in fuel flow. A typical simplified fuel cooling system is shown in Figure 6.2. At low fuel flow operations the fuel is re-circulated back to the tanks and usually is cooled down by ram air. This prevents evaporation of fuel in the tank at the late stages of the mission profile and after landing, when fuel level is low.

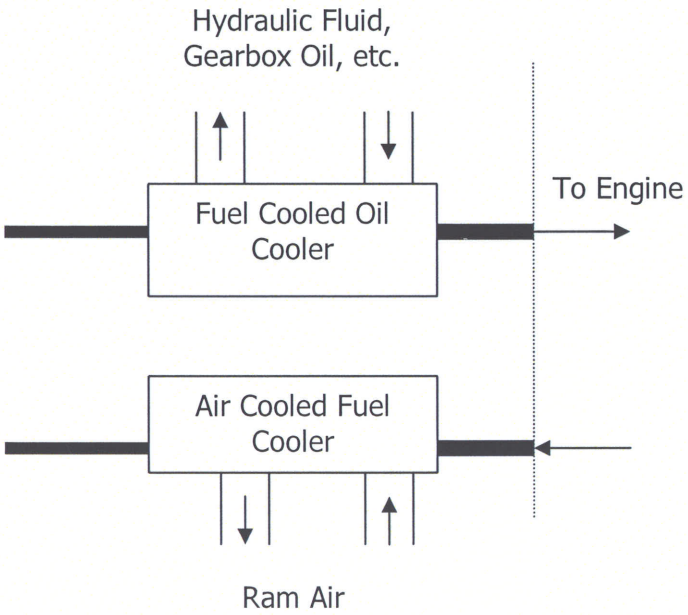


Figure 6.2 Simplified Fuel Cooling System[79]

As mentioned above, the main source of conditioning air is engine or APU bleed. This way the conditioning air can also be used to provide cabin pressurisation. In general, there are two types of bleed air systems:

- Open Loop System

- Closed Loop System

Open loop systems bleed large amounts of air from the engines or the APU, refrigerate it and then use it to control the temperature of the cabin and the equipment before rejecting it overboard. Closed loop systems collect the air after it has been used for cooling, refrigerate it and recycle it to use it again. Such a system reduces the amount of air bled from the engines and improves the efficiency of the engine. Despite their advantages closed loop systems have a number of disadvantages including cost, installation, mass and practical difficulties in collecting and re-using the conditioning air. For these reasons their application is limited and most present and past aircraft use some form of open loop system. The principle of operation of a closed loop system is shown on figure 6.3.

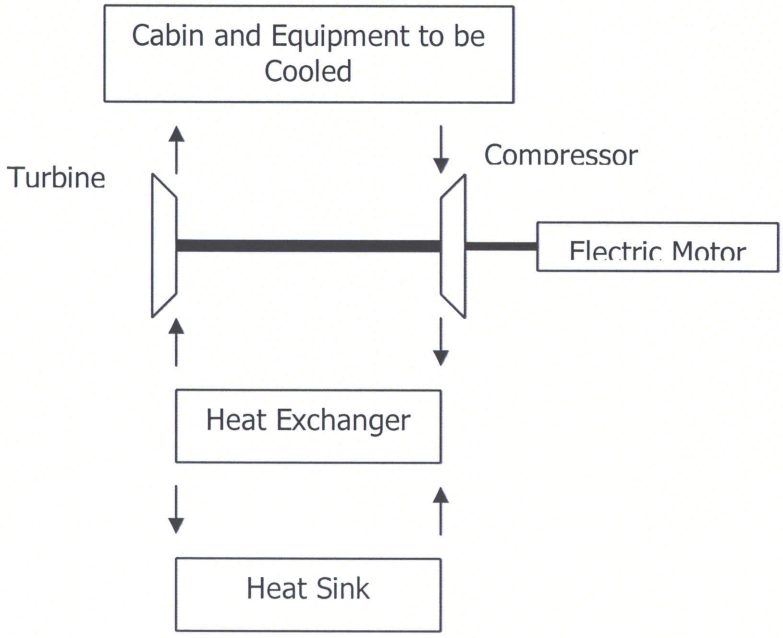


Figure 6.3 Simplified Closed Loop Cooling System[79]

Because the available cooling means are usually at a higher temperature than that required for cooling, some form of heat pump is usually necessary. For this reason

refrigeration systems are used. Two main types of refrigeration systems are usually available:

- Air cycle systems
- Vapour Systems

The basic air cycle system consists of a source of pressurised air, a heat exchanger and a cooling turbine. The process involves three main steps. First the air is compressed to a level higher than the pressure of the space to be cooled. Then the air is cooled down by a heat exchanger. Finally, the air is expanded and cooled through a turbine. The work delivered by the turbine can be used to drive the source of pressurised air and/or a fan used to draw ram air for the heat exchanger. The compressor and the fan also act as brakes, thus controlling the operation of the turbine. Such a system is often referred to as a turbo-fan system. A simplified diagram of the system is shown in Figure 6.4. The main advantages of the system are its simplicity and weight. However, the system possesses low efficiency and poor cooling effectiveness at low altitude and ground operations. In such cases, large amounts of pressurised air must be supplied to maintain temperature at required levels.

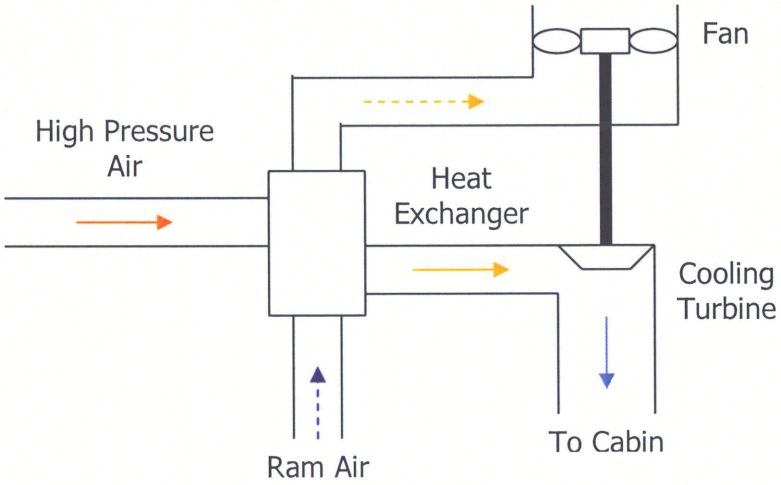


Figure 6.4: Simple Air cycle (Turbo-fan) System[133]

The most common variations of the basic air cycle system are the three-wheel bootstrap, evaporative and regenerative, or downstream chilled re-circulating, types. In the case of a bootstrap system, the pressure of the air delivered is increased to a higher level in the compressor section of the air cycle system before it is cooled down and expanded in the turbine section. The system consists of a primary heat exchanger,

a compressor, secondary heat exchanger and a cooling turbine. The bleed air is cooled by the primary heat exchanger. Then it is compressed to an even higher pressure and temperature in the compressor section of the system. The compression temperature in the air is removed by the secondary heat exchanger before entering the cooling turbine. The work output of the turbine is used to drive the bootstrap compressor and the fan used to draw ram air, hence the term three wheel system. The cooling mean in both the primary and secondary heat exchanger is usually ram air although other heat sinks such as fuel, oil and water may be used.

The use of a bootstrap compressor and secondary heat exchanger increases the cooling effectiveness of the system and reduces the required mass flow and pressure of bleed air. In the case of a turbo-fan air cycle, the pressure and temperature ratio of the turbine depends on the pressure of the bleed air. Therefore, for bleed air with fixed pressure the effectiveness of the system is limited thus requiring increased bleed mass flow and/or pressure. The use of a turbine driven compressor to increase the pressure of the bleed air increases the temperature and pressure drop at the turbine, thus improving the overall efficiency of the system in expense, however, of system weight. The main components of the system and the principle of operation are shown in Figure 6.5. Despite its advantages over the simple air cycle systems, the effectiveness of a bootstrap system is limited while on ground and at low speed operations. Variants of the bootstrap system described above are normally used in present and past high capacity civil transport aircraft.

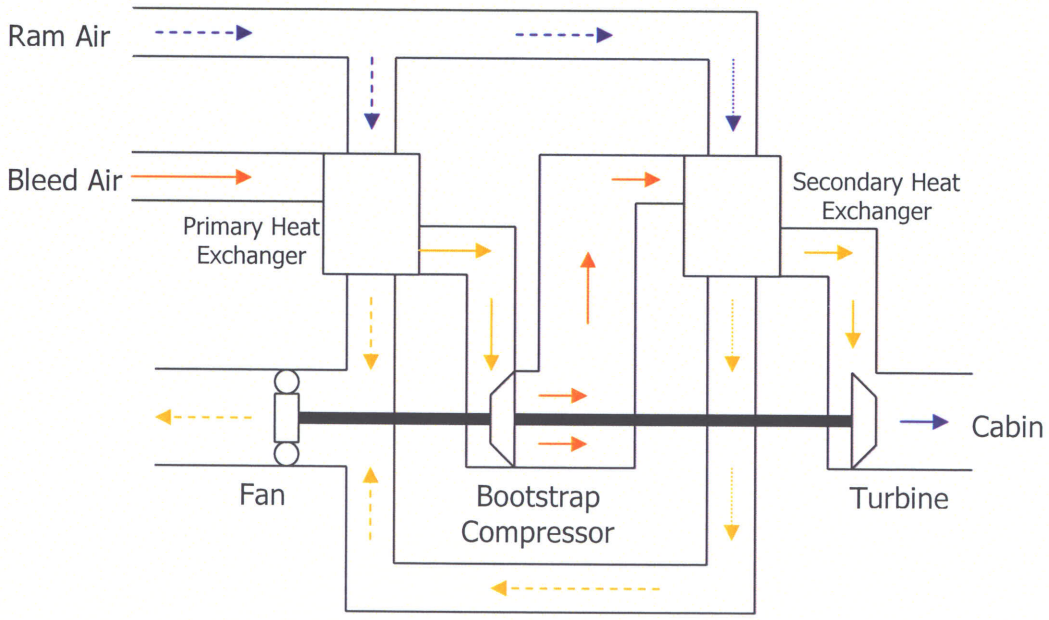


Figure 6.5: Simplified Three-Wheel Bootstrap Air Cycle[133]

The evaporative air cycle utilises the same components with the bootstrap air cycle or the simple air cycle system. However, it incorporates an evaporative cooler, which is placed downstream from the secondary heat exchanger and prior to the cooling turbine. An expendable type evaporant such as water, alcohol or ammonia is used as a heat sink for the evaporative cooler. Such a configuration increases the efficiency and effectiveness of the bootstrap cycle in the expense, however, of system weight, complexity, cost and maintenance.

The main disadvantage of the air cycle systems, previously described, is the limited cooling capability at high altitude or low speed operations where there is very little air available for cooling purposes. The effect is aggravated while on the ground, where the temperature of the ram air may be significantly higher than that required for cooling.

This problem is normally alleviated through the use of a regenerative air cycle type or a downstream chilled re-circulating cycle conditioning system. Once again the system incorporates the same components of the bootstrap cycle. Bleed air passes through the primary heat exchanger and the pressure and temperature increases in the compressor section of the air cycle system. Bleed air temperature is reduced in the second heat exchanger and then through the cooling turbine. However, instead of discharging the turbine-cooled air into the cabin directly, the air passes through a regenerative heat exchanger or a condenser to provide additional cooling to the air entering the cooling turbine. In such a system, the temperature of the air downstream the turbine may be sub zero and therefore a water extractor is incorporated upstream from the turbine to reduce the possibility of ice formation on the turbine blades.

The primary advantage of the regenerative air cycle system is that it extracts all the available heat energy from the exhaust air before finally discharging it into the cabin. Despite the complex ducting and additional equipment required it remains relatively simple and reduces significantly the pressure and mass flow of the bleed air required to provide cooling.

The system is typical of those used in modern high capacity civil transport aircraft such as those used in the present study. A simplified diagram of the system and its main components is shown in Figure 6.6. Normally, a bypass duct and temperature control valves are also incorporated to draw bleed air and deliver it downstream from the cooling turbine. Such a configuration is used to control the temperature of the air delivered into the cabin and provides additional heating in the case of cold operations.

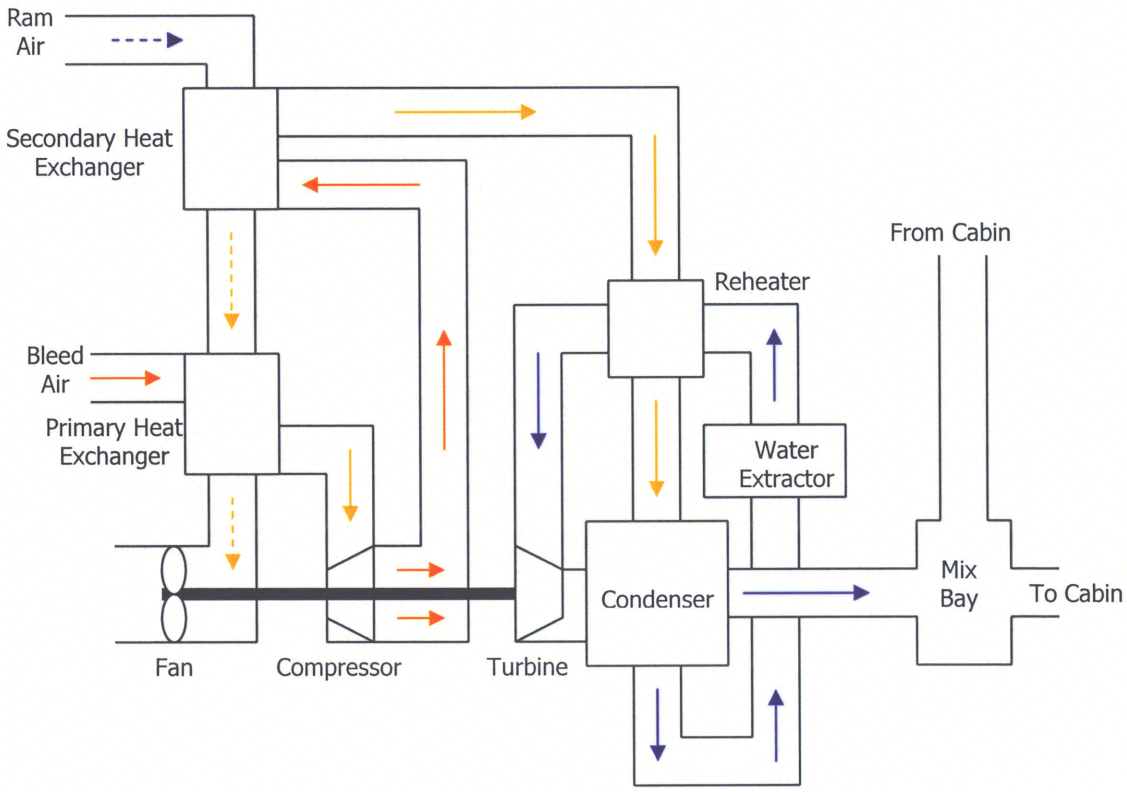


Figure 6.6: Simplified Downstream Chilled Re-circulating Air Cycle [67]

The vapour cycle system is a closed loop system where the heat load is absorbed by the evaporation of a liquid refrigerant. The refrigerant passes through a compressor before being cooled in a condenser, where the heat is rejected into a heat sink. A simplified diagram of the system and its components is shown in Figure 6.7.

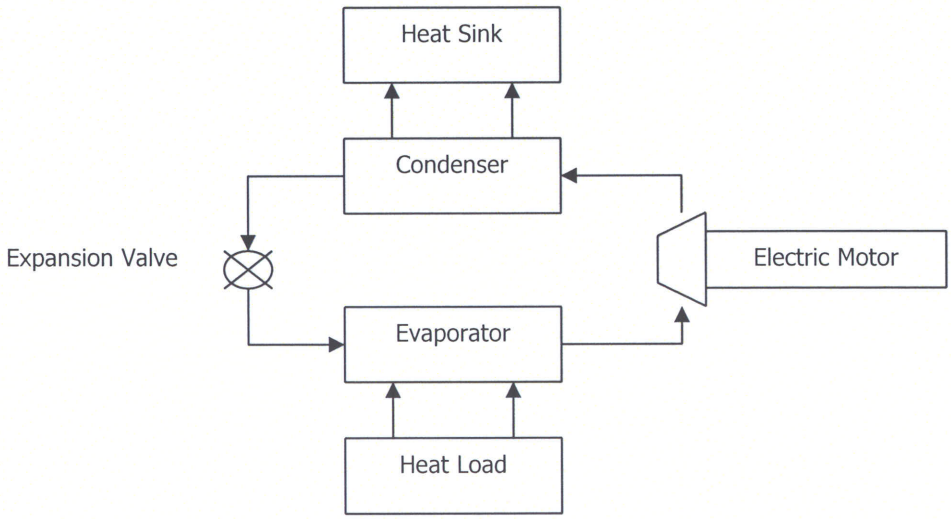


Figure 6.7: Typical Vapour Cycle [5]

Although vapour cycles are very efficient, they are heavier, more expensive, complex and suffer from poor reliability and maintainability compared to air cycle systems.

A combined air and vapour cycle system can also be used to limit the disadvantages of each system and improve the advantages. Such a system is shown in Figure 6.8. Pressurised air enters initially the air cycle system where it is pre-cooled in a heat exchanger. Then it enters the vapour cycle where more heat is rejected through an evaporator to a refrigerant. After leaving the evaporator the air expands through a turbine and then is discharged into the cabin.

The vapour cycle cools the pressurised air as it enters the evaporator. The refrigerant absorbs heat and then is passed to a compressor where it is compressed to a higher saturated state. The highly saturated vapour is then liquefied in a condenser. Part of ram air is used to as heat sink for the vapour cycle. The power delivered by the turbine is also used to drive the compressor of the vapour cycle.

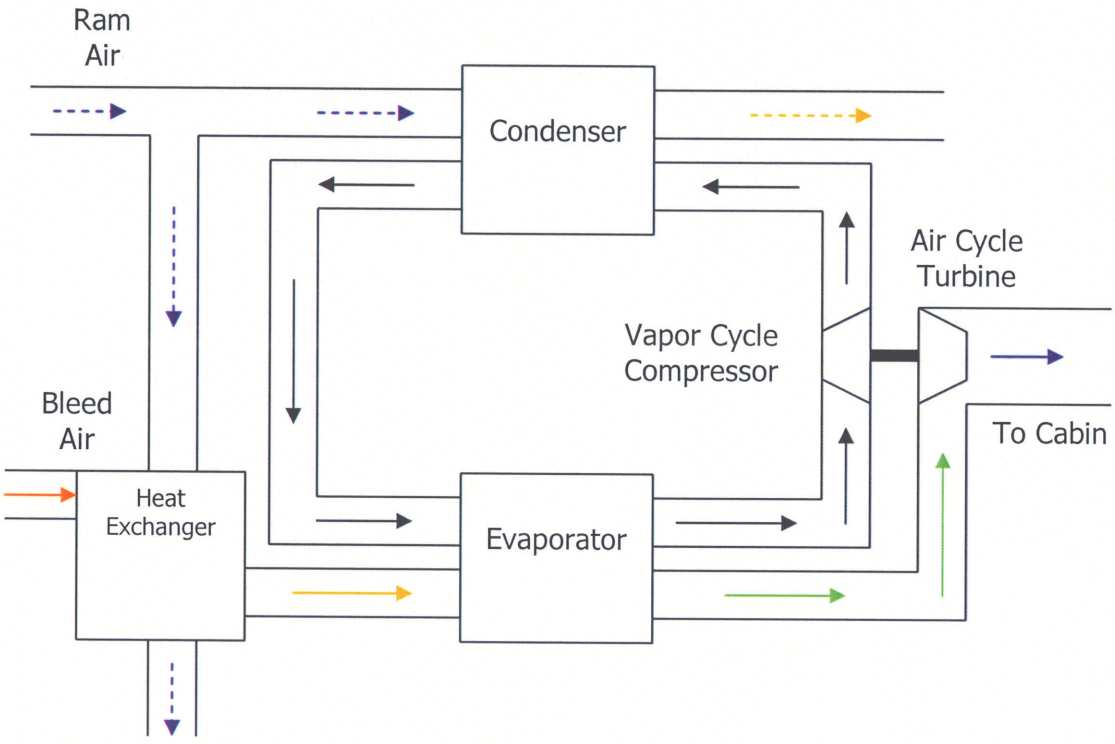


Figure 6.8: Combined Air and Vapour Cycle [67]

Such a combined cycle offers improved cooling efficiency thus requiring significantly lower bleed air pressures and mass flows.

6.3.2 Engine Bleed Air System

Compressed air is usually supplied by bleeding air from the engine's compressors. Such a system is shown in Figure 6.9. Because the available temperatures and pressures at a given bleed port change with altitude, flight Mach number and engine power setting, multiple ports are used to provide air close to the required pressure over the full operational envelope. Typically, air at a pressure of about 650 KPa and a temperature of about 100 °C is required to provide sufficient flow, pressure and temperature for pneumatic functions. However, the air bled from the engine compressors is often at higher pressures and temperatures than those required. For this reason a pressure control valve and a pre-cooler are normally incorporated. The bleed air pressure and temperature need to be reduced as soon as possible to the required values for safety reasons and to reduce the mass and complexity of the components. The pressure control valve maintains the downstream pressure to a fixed constant value regardless the pressure delivered by the engine. This action also controls the mass flow of the bleed air, which depends on the pressure difference upstream and downstream from the valve. Once the bleed air pressure is reduced to workable values the air passes through the pre-cooler where fan air is used to its temperature to around 100 °C. In some flight conditions, particularly on cold days, the outlet temperature of the pre-cooler may be less than the temperature required. For this reason a bypass duct is incorporated and some bleed air is directed downstream from the pre-cooler and then it is mixed with the cooled air to maintain constant temperature.

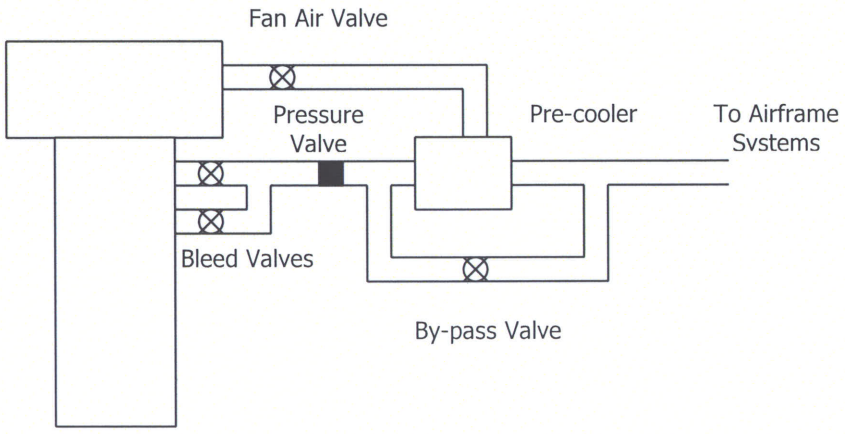


Figure 6.9: Typical Bleed Air System used on Past and Present Civil Transport Aircraft[67]

6.3.3 Pneumatic Engine Starting System

Since bleed air from the engines and the APU is used for the supply of the environmental control system, one of the simplest and lightest engine starting systems available is an air turbine system. This type of starting is used on the majority of larger gas turbine powered aircraft. The large ducts required to connect the main powerplants and APU to the air-conditioning packs coupled with the availability of high pressure air from these units offer the basis of a pneumatic engine starting system. Bleed air can be cross-fed from a running engine, a ground supply or the APU. The only additional equipment and penalty of an air starting system is that of a small length of ducting, shut-off valve and the turbine itself. A typical system is shown in Figure 6.10.

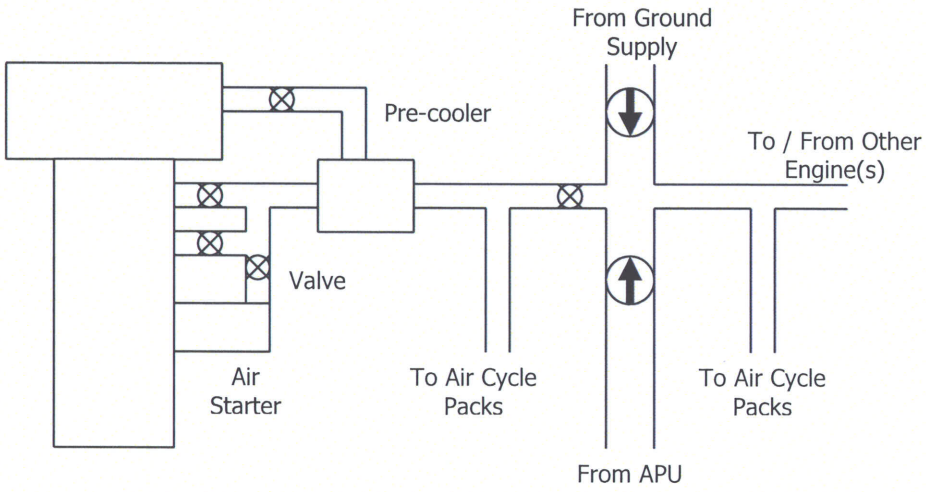


Figure 6.10: Diagram of a Bleed Air System with Additional Components for Air Starting[67]

6.3.4 Ice Protection Systems

Because of its high temperature, bleed air can be used to supply the energy required to prevent the formation and/or remove ice from various parts of the aircraft. Therefore, bleed air can be ducted to the critical parts of the aircraft and used to heat the surfaces. Two main types of ice protection are available:

- Anti-icing systems
- De-icing systems

Anti-icing systems prevent the formation of ice and are operated continually at conditions where ice may be formed. Ideally an anti-icing system should be incorporated as any ice formed on the wing can disturb the flow causing an increase in drag. However, provided that the ice formed is limited in extent, a de-icing system can be used to remove periodically the ice. Such a system is known as a de-icing system.

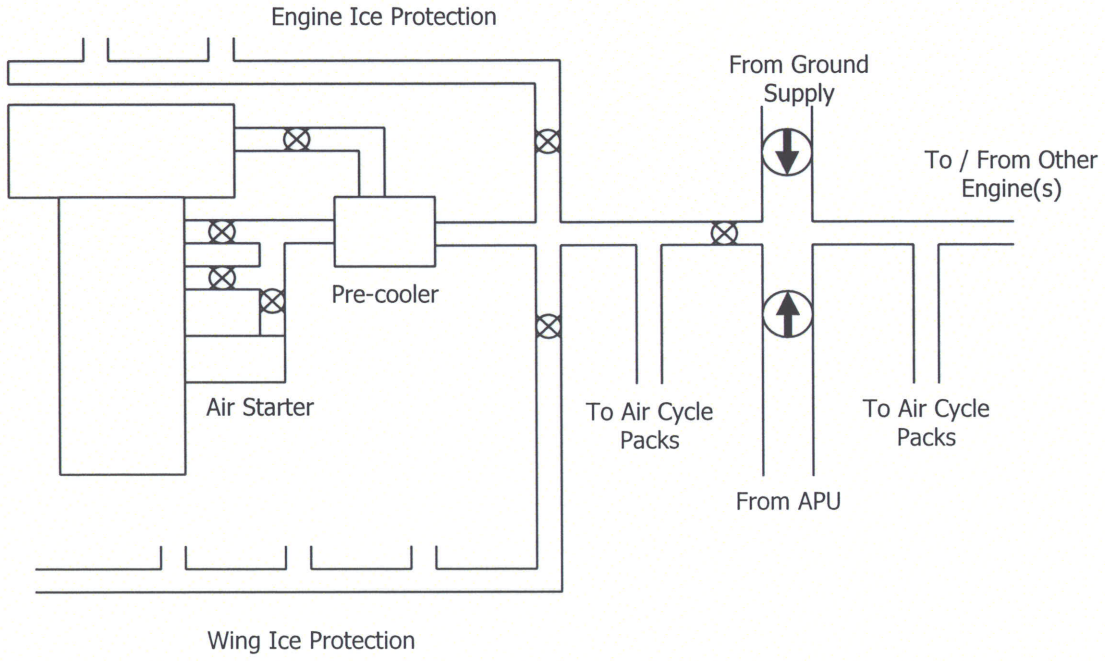


Figure 6.11: Schematic of a Pneumatic System with Bleed Air Thermal Ice Protection System[67]

In the case of large civil transport aircraft, a bleed air thermal ice protection system is incorporated. In such a case, ice protection is required for the wing leading edge and the engine cowl lips. Pre-cooled bleed air is ducted to a piccolo tube in the wing leading edge and the engine cowl lips. In the case of wing ice protection, the high temperature air impinges on the leading edge and is exhausted through ports on the lower surface of the leading edge. The system is normally sized and analysed to provide a so-called 'running wet' surface. This implies that impinging water flows beyond the heated zone and ice ridges may form aft of the leading and thermally protected area. For this reason, and to minimise power consumption, the wing ice protection system is normally operated in the de-icing mode. For the cowl anti-icing system, the pre-cooled air is ducted to the cowl spray tube in the engine cowl lips. The high temperature air impinges upon the leading edge of the cowl and is exhausted downstream of the impingement zone. The system is sized and analysed to provide enough heat for a totally evaporative surface, i.e. all impinging water is evaporated in

the heated zone. Therefore the power required for engine ice-protection is normally higher than the power required for the wing. The pneumatic system shown in figure 6.9 and 6.10 is modified by the addition of ducts to these areas, as shown in Figure 6.11.

On very large aircraft, such as those examined in this study, only part of the wing requires ice protection. As a rule of thumb, chords above 5 meters do not require ice protection and normally only the outboard portions of the wing are protected. If the thickness of the wing is relatively small, then the area on which ice may form is reduced significantly and ice protection on large aircraft is only required for the mid section of the wing and the area near the engine(s). As a result the mass and power requirements of the ice protection system are reduced significantly.

6.3.5 Additional Pneumatic Functions

Pneumatic power in the form of bleed air can be and is used for a number of actuation functions. Few modern aircraft use pneumatic actuation due to the high volume, mass of the system and bleed air required. However, the slats on the Boeing 747 can be pneumatically actuated. One particular system that is often powered pneumatically, is the thrust reversers system. This is partly due to the proximity of the actuators to the engines and also the high temperatures involved.

In addition to its use for actuation, pneumatic power is also used with a turbine for the provision of shaft power. Bleed air can be used to drive a turbo-compressor and is also used on Boeing aircraft to drive hydraulic pumps during peak loads periods.

6.4 Hydraulic Power Systems

On both civil and military aircraft, hydraulic power is manly used for actuation functions. The actuation functions required depend on the type of aircraft and can vary greatly. Figure 6.12 shows a schematic diagram of the hydraulic system of the Airbus A340-500. In this case, hydraulic power is used for all flight controls, i.e. both primary and secondary, utilities such as landing gear and brake operations as well as door operation.

Large civil transport aircraft would normally incorporate three separate hydraulic systems. The probability of failure for hydraulic system is approximately $10—^{4}$ per hour of flight. To ensure that failure probability is less than 10^{-9}, as required for flight

critical systems, three individual channels are required. In most cases the main source of power is a pump driven by the engine.

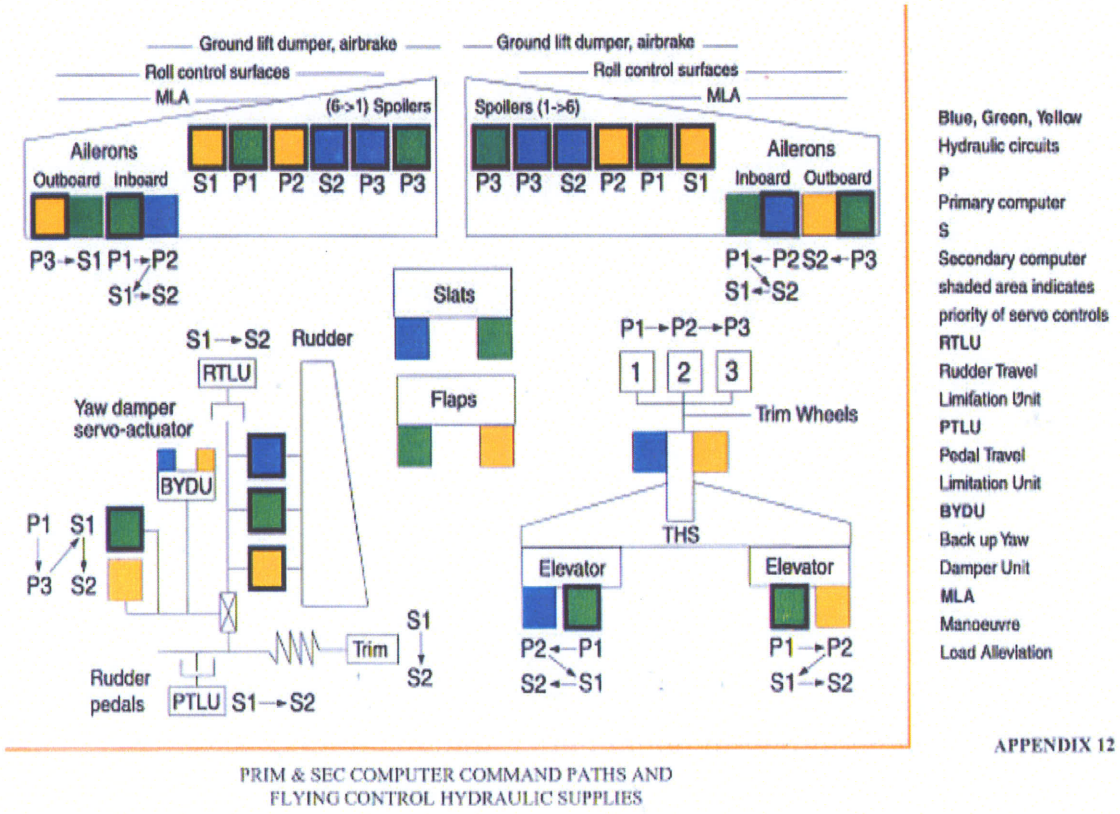


Figure 6.12: A340-500 Hydraulic Flight Control System

Electric driven pumps are often used for redundancy and to provide additional power during peak power demands. RAT and bleed air driven pumps are also incorporated. Finally, Power Transfer Units, PTU, are often incorporated to transfer power from one channel to another without transferring fluid. These units consist of a pump/motor combination that can be driven in both directions.

As the requirements for fluid power have grown, the pressures used in the systems have increased accordingly to reduce the mass, size and power penalties. Present civil transport aircraft utilise systems that operate at 3000 psi with 4000 and 5000 psi used on military aircraft. However, the Airbus A380 is known to incorporate two hydraulic channels that operate at 5000 psi. Advances in materials, hydraulic fluids and seal technology have allowed this increase in operating pressure.

The majority of present actuation systems make use of the power supplied by the three hydraulic systems either to drive linear actuators or motors. The input signal may be either in the form of mechanical signals through a series of linkages or electrical signals via a flight control computer, depending on the application and type of aircraft. The latter configuration is known as fly by wire. Different design and system philosophies are possible however the main principle of operation remains the same and is shown in figure 6.13.

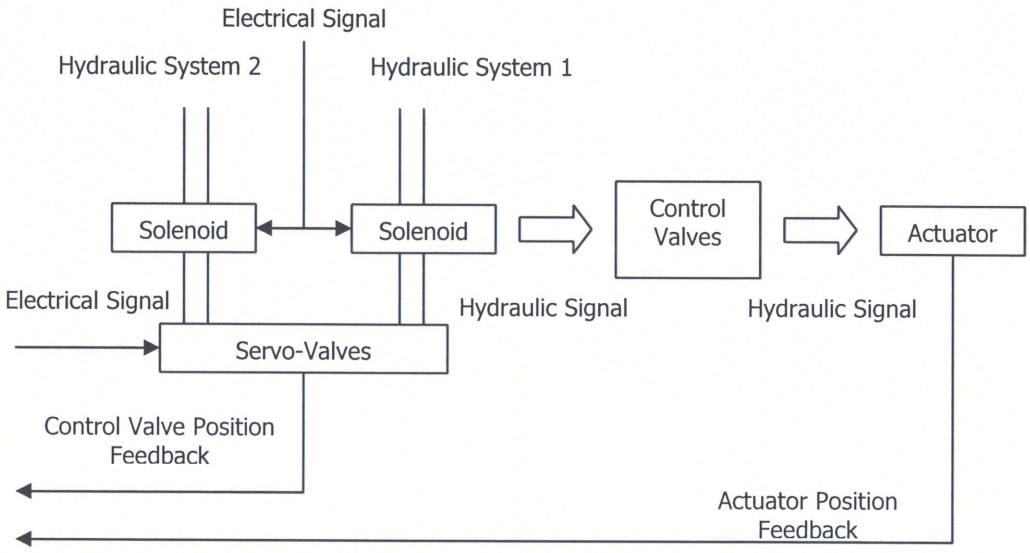


Figure 6.13: Simplified Diagram of Electrically Signalled Actuator[79]

Electrical signals are used to operate solenoid valves that supply hydraulic power to the actuator. Control demands from the flight control computer are also fed to the servo-valves. The servo-valves control the position of the first stage valves that are mechanically summed before applying demands to control valves. The control valves adjust the position of the actuator. Linear Variable Differential Transformers (LVDT) are incorporated to measure the position of the first stage control valves and the hydraulic actuator providing feedback to the flight control computer and closing the control loop.

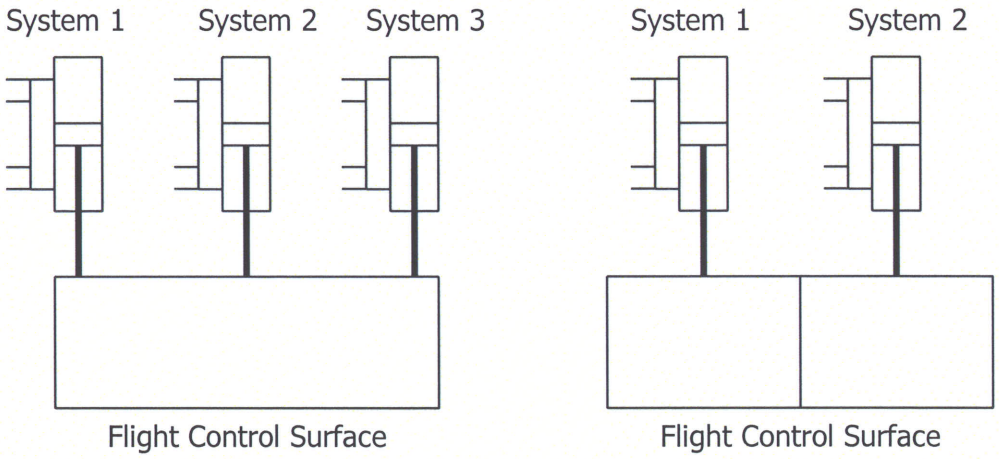


Figure 6.14: Civil Aircraft Control Surface Actuation [79]

Because of their simplicity and design, hydraulic linear actuators (often referred to as hydraulic rams) are suitable for fail-safe designs and hence actuation of primary flying controls. To provide the required level of redundancy hydraulic rams can be connected either in parallel or in series. The former approach is used in the case of civil aircraft and incorporates the use of two or three separate hydraulic rams for the actuation of a flight control surface. To increase the level of redundancy provided, control surfaces can also be divided into two or more sections. Each actuator is usually powered by a separate hydraulic system, Figure 6.14. The latter approach makes use of what is known as tandem actuators. This way two rams powered from two separate hydraulic systems can be combined to a single unit for powering flight control surfaces. Since most military aircraft incorporate two hydraulic channels this design is most suited to the actuation of such aircraft. The configuration of a tandem actuator is shown in Figure 6.15.

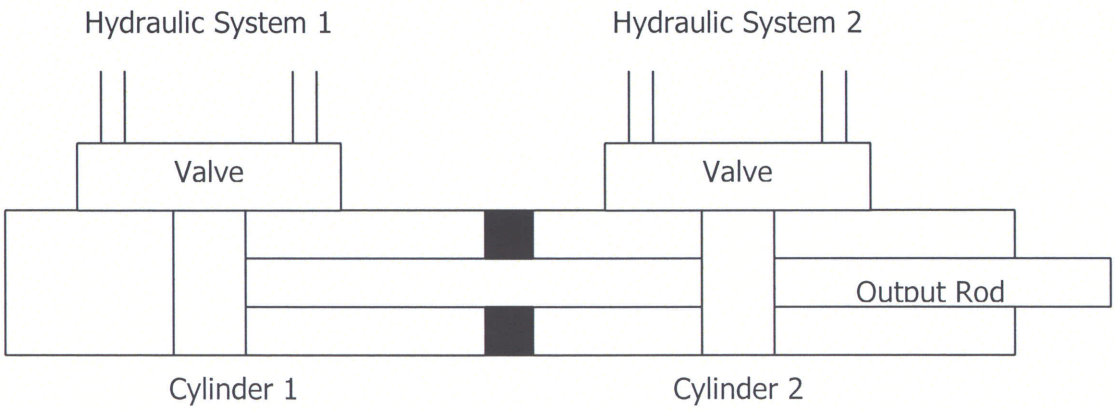


Figure 6.15: Schematic of Tandem Actuator[79]

The hydraulic rams are designed in such as way that a single actuator is capable of meeting the thrust requirements of the control surface. As a result, in the case of a structural failure, leakage and/or seal failure in one actuator or chamber the remaining actuator(s) can still function. To cater for actuator locking, i.e valve failure, a pressure relief valve is usually implemented to release the pressure within the cylinder and allow movement of the surface. This is a significant advantage of hydraulic rams that makes them suitable for use with primary flight control surfaces and other flight critical systems.

6.5 Electrical Power Systems

On most aircraft generators driven from the engine, APU, ram air turbine and in some cases hydraulic motors supply electrical power. On small aircraft the generator would normally supply DC power at 28 volts. This is done to simplify the electrical system and allow direct supply to equipment powered by batteries. On larger aircraft, the need to supply higher power while limit the mass and the losses of the electrical system requires higher voltages. The present standard is 115 V line to neutral (or 200V line to line) 400 Hz, 3 phase AC power distribution. However, limited 28 V DC distribution is also available through transformers rectifiers and batteries.

Most forms of generators will produce variable voltage and frequency when driven by a variable speed source. By varying the exciting field accordingly, voltage can be kept constant but frequency will still vary. While some loads can operate with varying frequency, the design of many systems is simplified and improved if both frequency and voltage are kept constant. Since the engine rotational speed will vary with altitude, flying Mach number and power setting a mean is required to control the speed of the generator. The method normally used to achieve constant voltage and frequency is by incorporating a constant speed drive between the engine gearbox and the generator.

Most large modern aircraft use a combined brushless generator and constant speed drive known as an Intergraded Drive Generator (IDG). Such an arrangement reduces the combined size and mass of separate units and allows common cooling systems.

The circuit of a typical brushless generator is shown in Figure 6.16. The generator consists of three main components:

- A permanent magnet generator
- The exciter generator

- The main generator

The permanent magnet is connected to the rotor and is used to induce an alternating current into the stationary permanent magnet generator armature. The generator control unit rectifies the AC armature current and sends a DC voltage to the exciter field winding. The exciter field induces an alternating current to the exciter armature. The latter is connected to a rotating rectifier that changes the alternating current to direct current and sends it to the main generator field. The main field induces an AC voltage into the main generator armature.

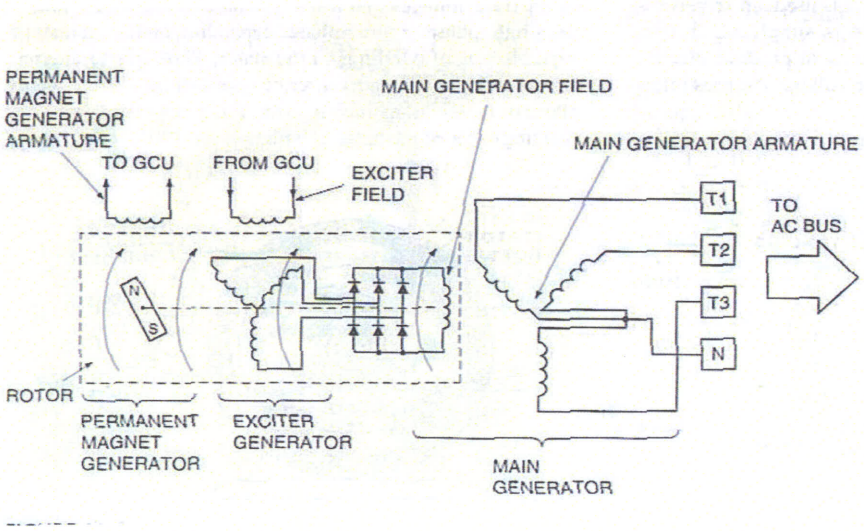


Figure 6.16: Typical Brushless Generator[134]

The main generator armature is a three phase 'Y' or star connection. The generator produces three sine waves each one positioned 120° out of phase with others. In such configuration the phase voltage is 115 V, while the line voltage is 200 V. One terminal of each separate stator winding is connected to the ground and the other terminal is the main output terminal. For aircraft circuits requiring 115 V single phase power, the circuit is connected between one main phase and the ground. For three phase power circuits such as those for motors, all three phases are connected to the motor.

On aircraft with AC primary generation there will be some equipment requiring DC supply. This will be supplied by Transformer Rectifier Units. In these units AC power is fed to a transformer to lower voltage and then to a rectifier bridge to produce a DC supply. Because of their simplicity and high reliability permanent magnet alternators are often used even in the case where DC power is distributed. For this reason all the

output power from the alternator is fed into transformer rectifies to produce a DC supply.

Electrical users are divided into three types:

- Vital – required even after a crash
- Essential – required to ensure safe flight
- Non-Essential – can be shed in an emergency

By grouping the equipment this way, the distribution system can be simplified and the best choice of power source from those available can be made. On large aircraft with multiple power sources and generators, two main philosophies for electric power distribution exist. Loads can be supplied by channels connected to two or more generators, which share the load. This is known as paralleling. The other option is to supply channels with only one generator at any time. Figures 6.17 and 6.18 show schematics of the electrical power system of an Airbus A300 and A340 that follow these two different philosophies.

Paralleling of power sources has the advantage that loss of a generator will not interrupt power to the loads. However, to allow paralleling, careful control of the generators is required and it is possible that faults in one generator could affect others.

On aircraft with DC primary power supply there may be equipment that require AC supply. In addition, on aircraft with AC primary supply, batteries must provide the power required in the case of an engine or generator failure. For these reasons, DC power is converted to AC power through static invertors, which use solid-state electronics to produce an AC output from DC input.

6.6 Inefficiencies and Limitations of Conventional Secondary Power Systems

Conventional secondary power systems evolved during the years that fuel was relatively inexpensive and despite their simplicity and advantages they offer, their design is not always optimised for minimum fuel consumption. Present forms of secondary systems have two main disadvantages. Firstly, bleeding air from the engines is highly inefficient and has detrimental effects on the performance of the powerplant. Secondly, existence of three different forms of secondary power reduces the utilisation of each system and increases the mass penalties. Although pneumatic and electric

power requirements remain relatively constant throughout the mission profile, hydraulic loads vary greatly with peaks occurring during the take-off and landing phases. The hydraulic system will be sized for the peak loads and under most operating conditions it will operate at significantly lower power requirements.

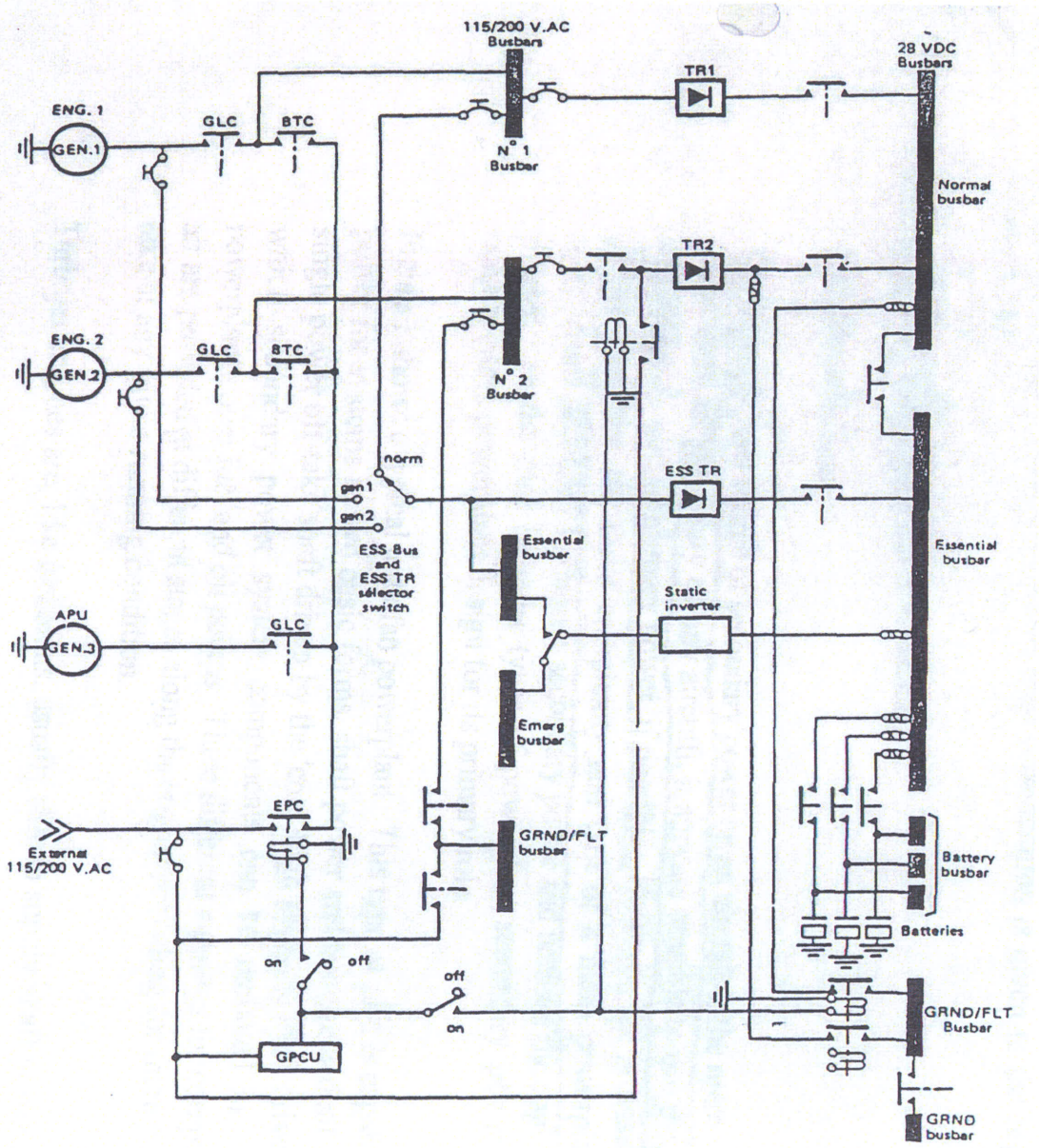


Figure 6.17: A300-600 Electrical System[67]

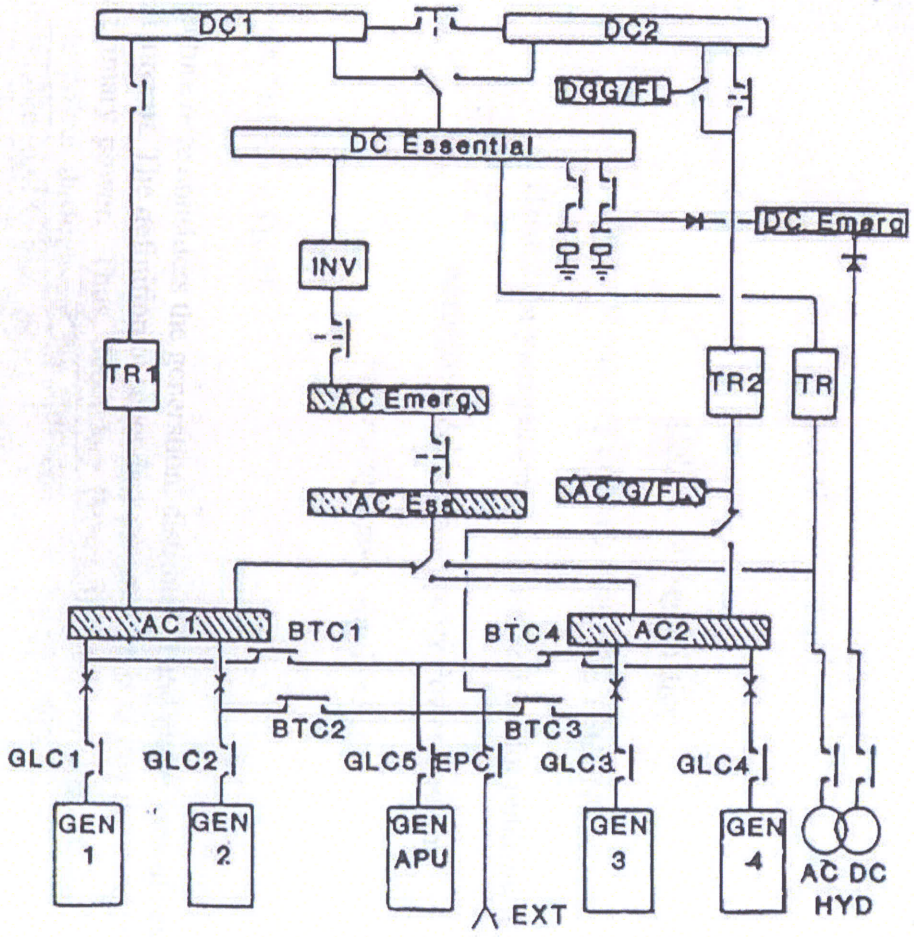


Figure 6.18: A340-500 Electrical System[67]

6.6.1 Bleed Air Inefficiencies and Considerations

Present transport aircraft utilise bleed air from the engine to power and supply the anti-icing and air-conditioning systems. Even if a change in the form of secondary power is to be made, the environmental control system will continue to supply the cabin with pressurised air and therefore it is worth considering the possible alternatives for the need to supply this air.

Due to its size and the design effort applied, engine compressor is indeed the most efficient source of air for the supply of the ECS. Even if a separate compressor is to be used, considering the low flow rates required, the efficiency would not be near that of the main engine compressor. Therefore in theory it appears that bleeding air from the

engine compressor is the most efficient option even if the engine has to be oversized. However in practise large inefficiencies are introduced.

The required airflows and pressures for the operation of the ECS vary with speed and altitude. In most cases, the cooling requirement, and therefore mass flow rates, on a hot day is the major power demand on the ground, while the power for pressurisation is the major power demand at higher altitudes. Although altering the operating speed, altitude or thrust setting of the engine would change the pressure and flow rate through a given compressor stage, the airflow and pressure requirements of the ECS do not necessarily correspond to those delivered by a given compressor stage. Therefore ECS requirements do not constitute a constant proportion of the total flow through the engine. Ideally, the ability to bleed air at any point along the compressor to obtain the necessary pressure would be therefore desired [68, 69].

Since, provision of bleed ports tent to increase the mass and complexity of the engine, the number of ports is usually limited to three. One of these ports is for fan air, which provides insufficient pressure for the ECS and therefore is used for cooling. The other two ports are usually positioned at the intermediate and final stage of the HP compressor.

While the HP port is used during descent and ground idle conditions, the IP port is positioned to provide sufficient pressure for the ECS under all cruise and climb conditions. However, this means that for all but their design point, both the IP and HP port supply air at higher pressures and temperatures than necessary.

In all cases the bleed air is to hot and therefore it must be cooled before entering the aircraft. For this purpose a pre-cooler positioned in the engine pylon is used. The fan normally supplies the cold air. The considerations discussed above lead to a series of performance penalties. In most operating conditions, the air is over-compressed for no useful purpose and the need to cool it demands further off-takes and implies mass and drag penalties. Secondly, the pressure losses incurred by the precooler may lead to a further increase in pressure required.

Moreover, since the IP port is also used by the anti-icing system, assuming the use of thermal anti-icing, which is usually the case, it is the temperature rather than the pressure, which determines the position of the bleed port. Therefore, the position of the IP port is usually determined by the anti-icing requirements leading to further inefficiencies and losses. Considering now that a particular engine may be intended for

use with different aircraft the temperature of the air supplied must satisfy a wide range of requirements.

The penalties imposed to the propulsion system are therefore described in References 43 and 48 to be the following:

- Fuel is burnt to over-compress the bleed air for no useful purpose
- The thrust loss leads to a larger engine to meet the installed thrust requirement, which in turn burns more fuel for a given installed thrust
- The use of ram air or "fan tapping" air, necessary to reduce the temperature to prevent hot air from reaching the airframe, increases the drag penalty and leads to a more inefficient system
- The engine becomes larger, heavier and more expensive than an engine that supplies no bleed

The ways to reduce or remove the penalties for supplying bleed air may include:

(a) Reduced bleed flow rates

(b) Reduced bleed pressures

(c) Elimination of bleed off-takes

While bleed flow rates can be reduced by using alternative cooling methods, reduced pressures can be obtained by a combination of different cooling systems and/or altering the number of bleed ports. Finally, elimination of bleed off-takes implies the use of a separate compressor. These options will be examined in turn.

6.6.2 Methods for Reducing Bleed Air Flow Rates

If the bleed air extracted from the engine was being used by a system requiring only its energy content, then the airflow could be reduced to compensate for the higher pressure and vice-versa. However, current ECS require the flow rate as well as the pressure of air being bled and therefore excess amount of power is being extracted from the engine. The idea behind reduced airflows is to improve efficiency by using the energy content of the bleed air to minimise flow rates.

FAR and JAR require that in normal operation, the airflow into the cabin should be no less than 0.5 lb/min per passenger. Should one source fail the minimum airflow reduces to 0.4lb/min. However, since this is the minimum airflow defined by the airworthiness requirements, it may not be sufficient to hold cabin temperature and pressure at the levels required. In practise the airflow into the cabin is usually around 1lb/min per passenger. During cruise conditions the pressure delivered by the ECS is high enough and therefore the temperature defines the mass flow required. Obviously the lower the temperature delivered into the cabin the lower the mass flow required.

More recently the ability to produce sub-zero temperatures at turbine outlet, has allowed a considerable reduction in flow rates required to maintain cabin temperature. Use of this system may allow the reduction in mass flows to those defined by the airworthiness requirements. However, this would require an increase in the air pressure delivered from the engine.

Further reduction in flow rates required, could be accomplished by a more efficient use of the energy content of bleed air. One such a system involves air being bled from the engine and then expanded through a turbine to drive an independent compressor taking ram or fan air. Air can be bled either across the compressor or the turbine, preferably the HP and LP turbines. The former system, i.e. bleed air taken from the compressor, has been used on Boeing 707.

A variation of the so-called turbo-compressor system is the flow multiplier, which mixes the compressor and turbine exit flows reducing even more the amount of air compressed and increasing the energy transferred through the system. Although losses can be minimised, control of the system becomes more complex.

An alternative method is the ejector mixer, which provides a way of producing the same results as with the flow multiplier but with the absent of the turbo-machinery. This type of device operates as a jet pump, where high-pressure air is expanded through a nozzle to produce a pressure differential and increase airflow. Although, the system has been used successfully on the Lockheed L1011 aircraft, large mixing losses limit its effectiveness.

6.6.3 Increase Number of Bleed Ports

Rather than minimising the bleed flow rate, performance penalties can be limited by reducing the pressure of the off-take. As mentioned before, for all but their design point, both the IP and HP ports supply air at higher pressure and temperatures than necessary. A way around this problem is to simply increase the number of ports so that air can be bled at closer to the required conditions. For example, as discussed in Chapter 2, the design point for a HBPR is likely to be at the top of the cruise. Considering that the position of the IP port is selected to satisfy cruise conditions, the air delivered at the top of climb will have higher pressure than necessary. This would lead to an unnecessary oversizing of the engine in question. More over, increasing the number of ports would allow the anti-icing system to be supplied with air at appropriate temperatures without compromising the performance of the ECS. Although the idea seems promising and was to be used with the proposed GE36 UDF production engine, great inefficiencies and difficulties must be resolved. Setting aside the weight, space and complexity penalties involved, matching the engine components would be more difficult in the case of increased bleed ports.

If no bleeds are required the compressor can be sized and matched to the turbine. This is also true even if a fixed proportion of bleed air is to be taken from a given stage, where the compressor can be oversized. However, if the proportion of bleed air and the position from which it is taken varies, then it is difficult to size and match the engine according to the off-take [48, 49].

An alternative type of cooling system, which is suitable to low pressures, is the refrigerant or vapour cycle system. A major difference between this and the air-cycle system is that it requires much lower supply pressures but it also requires an independent power source to drive the cycle compressor. Despite the additional power requirements, refrigerant cycles can be more efficient and they have been used on aircraft such VC-10. However these systems tent to be heavier, more complex, and more expensive. In addition, given the past experience they present problems in terms of reliability and maintainability. Although previous studies marked their increased efficiency and low power requirements, there are main concerns with the problems outlined above. However reference 49, suggests that further work devoted to the subject of cooling systems, and the refrigerant cycle in particular, would overcome the current difficulties and may indeed lead to a more efficient system.

6.5 Elimination of Engine Bleed Air

6.5.1 Air Supply Compressor Options

If bleed air is not available for ECS supply then a separate compressor must be provided. All previous studies [3, 4, 5, 58, 59, 60] considering elimination of engine bleed air concluded that the use of centrifugal compressors is the best option especially in the case of large transport aircraft.

Axial compressors have been discarded on the basis that they present low efficiencies at low flow rates. In addition the relative low pressure rise per stage, would require a multiple stage device that would increase weight.

Screw type compressors, although they posses a number of advantages, such as suited to low flow rates, no need for surge control and compression been independent of speed, they have disadvantages which include mass, size, noise and limited range.

For the airflows and pressures required, centrifugal compressors found to be more efficient and therefore the most suitable option. Despite their advantages centrifugal compressors do present some problems particularly if high efficiency is to be maintained over the range of operation.

6.5.2 Compressor Drive Options

Whatever type of compressor is specified, it will require a source of power to drive it. There are a number of means to provide the required power and these will be discussed in turn.

a) Direct Shaft Drive: To minimise inefficiencies imposed by conversions of power type between the source and the user, the ECS compressor can be driven by a power off-take shaft from the main engine. Although such a system presents high power transmission efficiencies, control of compressor speed and therefore pressure ratio is hard to achieve. Moreover the location of the compressor is restricted by the mass, and length of the mechanical link.

b) Hydraulic Drive: Application of this system removes some of the constraints on compressor location and also offers a possibility of speed control. However, weight, maintenance and efficiency disadvantages are present.

c) Electric Drive: Use of an electric drive, including a generator mounted on the engine and a motor driving the compressor, offers location flexibility and speed control without the disadvantages of the hydraulic system in terms of weight and overall efficiency. Moreover maintainability issuers favour the application of such a drive compared to a hydraulic drive.

d) Bleed Drive: As described above the compressor can be driven by extracting air from the engine and expanding it through a turbine. However since this paragraph is dealing with possible methods of eliminating bleed air, rather than minimising it, no further investigation of this type will be included.

Of the remaining three drive types, only the direct and electrical drives will be considered in a more detailed level. This is on the basis that the former has the highest efficiency, while the latter allows for installation flexibility and reduced weight.

Considering the direct drive firsts, although the compressor can be driven through a shaft and a fixed ratio gearbox by the engine, the compressor must be sized for the worst case. Since the requirements and driving speed will vary, it is estimated that considerable greater flow rates and pressures will be delivered than necessary. The inclusion of variable geometry in the form of inlet guide vanes (IGV) can control output to some extent but a more effective mean of control is through speed. In the past a two or even three ratio gearbox has been used. However according to reference 49, a centrifugal compressor with a two ratio drive and IGVs may run at barely more than 30% isentropic efficiency in some operating cases whilst design case efficiencies of 80% would be possible. Installation constrains is another disadvantage of the direct shaft drive. The shaft should be as sort as possible and space must be available for the compressor. The cowl can be extended to accommodate the compressor but this action will introduce mass and drag penalties. Locating the compressor further from the engine would require a longer and heavier mechanical link and will introduce pressure losses across the engine's flow ducts. In the case of HBPR engines the penalties would be more severe.

The advantages of electric drive, on compressor location and control are achieved at the cost of inefficiencies introduced into the drive system due to the conversions of power. Driving a generator from a variable speed source, i.e. engine shaft, produces power whose frequency and voltage is directly proportional to the driving speed. For the types of motor suitable to drive a compressor these variations translate to variation in driving speed and therefore compressor output. Although most of present aircraft

utilise a Constant Speed Drive (CSD) unit to provide constant frequency from variable speed, efficiency limitations and weight considerations are introduced. Therefore if the benefits of precise control are to be realised through the use of an electric drive, some form of frequency control must be introduced between the generator and the motor. Eventually the system will be in the form of Variable Speed/Constant Frequency (VS/CF). According to references 58, 59, 60, this frequency control can in fact be relative crude since a well conditioned sine wave power is not necessary. This in turn allows the use of simple circuits. Frequency and therefore speed control can be achieved with the use of either a cyclo-converter or a DC link. The cycloconverter system use Thyristors to chop-up power supplied into a form approximating 400Hz then filter it to produce a smooth sine wave. A DC-Link system simply rectifies the variable frequency produced the use power transistors to reconstruct 400Hz AC.

Using speed control available through converters, the outlet conditions from the compressor can be reasonably controlled, though some form of variable geometry is still desirable in keeping compressor's efficiency close to optimum. However additional losses are introduced due to the efficiency of the generator, motor and the presence of the converters. In the open literature it was found that a large variations in the overall train efficiency is present. Older references, such as 52 &54, claim that the overall efficiency of the drive system will never exceed 60%-65%. However with the advances in electronics more recent studies claim efficiencies around 80%-90% [48,49, 59, 60]. Assuming a value of 85% the overall efficiency of the system falls to 68%, with an 80% efficient compressor. Although this is considerably less than the design efficiency of a direct shaft, off-design efficiencies will be considerably higher for the electrical system giving values not less than 60%.

In addition to a more efficient speed control, the electric drive would enable the location of the compressor to be away from the engine and therefore the nacelle could be sized for maximum aerodynamic efficiency. Although the generator and its controller will require extra mass, it is anticipated that the elimination of the AGB and the mechanical link will offset these weight penalties. Furthermore the pressure losses incurred from passing the electrical cables through the fan stream will be considerably lower than those introduced from a mechanical link.

It is for these reasons that previous studies concerned with the elimination of bleed air favoured the application of the electrical drive system. As far as the rest of the ECS is concerned, either an air-cycle or a hybrid air/vapour cycle can be used. Previous studies[49, 50, 59, 60] indicated that at high altitude conditions, like top of climb and cruise, a pressure ratio around 4 is needed to ensure adequate cabin pressure. At these

conditions the temperature of the air delivered is low enough to be cooled by a simple heat exchanger utilising ram or fan air. However at sea level static conditions and on a hot day either an air cycle or a hybrid system is required. This will also increase the pressure ratio of the compressor and therefore the power required to drive it.

6.5.3 Design Considerations for All Electric Environmental Control System

Design of an environmental control system suitable for the all or more electric concepts is a complex task requiring a lot of unavailable information such as delivery pressures and temperatures in addition to mass flow rates at different operating conditions. Design choices with regard to the type of compressors and driving options would have an impact on the final design and its characteristics including mass and power requirements. These will also be affected by the operating requirements of the environmental control system. For these reasons only a qualitative analysis is presented and some general conclusions are drawn. The analysis is based on the requirements, findings and methods considered by previous studies on the subject. The complexity of the task becomes evident considering that findings from previous studies conflict with each other and different design options have been proposed and adopted. However, for the purpose of this study it is possible to perform a general qualitative analysis and cover some of the uncertainties with a sensitivity analysis.

The main difficulty in performing a detail design of an electric environmental control system is that with the information available in the open literature it is not possible to set design targets and calculate the heat input or removal required during various operating conditions. As a result it is not possible to define pressure, temperature and mass flow requirements for the operation of the system. This in turn does not allow a detail design of the compressor and its driving system. Based on the findings of previous studies, two main points of interest are identified. At high altitude operations the main requirement is that of cabin pressurisation and it is possible to incorporate a design with low-pressure ratio. The pressure of the compressor will be defined by the pressure losses of the system and the pressure requirements within the cabin. Cooling can be achieved with the use of ram air heat exchangers and the air cycle pack can be by-passed, as in the case of the Lockheed study. This approach was also favoured by Boeing in their study.

Therefore, at cruise conditions and assuming 80% compressor intake recovery, the pressure ratio of the compressor can be set at 4:1. Previous studies performed during the eighties at the Cranfield College of Aeronautics used a similar compressor pressure ratio at cruise conditions. Assuming 80% compressor efficiency the total power

extracted from the engines to drive the separate compressors varies from 440KW for the A340-500 class aircraft to 354 KW for the A330-200 class aircraft and 777KW for the A380 class aircraft.

The pressure ratio and power requirements for the separate compressor can be reduced if the compressor receives air from the bypass duct of the engine. Such an approach will also eliminate the pressure losses and drag incurred by the intake of the compressor. In this case the power requirements can reduce to half those required for a compressor receiving free stream air. However, such an action would require extraction of bleed air from the engine bypass and would incur additional engine performance penalties. In order to minimise pressure losses the compressor should be placed close to the engine. Normally, the air cycle packs would be placed at the wing box, which is located at the point where the wing intercepts the fuselage. Although, the size and power requirements of the compressor would reduce, thus requiring smaller generators, motors and control units, the ducts from the engines to the air-conditioning packs will be retained. Finally, if the compressors are to be installed close to the engines use of direct driven compressors may increase the overall mass savings.

The second point of interest is the performance of the system on the ground and during hot days operations. Findings from the Boeing study indicate that these conditions would require higher compressor pressure ratio due to the fact that cooling must be provided from the air cycle packs. For this reason the pressure ratio of the compressor was set at 5:1. However, the study performed by the College of Aeronautics indicates that the highest pressure ratio will be required at high altitude low speed operations where the delivery pressure is necessary to maintain the required pressure within the cabin. The same is true in the findings presented by Lockheed[5]. However it is noted that Lockheed adopted a vapour cycle and therefore the cooling requirements on the ground would not require air at high pressure.

Assuming the operation of a vapour cycle the pressure ratio of the compressor can be close to 2:1. In the case of a bootstrap cycle and according to Boeing's study, the pressure ratio of the compressor must be set at 5:1. Finally the study performed by the College of Aeronautics suggests that the pressure ratio of the compressor must be set at 2.5:1 and cooling can be achieved through a higher mass flow rate. The power requirements for the three baseline aircraft at different operating conditions and based on the findings of previous studies are shown in Table 6.1.

Operating Conditions	A330-200	A340-500	A380
Cruise			
Free Stream Air	2*177 KW	4*109 KW	4*194 KW
Fan Air	2*92 KW	4*57 KW	4*101 KW
Vapour Cycle	2*156 KW	4*97 KW	4*170 KW
Ground Hot Day			
Free Stream PR 5:1	2*250 KW	4*155 KW	4*275 KW
Fan Air – Total PR 5:1	2*184 KW	4*114 KW	4*200 KW
Free Stream PR 2.5:1	2*151 KW	4*105 KW	4*187 KW
Fan Air – Total PR 2.5:1	2*70 KW	4*43 KW	4*77 KW
Vapour Cycle	2*90 KW	4*80 KW	4*100 KW

Table 6.1: Environmental Control System Power Requirements

In the case of an air cycle, the design point and power required to size the generators and motors depends on the overall configuration of the system and the compressor pressure ratio required on the ground. If a pressure ratio close to 5:1 is required then this condition would be the design point for the motors, controllers and generators. In this case, the high power requirements lead to a mass penalty relative to the conventional system, despite the removal of several components such as ducts, valves, pre-coolers, etc. Almost half of the mass added is that of the controllers required to control the generators during engine starting and the motors when the compressors are running. Use of fan air instead of free stream air would allow a small reduction in the power required and therefore the mass of these components would reduce. However, if fan air is to be used then not all the ducting in the pylons, nacelles and wings will be replaced and the reduced mass of electrical equipment would be counteracted by the weight of the components that will be retained. This is shown in Table 6.2.

Electrically Driven System / Components	A330-200	A340-500	A380
Free Stream PR 5:1	2*250 KW	4*155 KW	4*275 KW
Added Generators	150 Kg	186 Kg	330 Kg
Motors	150 Kg	186 Kg	330 Kg
Controllers	340 Kg	422 Kg	748 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, air starting, etc)	-603 kg	-783 Kg	-1405 Kg
Total	**+97 kg**	**+114 Kg**	**+125 Kg**
Fan Air PR 5:1	2*184 KW	4*114 KW	4*200 KW
Added Generators	110 Kg	137 Kg	240 Kg
Motors	110 Kg	137 Kg	240 Kg
Controllers	250 Kg	310 Kg	544 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, air starting, etc)	-362 Kg	-422 Kg	-767 Kg
Total	**+168 Kg**	**+256 Kg**	**+379 Kg**
Free Stream PR 2.5:1	2*177 KW	4*109 KW	4*194 KW
Added Generators	106 Kg	131 Kg	233 Kg
Motors	106 Kg	131 Kg	233 Kg
Controllers	241 Kg	297 Kg	527 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, air starting, etc)	-603 Kg	-783 Kg	-1405 Kg
Total	**-90 Kg**	**-120 Kg**	**-290 Kg**
Fan Air PR 2.5:1	2*92 KW	4*57 KW	4*101 KW
Added Generators	55.2 Kg	68 Kg	121 Kg
Motors	55.2 Kg	68 Kg	121 Kg
Controllers	125 Kg	155 Kg	275 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, air starting, etc)	-362 Kg	-422 Kg	-767 Kg
Total	**-66 Kg**	**-27 Kg**	**-128 Kg**
Vapour Cycle	2*156 KW	4*97 KW	4*170 KW
Added Generators	93.6 Kg	116.4 Kg	204 Kg
Motors	93.6 Kg	116.4 Kg	204 Kg
Controllers	212 Kg	264 Kg	462.4 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, air starting, etc)	-603 Kg	-783 Kg	-1405 Kg
Total *(Mass of Vapour System not Included)*	**-144 Kg**	**-182 Kg**	**-412.6 Kg**

Table 6.2: Environmental Control System Mass Changes for Electrically Driven Compressors

It is noted that in all cases use of fan air and electrically driven compressors led to higher mass penalties. If the compressors are driven by the engine, through a power shaft and a gearbox, then use of fan air would lead to an overall mass reduction, both relative to the conventional system and the electrically driven compressors. This is shown in Table 6.3. However, extraction of bleed air from the engine bypass will affect the performance of the engine and although shaft power extraction will reduce, the net effect is a slight increase in sfc.

Mechanically Driven System / Components	A330-200	A340-500	A380
Fan Air PR 5:1			
Added Gearbox	50 Kg	61 Kg	107 Kg
Shaft	80 Kg	100 Kg	174 Kg
Controller	12 Kg	15 Kg	27 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, etc)	-237 Kg	-293 Kg	-521 Kg
Total	**-35 Kg**	**-14 Kg**	**-91 Kg**
Fan Air 2.5:1			
Added Gearbox	24 Kg	30 Kg	54 Kg
Shaft	40 Kg	50 Kg	88 Kg
Controller	6 Kg	8 Kg	14 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, etc)	-237 Kg	-293 Kg	-521 Kg
Total	**-106 Kg**	**- 102 Kg**	**-243 Kg**
Free Stream Pr 5:1			
Added Gearbox	67 Kg	83 Kg	147 Kg
Shaft	109 Kg	135 Kg	239 Kg
Controller	19 Kg	20 Kg	36 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, etc)	-237 Kg	-293 Kg	-521 Kg
Total	**+15 Kg**	**+49 Kg**	**+24 Kg**
Free Stream PR 2.5:1			
Added Gearbox	47 Kg	58 Kg	104 Kg
Shaft	77 Kg	95 Kg	169 Kg
Controller	11 Kg	14 Kg	26 Kg
Compressors	60 Kg	104 Kg	122 Kg
Removed (Ducts, precooler, etc)	-237 Kg	-293 Kg	-521 Kg
Total	**-41 Kg**	**-22 Kg**	**-100 Kg**

Table 6.3: Environmental Control System Mass Changes Engine Driven Compressor

The effect is more noticeable in the case of the Trent 700 class engine due to the lower bypass ratio and hence, higher percentage of fan bleed air. The small mass reduction benefit of the engine driven compressors is counteracted by the increased sfc and fuel flow required at the early stages of the cruise phase and the net effect is increased aircraft weight and fuel load.

It is noted, however, that the results shown in Table 6.2 do not include the mass of the APU generators. In the case of electrically driven compressor, the capacity of the APU generators should increase to meet the power requirements of the compressors for ground and hot day conditions. In this case, the APU can still be used to supply power to the environmental control system while on the ground and with the engines inoperative. The design of the APU will change and the bleed air currently supplied will be eliminated. The power output from the generators will also be used to start the engines from the APU.

In the case of engine driven compressors, the design of the APU will remain the same and therefore bleed air will also be supplied. The air can be used to start the engines and also provide air for the operation of the environmental control system. Although, elimination of bleed air from the APU will lead to a simpler and lighter unit the increase in generator mass would cause additional mass penalties. The problem may be alleviated if the APU generators are sized for starting the engines and the power supplied to the electrically driven compressors would be supplied from the engines. This, however, requires running the engines while on the ground and when the environmental control system is required. With the engines running, some or all of the remaining secondary power could be supplied from the APU. The results for all cases are summarised in Table 6.4

In the case of engine driven compressors the results suggest that in terms of equipment mass the compressors should receive fan air. Considering the bleed air penalties and the overall performance of the aircraft then the case of free stream compressor should be adopted. The choice between electrically driven compressors and engine shaft driven compressors would depend on the delivery pressure required on the ground and the configuration of the electrical system. If a pressure ratio close to 2.5:1 is required then the design point for the motors generators and controllers will be the cruise conditions. In this case, use of electrically driven compressors would lead to mass reductions relative to both the conventional system and the engine driven compressors. If the APU generators need to be sized for the environmental control system then a small mass penalty should be expected.

If, however, a pressure ratio of 5:1 is required, then the engine driven compressor option would lead to smaller mass penalties, especially in the case where the APU generators would be sized to supply the power required for air-conditioning. Based on the derived values presented in Tables 6.1-6.4 engine driven compressors should be adopted for minimising mass penalties. Both mass penalties incurred form electrically driven compressors and mass savings of engine driven compressors increase with aircraft size and number of passengers. However in terms of percentage penalties, relative to the operating empty weight and/or maximum take-off weight, penalties are higher for the A330-200 class aircraft. This is shown in Table 6.5.

Despite the advantages of engine driven compressors in terms of mass and reduced size of generators and motors, the configuration presents a number of disadvantages that may prohibit its use.

Firstly, the operating conditions of the compressor are hard to control especially in off-design conditions. This may compromise the efficiency of the compressor as well as the mass flow and pressure ratio at the design point. These effects may have a negative impact on the size, mass and power requirements of the components, including compressors, drive shaft and gearbox. Previous studies, however, reported that the mass and power requirements could be the same with those of electrically driven compressors provided that correct component characteristics are selected.

Secondly, such a design requires the use of a two or three ratio gearbox. This in turn would affect the structural integrity of the design and may lead to additional problems concerning the reliability and maintenance requirements of the system. The situation becomes worst considering the high rotational speed required for efficient operation.

Finally, due to the existence of a driving shaft the compressors should be located close to the engines. The proximity to the engine and the existence of high temperatures may lead to further reliability and maintainability issues. Although the compressors could be placed within the pylon and the position currently occupied by the precoolers, it is not certain that enough space exists and as a result the structural layout of the pylon, nacelle and engine may have to change. Measures must also be taken to ensure that a compressor, shaft or gearbox failure would be contained and would not affect the integrity and operation of the engine. Such a provision would lead to additional mass penalties.

	A330-200			A340-500			A380-100		
	Elec APU	No APU	Mech	Elec APU	No APU	Mech	Elec APU	No APU	Mech
Free 5:1	247 Kg	97 Kg	15 Kg	300 Kg	114 Kg	49 Kg	455 Kg	125 Kg	24 Kg
Free 2.5:1	16 Kg	-90 Kg	-41 Kg	11 Kg	-120 Kg	-22 Kg	-57 Kg	-290 Kg	-100Kg
Fan 5:1	278 Kg	168 Kg	-35 Kg	393 Kg	256 Kg	-14 Kg	619 Kg	379 Kg	-91 Kg
Fan 2.5:1	-11 Kg	-66 Kg	-106 Kg	+41 Kg	-27 Kg	-102 Kg	-9 Kg	-128 Kg	-243 Kg

Table 6.4: Comparison of Electrically and Engine Driven Compressors

In the case of a vapour cycle, separate compressors would be required to maintain the cabin pressure at the necessary levels. At cruise conditions the pressure ratio of the compressors would depend on the pressure losses of the system and the intake pressure recovery. Typical values of pressure ratio would be around 3.5:1. Since pressurisation is not required while on the ground and assuming that the vapour cycle would not be operated at the cruise section then, power requirements at the cruise conditions would size the motors, generators and controllers. Although the reduced power requirements would require smaller electrical equipment, vapour cycles are known to be heavier than air cycle systems and the net effect would be a significant increase in aircraft mass. In terms of off-takes the power requirements at cruise would be similar to those of an air cycle, as shown in Table 6.1, and therefore engine performance penalties would remain almost the same and the mass penalties of the vapour cycle will predominate. For these reasons the use of a vapour cycle is not recommended.

	A330-200		A340-500		A380-100	
	% OEW	%MTOW	%OEW	%MTOW	% OEW	%MTOW
PR 5.1:1						
Elec APU	+ 0.2%	+0.1%	+0.17%	+0.08%	0.16%	0.08%
Elec No APU	+0.08 %	+0.04 %	+0.07%	+0.003%	0.04%	0.02%
Mech	+0.01 %	+0.004 %	+0.03%	+0.013%	0.008%	0.004%

Table 6.5: Percentage Mass Penalties for Separate Compressors

6.6 Non Bleed Anti-Icing Systems

Several options for replacing bleed air anti-icing have been identified by previous studies. As mentioned above use of bleed air for the operation of the anti-icing system requires the IP bleed port to be positioned at higher stages across the compressor to provide the necessary temperatures. Hence, elimination of anti-icing bleed air would

allow the IP port to be positioned solely for the requirements of the environmental control system.

Two proposals for such systems have been investigated in the past and include Electro Impulse and Microwave anti-icing systems. Since the former has been studied in more detailed level, including extensive experimental work, it seems more suitable. However there are some concerns regarding its effectiveness, electromagnetic interference and fatigue problems [48,61, 62,63].

6.7 Engine Starting Systems

At present starting of large turbofans is accomplished using an air-starter motor mounted on the AGB. The motor is supplied with high-pressure air from a ground source, APU or another already running engine. The high-pressure air uses the same manifold with the ECS to minimise weight penalties. If bleed air is not used for ECS these ducts will not be necessary and therefore the method for starting the engine should be re-examined.

In the case of a direct shaft driven ECS, The compressors would be close to the engines and most ducting from the engines to the cooling units will be retained. Therefore an air starter could be used, only this time the air supplied will be from the ECS compressor, driven from an engine already running, APU or ground source.

If, however, an electric drive is used it is likely that the compressor would be located near the cooling packs to minimise pressure losses and weight. To retain air starting in this case would impose mass penalties and therefore an alternative system is required.

The generator used to drive the compressor can be used as a starting motor controlled by the existing controllers/converters. Once again as with the existing system the same power sources will be available, only this time they will supply electrical power instead of pressurised air. Although no actual demonstrations have been performed, calculations have shown that the generator capacity required per engine to supply ECS and electrical loads is more than sufficient when acting as a generator [58, 59, 60,].

6.8 Electrical Power Actuation

The subject of actuation systems for the more or all electric aircraft is covered in more detail in reference 135. The present work provides a summary and highlights some of the issues involved. For more information Reference 135 should be consulted.

In general there are two main types of electrically powered actuators available:

- Electromechanical
- Electro-hydrostatic

6.8.1 Electromechanical Actuators

Electromechanical actuators convert electrical power to mechanical power through the use of bi-directional electric motors. Mechanical devices, such as gears, are then used to convert mechanical power and produce the required motion. Several types of electromechanical actuators exist and more details are given in Reference. The two main types most commonly considered are the following:

- Geared rotary actuators that produce a power hinge
- A combination of gearing and a screw-jack to produce a linear output motion

Depending on the application and form of motion required either geared rotary or screwjack actuators could be used to replace existing hydraulic systems used for actuation.

6.8.2 Geared Rotary Actuators

Geared rotary actuators incorporate either multi-stage simple epicyclic gearboxes or compound planetary gearboxes to provide the movement, torque and speed required. The actuators can be powered from a hydraulic or an electric motor. Because of their capability to provide a rotating output motion, hydraulically powered geared rotary actuators are already used in a number of civil and military applications. For example, multistage simple epicycle gearboxes are used to power the leading edge slats on the Boeing 777 aircraft. Compound planetary gearboxes, on the other hand, are used in applications such as the trailing edge flaps of the Airbus A330 and A340 aircraft and the leading edge flaps of the Saab Grippen. Additional applications include the leading

edge flaps of the Boeing 747-400 aircraft as well as the payload doors and rudder actuators of the space shuttle.

6.8.3 Screw Jack Actuators

Ball screw and roller screw actuators are devices, which use nut and screw elements to achieve a linear output movement from a rotational input. Either the screw or the nut can be held from rotating with the remaining component being allowed to rotate and also move linearly. Presently, screw drive systems are used in some aircraft to provide actuation for secondary flight controls and utilities systems including high lift systems, i.e. flaps and slat, horizontal stabilisers, engine thrust reverses and nozzle actuation. In these cases the screws are driven in parallel by a motor and through a series of torque tubes and shafts that provide transmission. Moir[78] describes how such a system is used to drive the trailing edge flaps of the BAe 146 aircraft.

6.8.4 Electro-Hydrostatic Actuators

Electro-Hydrostatic actuators utilise conventional hydraulic rams to achieve the output movements required. However, the pressurised hydraulic fluid is supplied by electric motor driven pumps rather than using a central hydraulic circuit with pumps driven from the accessory gearbox. Directional control of the pump can be achieved in two ways, either by changing the rotation of the pump or by using valves. The latter are known as integrated actuator packages. These packages are similar to the electro hydrostatic actuators previously mentioned but the electric motor rotates in one direction only and directional control is achieved through the use of flow control spool valves similar to those used with conventional hydraulic systems. Electro-hydrostatic actuators are simpler in their design and construction relative to electro-mechanical actuators and share a lot of the advantages present on conventional hydraulic rams.

6.8.5 Limitations of Electrically Powered Actuators

Both electromechanical and elector-hydrostatic actuators have a number of disadvantages that limit their applications. Their suitability depends on many factors such as type of load and output movement, type of application, type of aircraft, etc. In theory both forms of actuators can be used for powering secondary flight control systems.

Conventional hydraulic systems have number of significant advantages, which are hard to ignore. In general, hydraulic rams provide a simple and reliable mean of actuation.

In addition, hydraulic actuators tend to be lighter, easier to design, assembly and manufacture while they cost less than alternative actuation systems including electromechanical and electro-hydrostatic actuators. The main advantage of hydraulic actuators, however, is their level of reliability and the inherent fail-safe design. Because of their construction and operation, hydraulic actuators are less likely to jam. Even if they jam, a relief valve can be incorporated to allow movement. As a result multiple actuators can be used to control and/or power a flight control surface thus increasing reliability and redundancy standards. Provided they are designed properly hydraulic actuators can be overloaded with little mass and cost penalties and without suffering structural damage. If they are overloaded, they can stall and return in working condition when the overload is removed. Finally, they can withhold high loads with out requiring any form of energy or power. It is mainly for these reasons that hydraulic actuators are widely used and on the majority of present aircraft are preferred over alternative actuation systems, especially for powering flight control surfaces and other flight critical systems.

Despite their advantages, however, conventional hydraulic systems also present a significant number of disadvantage including, increased maintenance requirements, mass penalties associated with long pipes, hydraulic fluid, valves, tanks, etc. The all-electric concept can offer significant savings if the pipes and other hydraulic components can be eliminated or replaced by electrical components. However, if these savings are to be realised then the whole hydraulic system must be eliminated. This is because electrically powered actuators are known to be heavier and more complex than conventional hydraulic rams.

Electromechanical actuators suffer from jam of the mechanical components. This may include failure or break up of components or/and foreign objects such as dirt, debris or ice entering the system and preventing movement. The means of overcoming a mechanical jam follows one of the following principles:

- A motion summing approach that involves the use of a second linear or rotary actuator on the back of the primary

- Incorporating some form of shear out or clutch device that would disconnect the jammed component from the rest of the system and/or the surface being moved, thus allowing the surface to continue to move in a manner unaffected by the jam

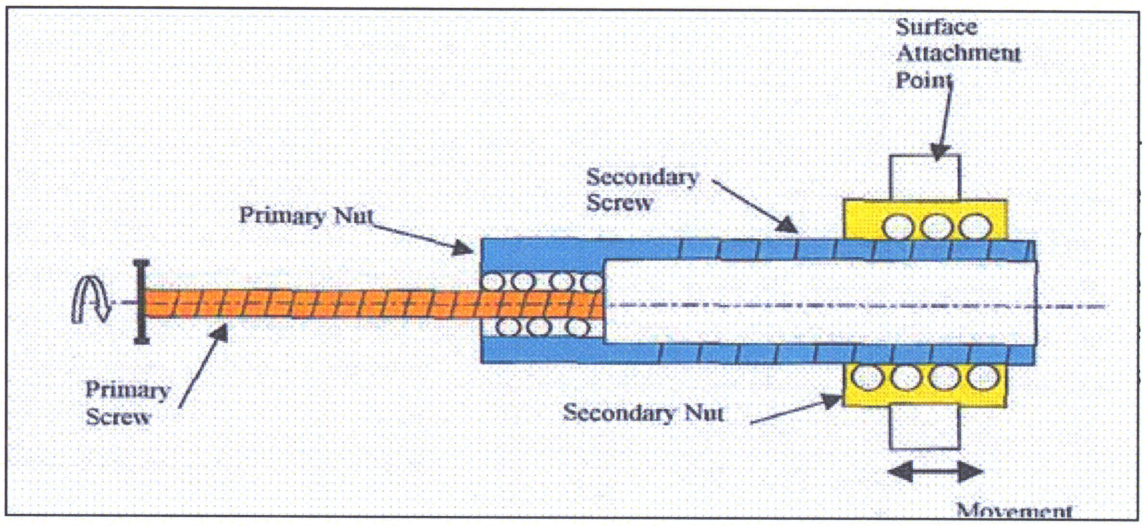


Figure 6.18: Jam Tolerant Electromechanical Architecture [135]

Numerous patents on the design of jam tolerant actuators exist and although several variations are possible the fundamental principle remains the same and is that of a differential gearbox which allows output movements when one of the load paths is jammed. The fact that the jam tolerant devices are differentials implies, that any structural failure along either path leads to uncontrolled surface movement and measure must be incorporated to eliminate such events. Such actions would increase the complexity and mass of the system. Maydew [135] gives more details on the design issues of jam tolerant electromechanical actuators covering both geared rotary and screw–jack actuators. Patents on the subject are also presented and discussed. The work described also includes two patents invented by the author of reference 135 which eliminate some or all of the disadvantages of previous designs.

Despite their application, i.e. high lift device on Boeing 777 aircraft, use of shear out devices is limited and has a number of disadvantages. In the event of an actuator becoming jammed a shear out device would disconnect the actuator or the system jammed and the remaining actuators would control and move the surface. However, the actuators must be designed for the additional load, thus increasing the mass and size of the individual components. Since the power or torque output of the remaining actuators would increase then the design of the shear-out device becomes critical and should be such as to allow the higher torque to be carried without breaking and disconnecting the remaining actuator(s). The latter effect is often called zipper effect. Use of clutches present additional problems since they may stick together if left

unutilised for long periods or may have failure modes that are unfavorable, i.e. fail in a disconnected position.

Use of electro-hydrostatic actuators eliminates the problems and limitations present in electromechanical actuators. The problem of an actuator becoming jammed can be addressed in a similar way to conventional hydraulic actuators and the implementation of a bypass or relief valve. Blocking valves can also be used to avoid uncontrolled movement of the surface in case of a load path failure. Blocking valves can also be used to withhold the surface at a given position without requiring additional power. In recent years a considerable amount of research effort has been directed towards the development of electro-hydrostatic actuators for powering primary and secondary control surfaces. As a result this type of actuators have being selected to provide back up power for the primary flying control system of the Airbus A380 aircraft. In the past there have been concerns over the capability of electro-hydrostatic actuators to achieve the control rates required by primary flight control systems and the large flow rates required. However it can be assumed that since this type of actuators have been already selected to replace some conventional hydraulic systems ways can be found, to alleviate these problems. Use of electro-hydrostatic actuators, however, partly addresses the problems of the existing hydraulic systems and share similar disadvantages. As a result they tend to be more complex, heavier and require additional redundancy levels than equivalent hydraulic rams.

If the full benefits of adopting the all-electric concepts are to be realized then electromechanical actuator designs and architectures must be developed to avoid and/or alleviate the design issues previously discussed. However, existing design philosophies indicate that most if not all of the problems associated with electromechanical actuation systems can be addressed with the final configuration depending on the application and type of aircraft. These issues are considered in the next chapter of this works, which deals with possible system architectures for the all or more electric aircraft.

6.8.6 Electric Power Required to Replace Hydraulic Primary Flight Control System

Elimination of the conventional hydraulic flight control system would affect the size and capacity of the generators. In addition, the loads required would size the electric motors used to power the actuators and control the surfaces. For these reason a method for calculating the hinge moments applied on the control surface was followed.

Although the values derived from the analysis would differ from the actual values required for the aircraft, the method gives an indication of the power levels required for both the motors and the generators. Therefore and in the absence of more detailed calculations it is assumed that the values derived are accurate enough.

The method relies on the calculation of hinge moment coefficients for each control surface. These values are derived using expressions found in ESDU [136]. The method relies on the geometric details of the aerodynamic surfaces and the control surfaces. Geometric details were gathered from publicly available drawings. Further information regarding the thickness to chord ratio of the control surfaces and the airfoil profile were gathered from studies[137-138] which deal with the design of hypothetical aircraft similar to those used in the present study. Hinge moment coefficients were then calculated as a function of surface deflection. The maximum surface deflection was taken from published data on existing aircraft. Figure 6.20 plots the calculated hinge moment coefficient with surface deflection for the rudder of the A340-500 class aircraft.

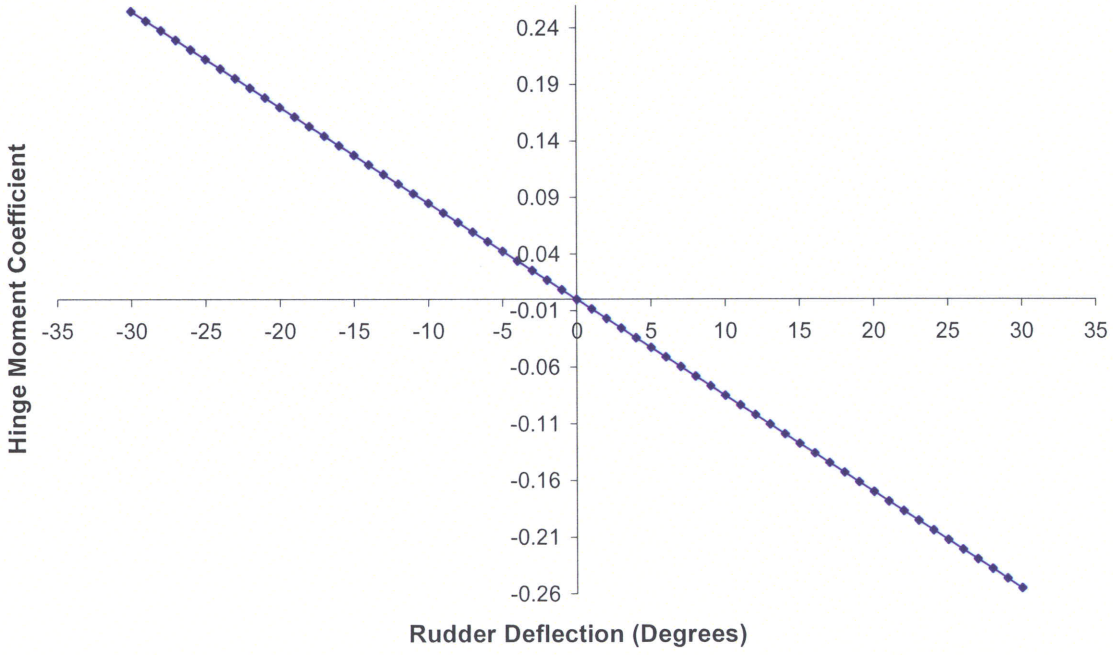


Figure 6.20: Calculated Rudder Hinge Coefficient for A340-500 Class Aircraft

The hinge moment on the flight control surface was then calculated using the following expression:

$$\text{Hinge Moment} = \frac{1}{2}\rho V^2 C_H S\bar{c} \text{ (ft-lb)}$$

Where
P = Air Density
V = Velocity
C_H = Hinge Moment Coefficient
S= Reference Area
C = average surface chord

Hinge moments were calculated for each flight phase. From the above equation it follows that hinge moment would depend on flight conditions and angle of deflection. Also it is noted that by dividing the calculated hinge moment with the moment arm the load on the actuator can be found. To calculate the maximum deflection at each flight condition it was assumed that the maximum load on the actuator would occur at take off conditions and a speed of around 84 m/s. The resulting hinge moments as a function of equivalent air speed for the rudder of the A340-500 class aircraft is shown in Figure 6.21.

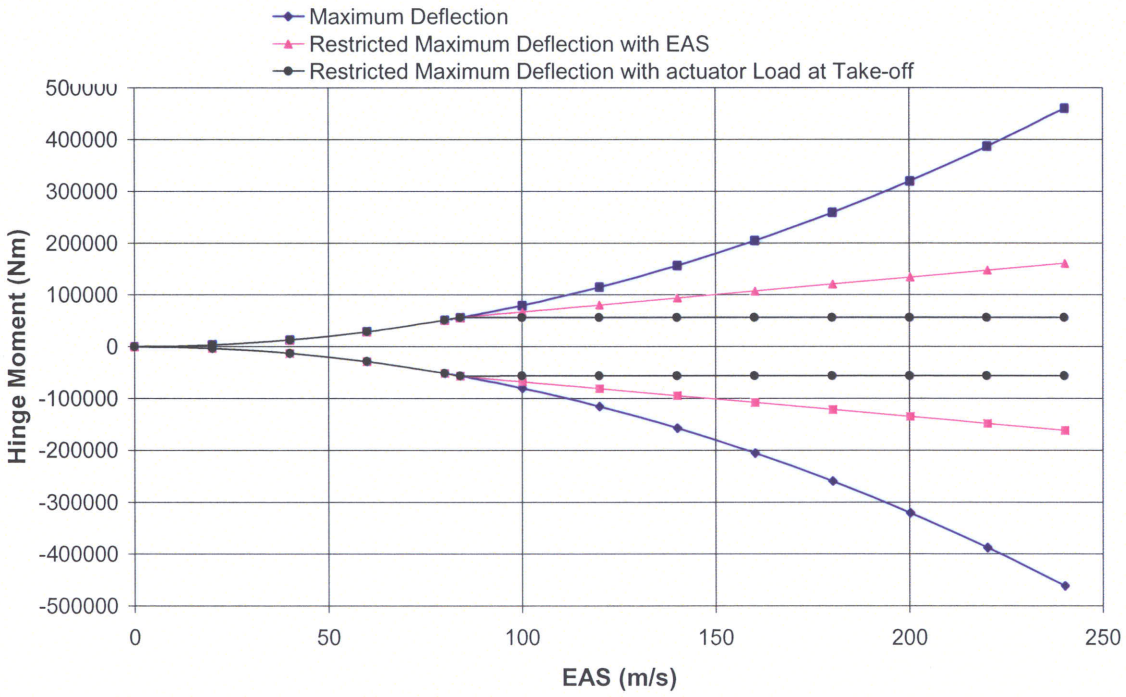


Figure 6.21: Calculated Rudder Hinge Moments for A340-500 Class Aircraft

Figure 6.21 also includes the calculated hinge moment coefficients assuming maximum rudder deflection at all speeds and maximum deflection as an arbitrary function of equivalent air speed. The resulting deflection is shown in Figure 6.22. Figure 6.23 shows the actuator load required to deflect the surface at various equivalent airspeeds.

Maximum rudder deflection of 25 degrees is included as well as the deflection defined by an arbitrary function of equivalent air speed. The moment arm was assumed to be the same with that of the Airbus A300-600 aircraft. Estimation of power at the surface requires the rotational rate of the surface. For the purpose of this study typical rates from published data on the Airbus A300-600 have been used. The maximum rate of rotation was assumed to occur at maximum deflection and the rotational speed for the remaining positions was varied linearly to achieve the same time required to rotate the surface. The resulting power at sea level take off conditions, i.e. 84 m/s forward speed, is shown in Figure 6.24. Actuator power is also shown assuming 85% efficiency.

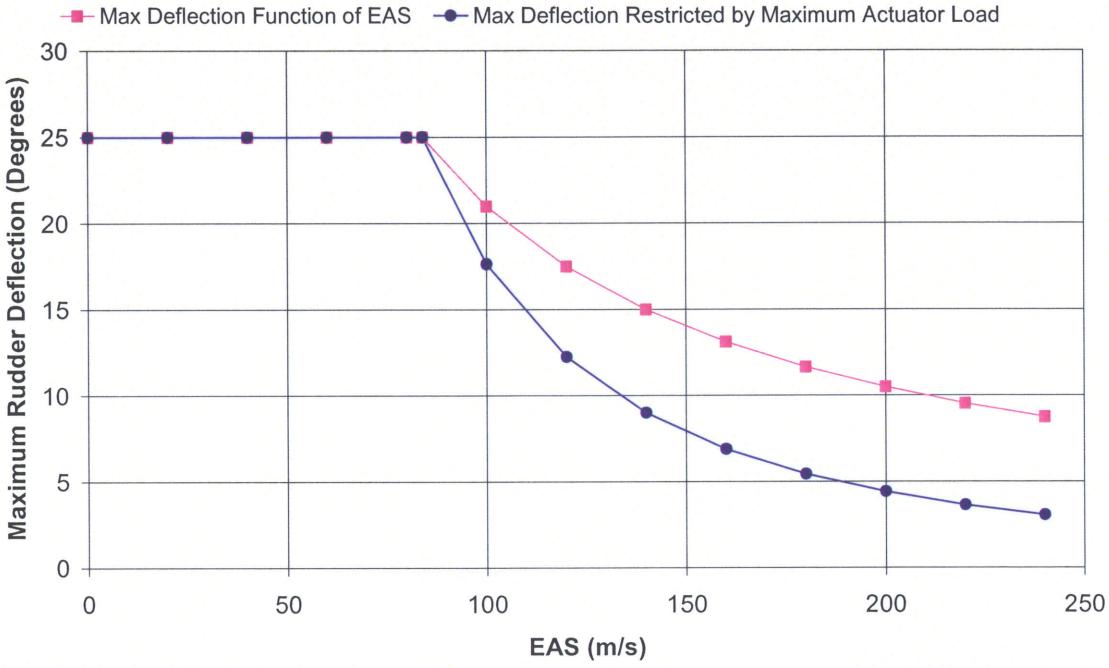


Figure 6.22: Maximum Rudder Deflection

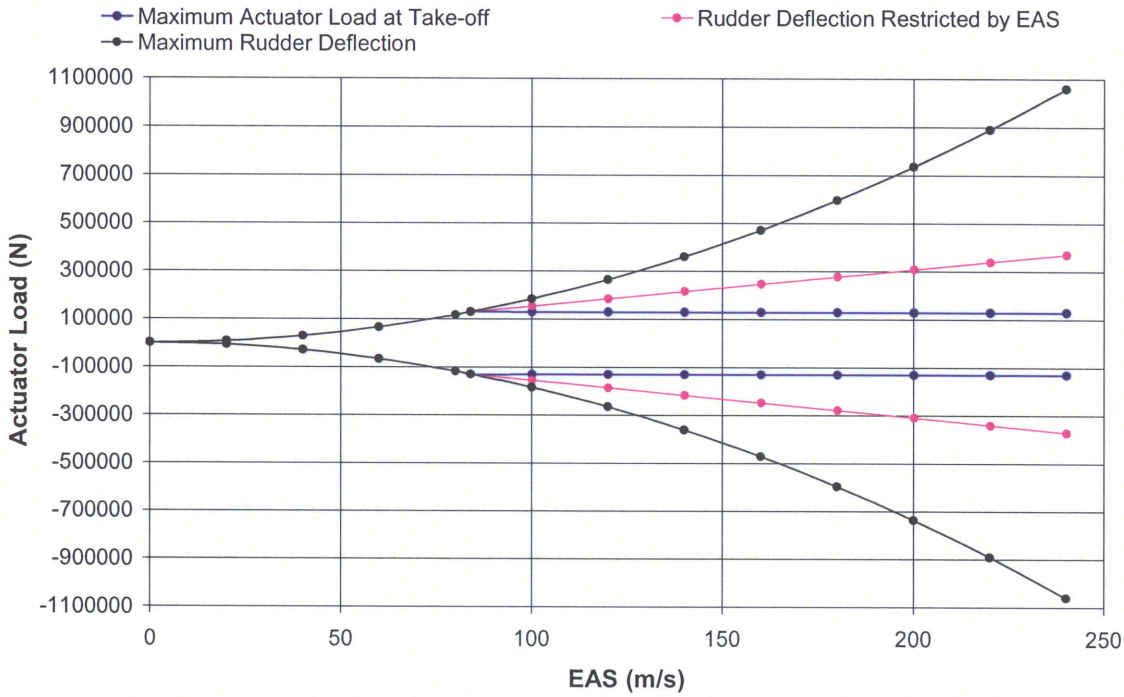


Figure 6.23: Calculated Actuator Load for Various Maximum Rudder Deflections and EAS

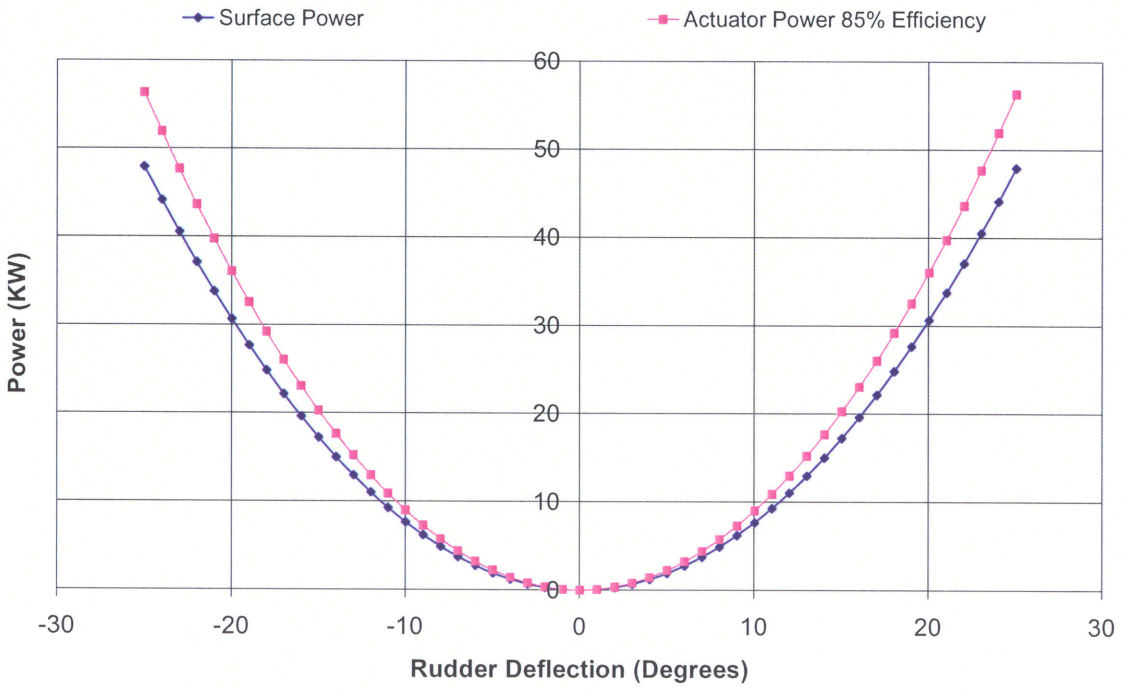


Figure 6.24: Power Required to Control the Rudder for the A340-500 Class Aircraft at Take-Off Conditions

Figure 6.25 plots the power requirements for maximum deflection restricted by actuator load as a function of equivalent air speed.

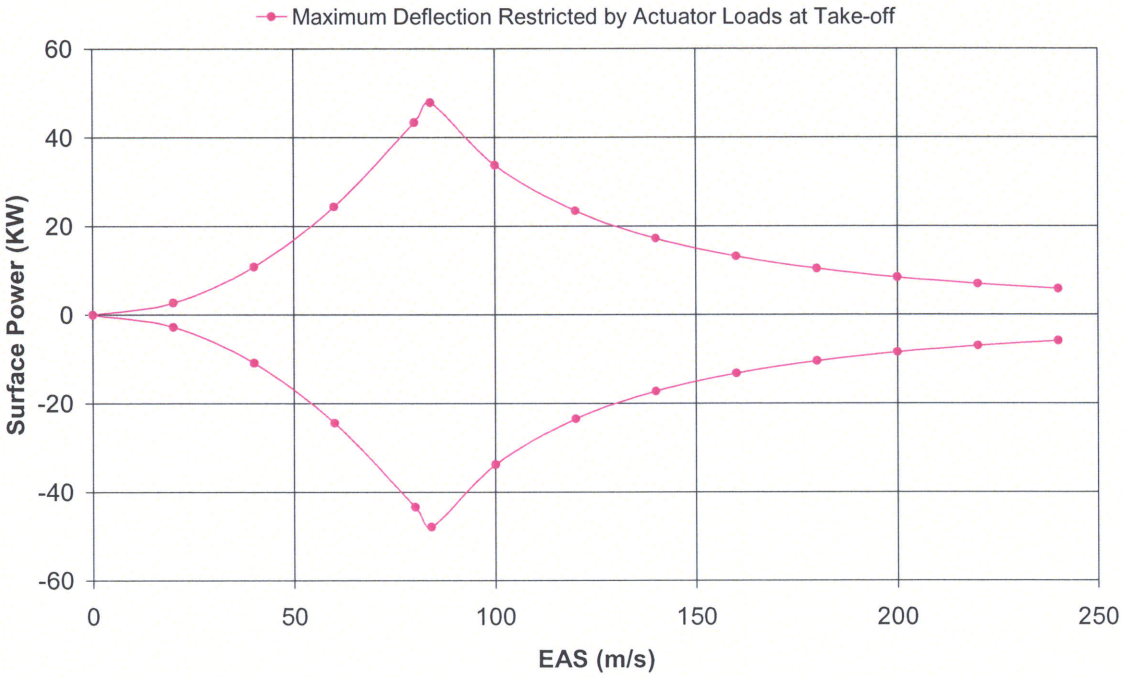


Figure 6.25: Rudder Power Vs EAS for A340-500 Class Aircraft

The calculated duty cycle for the rudder is shown in Figures 6.26 and 6.27 for deflection and power requirements respectively. The maximum deflection and power requirements are shown in addition to a typical duty cycle.

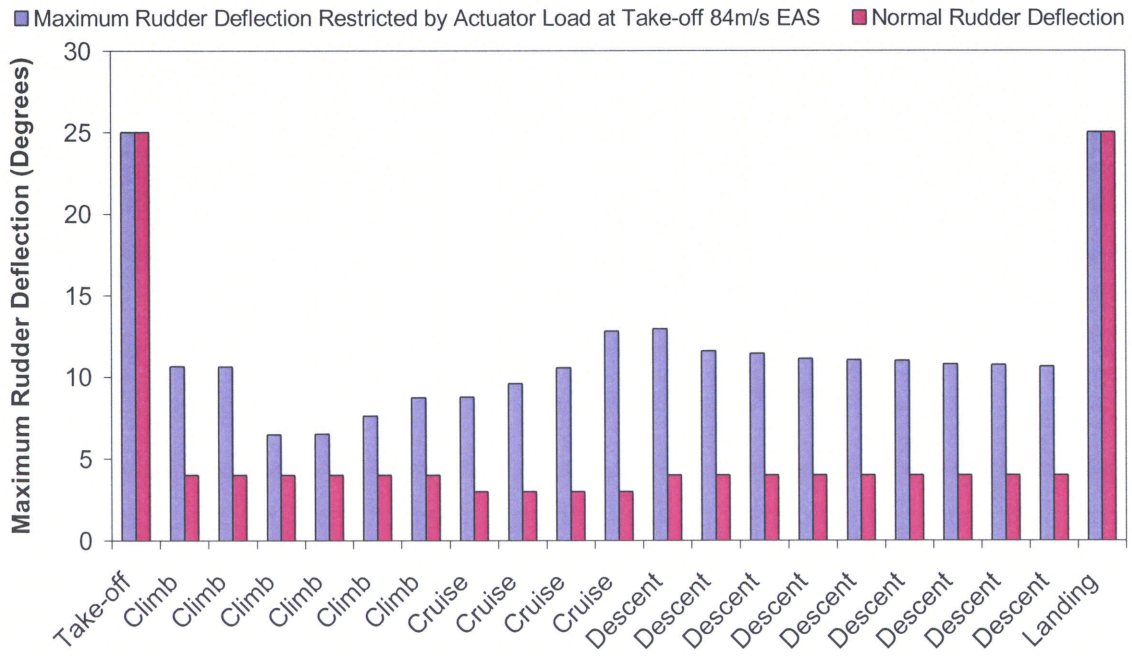


Figure 6.26: Rudder Duty Cycle for A340-500 Class Aircraft

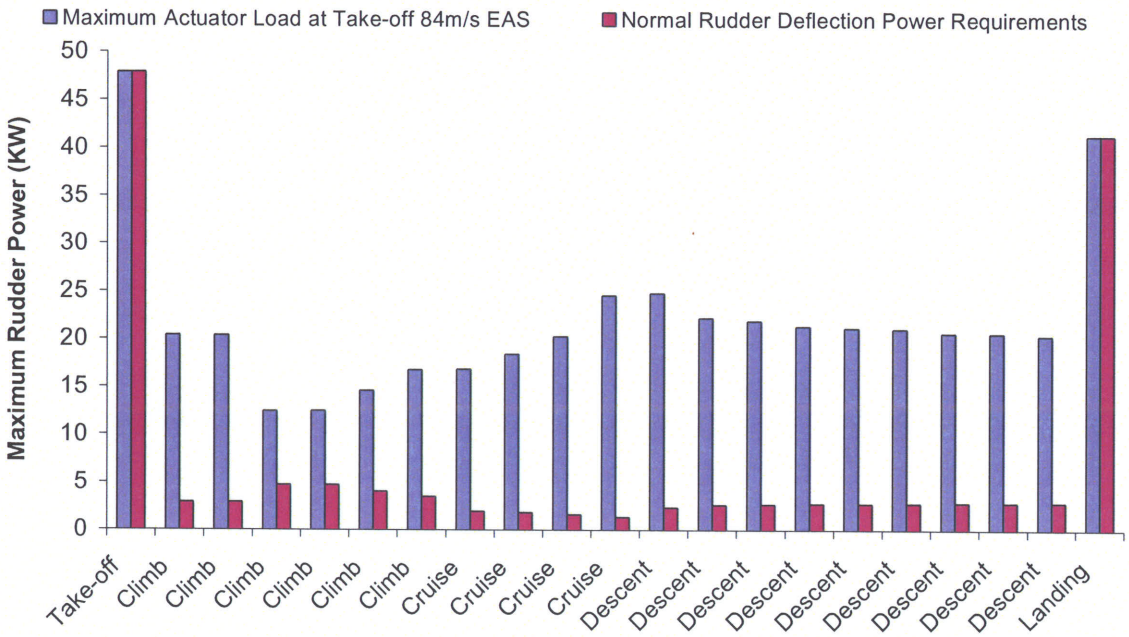


Figure 6.27: Maximum and Typical Rudder Power Requirements for A340-500 Class Aircraft

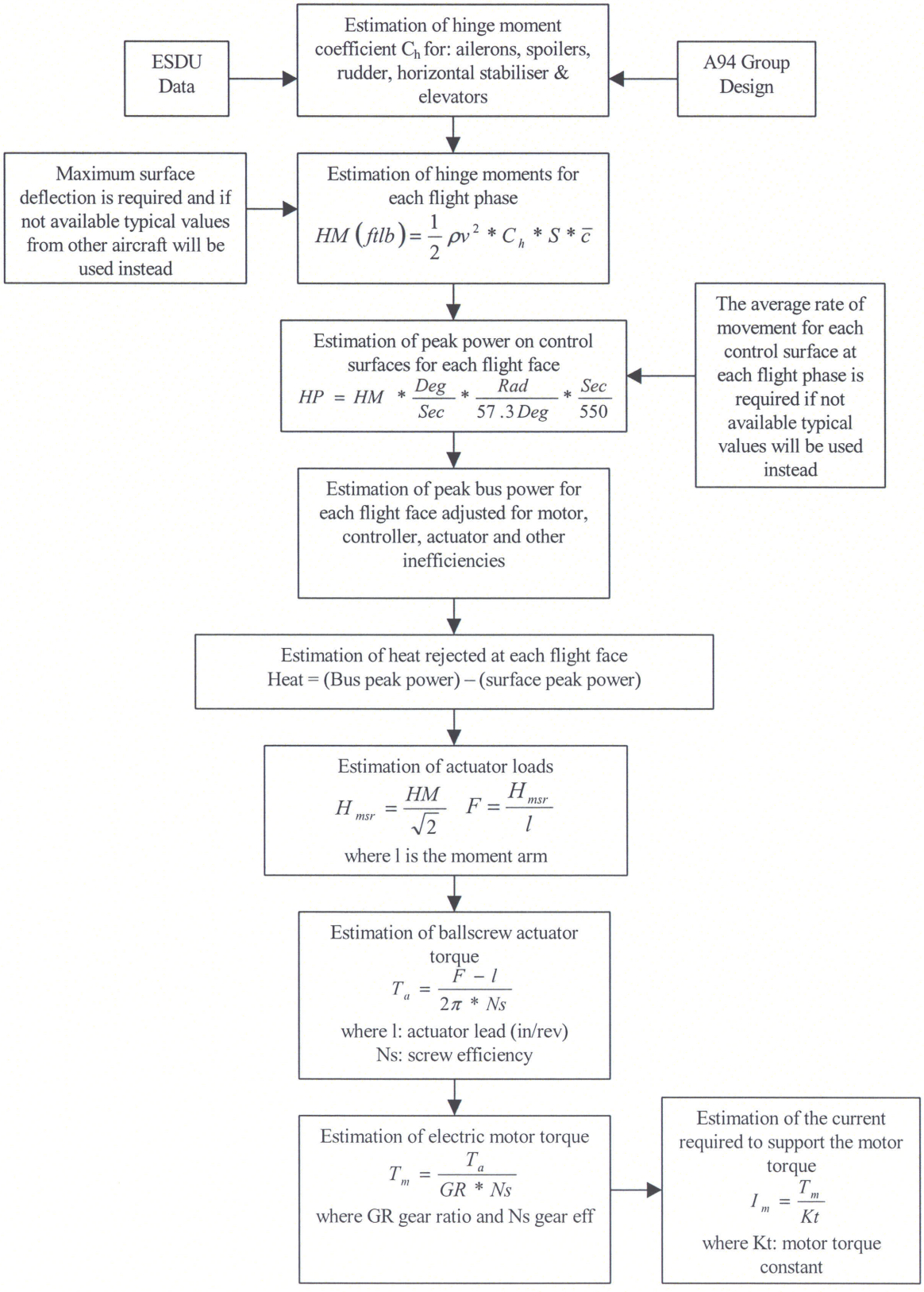


Figure 6.28: Method Applied for Estimating Power Requirements and Motor Characteristics for Electro mechanical Actuation of Primary Flight Control Surfaces

Figure 6.28 shows the computation sequence for estimating the power requirements and the characteristics of the motor for electromechanical actuated primary flight control system. The computation sequence also includes the estimation of heat rejected as well as the torque requirements for the motor. Based on the torque requirements the current required to support the motor can also be calculated. However, such a case requires unavailable geometric information and detail design of the motor. These steps have been omitted from the analysis. Detailed power requirements for the remaining flight control surfaces are shown in Table 6.6. The values have been used for sizing the generators in the case of an all-electric aircraft utilizing electric actuation.

Control Surface	Power (KW)
Elevator	2*17
Aileron	2*35
Spoiler	12*8
Rudder	50
Total	250

Table 6.6: Calculated Power for A340-500 Class Aircraft Flight Control System

The accuracy of the method has been validated against published data on the hinge moment of the Airbus A300-600 rudder. The results are shown in Table 6.7.

	A300-600	A340-500
Fin Area (m^2)	45.2	47.65
¼ Chord Sweep Angle (Degrees)	40	45
Fin Mean Chord (m)	5.58	5
Rudder Area (% of Fin Area)	33	37
Take-Off Speed (m/s)	80	84
Calculated Hinge Moment (Nm)	42707	56470
Hinge Moment for Airbus A300-600 (Nm) (Reference58)	41586	

Figure 6.7: Geometric Details and Calculated and Rudder Hinge Moment for A300-600 and A340-500 Class Aircraft

6.9 Electrical Power Generation

The type of generators suitable for the all or more electric aircraft depend on the system architectures and the loads required to be powered. In the past three main types of generators have been considered:

- Wound Rotor Machines
- Permanent Magnet Generators
- Switched Reluctance Generators

Wound rotor generators are currently used and have been described in the previous paragraphs of this report. A previous study[72] concluded that such a machine producing 200 KVA would weight around 91 kg. As explained earlier a wound rotor generator is a three stage machine and this alone increases complexity. In addition the windings and the rectifiers are on the rotor and are subjected on large forces. In addition from the currents flowing into the conductors require cooling of the rotor and the inner assembly.

The advances in rare earth magnet material during the 1970s combined with advances in high power semiconductor devices led to the design of high capacity permanent magnet generators. Such generators incorporate a permanent magnet mounted on the rotor and the winding are on the stator. As a result the windings are not subjected to rotational forces thus improving the reliability and life expectancy relative to the wound rotor generators. Since the windings are not placed on the rotor cooling of this component is not required.

Although the controllers of the generators may be complex they can be placed outside the engine in a less harsh environment. It is noted that all previous all electric studies assumed the use of permanent magnet generators. One particular problem of this type of generators is that excitation of the windings is always present provided that the magnet is rotating. As a result in the case of a generator failure and a short circuit, the generator must be disconnected from the engine. Operating temperature is also a problem for the rare earth magnets which at present have a working range maximum temperature around 200-250 ^{o}C.

An alternative to the permanent magnet generator is the switched reluctance generator. Such a machine has a very simple construction and as in the case of the permanent magnet generator the windings are on the stator. The rotor has no fewer poles than the stator and consists of stacked laminations with no windings. The machine can be used both as a motor and a generator. With current switched on in one set of winding the rotor will attempt to align a set of poles with those stator poles causing a torque to be produced as the poles approach each other. If the current is then switched off in these windings and on in the next set the rotor will continue to rotate and produce more torque. Therefore, by switching current on in the appropriate windings the machine can be operated as a motor. If current is switched on when the poles are parted the a resistance torque must be overcome causing a work input and the machine operates as a generator. Such machine provide a number of advantages including simple construction and elimination of winding from the rotor. In addition, excitation is only present with a winding voltage present and therefore the machine and each phase can be shut down externally. This implies that in the case of a short circuit or a phase being failed there is no need to disconnect the machine from the input driver but only to shut down the phase required. Therefore the machine can continue to generate partial power in the remaining phases.

The suitability of each machine depends largely on the type of application, the loads required to power and the overall architecture of the electrical system. These issues are considered in the next chapter of the study.

Chapter 7: Systems Architectures for the All/More Electric Aircraft

The chapter deals with possible systems architectures suitable for the all or more electric aircraft. Different technologies are discussed. The requirements for each system are considered separately and different architectures are presented. Possible electrical system architectures are then considered. The implications of component requirements and design are reviewed and overall conclusions are drawn. The analysis combines findings from previous studies on separate issues affecting systems layout and presents suitable architectures for the All/More Electric Aircraft and Engine concepts.

7.1 Introduction

Secondary power systems for the all electric aircraft depend on a number of factors including, type and number of power sources, type of generators, load requirements, type of aircraft and the overall requirements of each separate system. Over the years various electrical system technologies have been developed and reviewed. The main areas of research include the following:

- Power generation
- Power utilisation
- Power distribution, management and control

To be able to minimise the mass of the secondary power systems while improving their efficiency and performance consideration must be given to all the above issues. The type of generators used will have a strong impact on the form and design of the power distribution system and vice versa. The type of loads and motors used will also affect the design of the electrical power distribution system and its control. Finally, the requirements of each system, i.e. flight control system, environmental control system, etc., will also have an impact on the overall design of the electrical power generation, distribution and control system. For these reasons each topic is considered separately and the findings are combined to derive an overall electrical power system that would meet all the requirements.

7.2 Background of Electrical Power Generation and Distribution

Most forms of generators will produce variable voltage and frequency when driven by a variable speed source. Since the majority of generators installed on present aircraft are engine driven, the rotational speed of the engine will have a strong impact not only on the design and performance of the generator but also on the electrical distribution system. Most present aircraft incorporate a constant speed drive unit, which allows for the generation of constant voltage constant frequency electrical power. This form of power is best suited to the majority of present electrical loads. Another advantage of such a system is the fact that the generators are able to produce the same power regardless the speed of the engine. Use of constant speed drive systems comes with its implication and over the years various alternatives have been considered. A change in the form that secondary power is distributed would also affect the type of loads required to be powered from the electrical system and therefore constant frequency constant voltage power generation and distribution may not be the optimum form of power for all the loads involved.

This section considers the comparison of various power generation and distribution systems and their suitability to the all/ more electric aircraft/engine concepts. For the purpose of the study the following forms of power generation are considered:

- Constant Speed – Constant Frequency
- Variable Speed Constant Frequency
- High Voltage DC Systems
- Variable Frequency Constant Voltage
- Variable Voltage Variable Frequency

7.2.1 Constant Speed – Constant Frequency Systems

The system assumes the use of a constant speed drive unit or a combined integrated drive generator, as described in previous sections of this report. Such systems have been developed and used for the last 30 years and therefore they can be regarded as mature and proven systems. Because of their extensive development they have relatively low design, development and manufacturing costs and in some cases they

are lighter than alternative power generation and distribution systems. However their suitability is affected by a number of factors including complex design and reliability problems. One of their main advantages however is the fact that they can provide a constant power output regardless the rotational speed of the engine. As explained in Chapter 6 the power requirements of the conventional electrical power system remains almost constant throughout the mission profile. In addition, such a configuration is well suited in the case of paralleling and having two or more generators sharing one or more loads. However, for this to be possible the rotational speed of the constant speed drive unit must be controlled to avoid the generators fighting each other. Control over the rotational speed of the generator and its output frequency has also the advantage of un-interrupted power supply. This is essential in the case of avionics and navigation systems.

7.2.2 Variable Speed Constant Frequency Systems (VSCF)

In the case of VSCF systems the constant frequency is derived from static power converters that are equipped with variable speed variable frequency direct driven generators. VSCF systems operate on the principle that the variable frequency output of the generator is converted to constant frequency AC or constant voltage DC power. The output of the generators can be rectified within the generator housing or it may be rectified remotely in a static converter package. The following types of VSCF systems are available:

- Cycloconverter VSCF: AC to AC conversion
- DC Link: AC to DC to AC conversion
- High Voltage DV: AC to DC conversion

The Cycloconverter VSCF systems use Silicon Control Rectifiers (SCR), also called Thyristors, to convert the variable frequency power into a form close to 400 Hz then filter it to produce a smooth sine wave. The system has the advantage of being installed on a number of aircraft including the A-10, F117 F18 and V-22 tilt rotor. Despite its simplicity and the use of electronic assemblies, rather than mechanical and hydraulic components, the cycloconverter suffers from a number of drawbacks. One of its main characteristics is that the input frequency to the converter must be three to four times the output frequency. This implies that for a 400 Hz output the generator frequency at idle speed must be around 1200 Hz to 1400 Hz. As a result the generator must be driven at high speeds and incorporate a large number of poles. The former

requirement would increase the complexity and mass of the system requiring a step up gearbox between the engine and the generator, while the latter would increase the weight of the generator. Another disadvantage of the cycloconverter is that the power output of the generator must be 1.4 to 1.5 times the power at the distribution bus. As a result the generators must be oversized for no useful purpose.

An inherent disadvantage of all constant power VSCF systems is that the generator must be able to provide the full power at idle engine speed. In the case of high bypass ratio turbofan engines, the speed range of the high-pressure shaft can be close to 2:1. Because of this the generator is capable of providing twice its rated output at take-off and cruise conditions and as a result in the case of a constant power system the cycloconverter is heavier than the constant speed drive generators.

The DC link VSCF system is a DC to AC constant frequency/constant power system that rectifies the AC variable frequency power of the generator to DC power, then inverts it to 400 Hz AC power. Although the system does not have the disadvantages of the cycloconverter and it is simpler in its operation and designs it power output is also limited and has not been used extensively. Applications include the main generators on the MD 90 aircraft and the backup generators on the Boeing 777 aircraft.

The 270 DC system rectifies the output power from the generator to a high voltage DC power. This approach favours military applications but savings can also be present in the case of civil applications. One of the main disadvantages of such a system is that the majority of present loads can not be driven directly from DC power supply and therefore inverters must be employed for the loads that need to operate on AC power. Another disadvantage of the high voltage DC power is the development and testing costs required. An inherent problem of high voltage DC power supply is that natural arc extinction is not present (as it is in AC switchers) and this presents large problems in high altitude operations. One of the main advantages of high voltage DC systems is that they can easily be used to supply uninterrupted power to safe critical equipment.

7.2.3 Variable Frequency Constant Voltage Systems

This is one of the simplest and most reliable methods to produce and distribute electrical power. Only the voltage is regulated and the frequency is allowed to vary with rotational speed. However, the variable frequency has a detrimental effect on the size and efficiency of frequency sensitive equipment. In many cases the deteriorated performance of the electrical equipment may be accepted but in some applications converters or controllers may be necessary to ensure required performance. Variable

frequency constant voltage systems have been used widely in business jet aircraft such as the Global Express. Recently variable frequency has been selected as the baseline power generation and distribution on the forthcoming Airbus A380 aircraft.

7.2.4 Variable Voltage Variable Frequency

The system is different from the previous systems in that both the voltage and the frequency are allowed to vary with rotational speed. Unlike the previous system the power delivered by the generators is not constant but it is proportional to the speed of the generator and the engine. Although such a system may not be preferred for current applications where the power demands placed on the electrical system are constant it well suited with all electric aircraft applications. In reality, however this is a hybrid system, since some of the variable voltage variable frequency power delivered by the engine is used directly to power the majority of the loads, but a small percentage of each generator output is converted to high quality power. Although such a system is not in use at the moment of writing this thesis, all previous studies on the subject of more electric aircraft proposed the implementation of a variable voltage variable frequency system.

An important characteristic of such a system is that the voltage to frequency ratio must remain constant. Such an action may require a voltage regulation but results in more favourable motor performance characteristics. Cronin[5] reports that, voltage regulation is not always required and the constant voltage to frequency ratio could be achieved through the design of the generator. Such machines would be heavier than those requiring voltage regulation but would have improved performance. The main advantages of the system are its simplicity and high reliability, the fact that the output power is proportional to the speed of the engine and finally, having a constant voltage to frequency ratio improves the performance of electric motors.

7.3 Flight Control System Architectures and Requirements

If hydraulic power is to be replaced by electrical power then the system architecture will be influenced in great extent by the requirements of the primary flight control system. Power requirements for the primary flight control system could be used to size the generators. Although other functions such as retraction and extension of landing gear and/or flaps would place high power demands on the system these will be of short duration and can be provided by appropriate power management. The latter is a distinct advantage of the electrical power distribution system. In addition, the power requirements for the primary flight control system derived in the previous chapter of

this work assume maximum output force at maximum rate of rotation and simultaneous actuation of all primary flight control surfaces. Such a case is highly unlikely and therefore for the purposes of this work it is assumed that the derived power requirements would satisfy the demands of the present hydraulic system.

System architecture, on the other hand, would depend on the reliability requirements of the flight control system and the type of actuators used. Civil aircraft must achieve a failure in the order of $1.0*10^{-9}$/flight hour. In the case of military aircraft this target is reduced to $1.0*10^{-7}$/flight hour. To achieve these failure rates multiplication and high levels of redundancy are required. As mentioned before, military aircraft incorporate tandem actuations. In this case, two independent hydraulic systems are used. The reliability of the system can be approximated by using reliability block diagrams. Such a diagram for a tandem actuator is shown in figure 7.1.

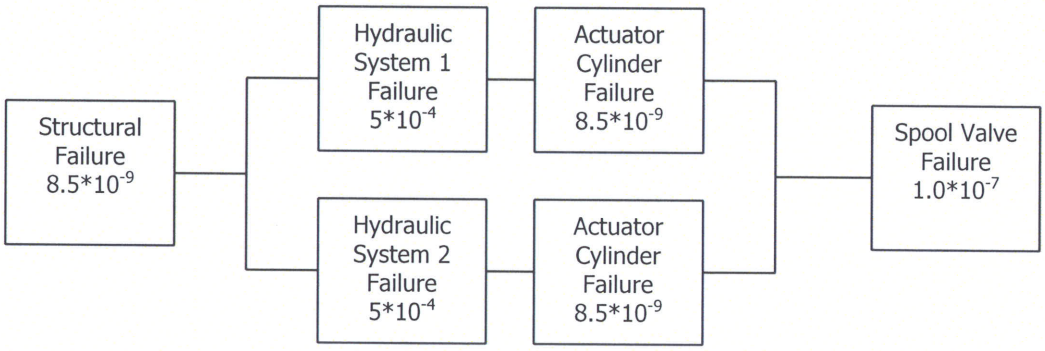


Figure 7.1: Simplified Reliability Block Diagram For Tandem Actuator

Structural failure refers to the failure of the rod, mounting lugs, input linkage and the rod end. The structural failure rate was taken from Reference 135. Hydraulic system failure refers to the loss of hydraulic power, including pump failure, pipe failure, etc. Once again the failure rate was taken from the same reference. Actuator cylinder failure was assumed to be the same with the structural failure. Tandem actuators are usually constructed into two separate cylinders that are mounted together. If on of the cylinder fails the remaining cylinder is able to provide the power required and such a failure would mean total loss of hydraulic power in one system only. Spool valve failure refers to the jam of the control valve. Some tandem actuator configurations employ a single spool valve while others use two separate valves. The overall failure of the system can be calculated by adding the failure rates of the components that are in series and multiplying the failure rates of the components that are in parallel. The main areas of concern in such a case is the failure of a hydraulic system which has a

higher failure rate than the required $1.0*10^{-7}$. Structural failure on the other hand is lower than that required, while valve failure is also a main concern. The overall failure rate of the system is around $3.58*10^{-7}$. In practise the failure rate of the hydraulic system is improved through the use of electrically driven back-up pumps and power transfer pumps previously mentioned. Additional redundancy is provided through the use of accumulators and RAT driven pumps. Such actions can bring the failure rate of the whole system close to the target required but failure of the spool valve is still an issue. This problem can be alleviated through the use bypass valves. Although the analysis is simplified and does not include failure of flight control computers it provides a better understanding of the issues involved and indicates typical system failure rates.

In the case of civil applications all failure rates, i.e structural failure, hydraulic failure and control valve failure are higher than those required. For these reason at least two separate load paths are required. A typical reliability block diagram for a primary flight control actuation system is shown in figure 7.2.

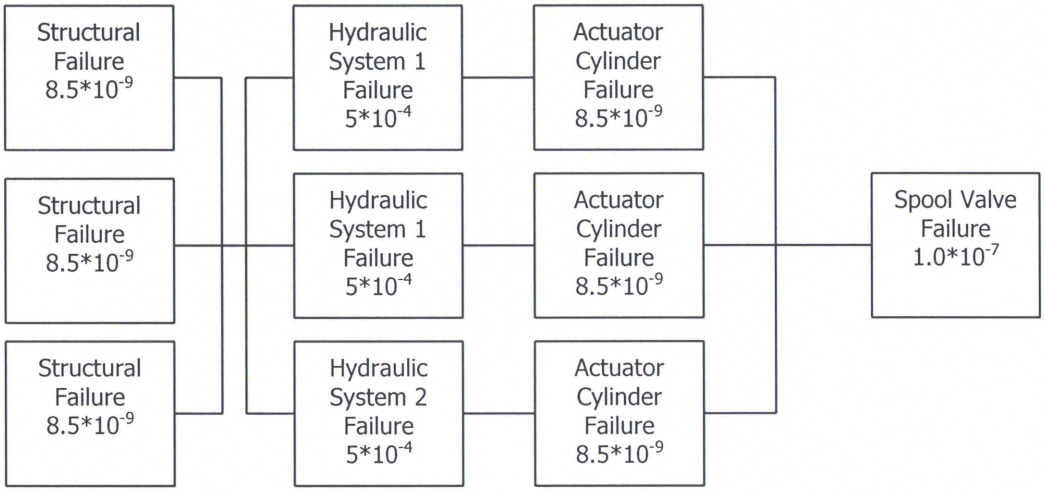


Figure 7.2: Simplified Reliability Block Diagram for Civil Primary Flight Control Actuation

In such a case both structural failure and hydraulic system failure are lower than the target and the only concern is the failure rate of the spool valve. It is noted that only a single spool valve is considered and is connected in series to the rest of the system. This is because even if three valves are incorporated failure of a single valve will lock the system. However if a bypass valve is connected in parallel to the spool valve then the required target is easily met.

In the case of electro-hydrostatic actuators the additional components are the electric system power supply, the electric motors and the controllers. A typical block diagram is shown in Figure 7.3.

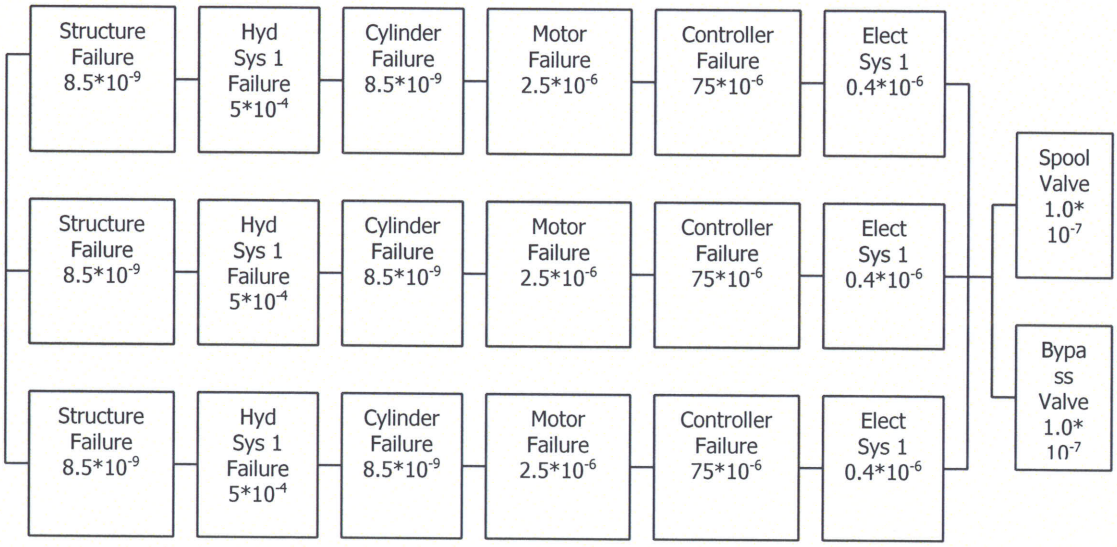


Figure 7.3: Reliability Block Diagram for Electro-Hydrostatic Primary Flight Control Actuation System

The system described above would utilise three separate electro hydrostatic actuators. Each actuator can be driven by a single electric motor that would be controlled by a single controller. Although the system shown above assumes three different electric power systems two separate electrical power systems can meet the reliability requirements of the system. This is important in the case of a twin-engine aircraft such as the A330-200 class aircraft.

In the case of electromechanical actuators Maydew[134] in his PhD thesis considered various system architectures. He concluded that two load paths would be enough to meet the reliability requirements. His work also emphasised the need for jam tolerant actuators to meet the reliability targets of primary flight control systems. The proposed architecture is shown in Figure 7.4. Because of the higher reliability of the electrical system relative to a hydraulic system only two such systems are required. With such a configuration, however, the output of the two motors must be added before entering the jam tolerant electromechanical actuator. Maydew[134] proposed the use of a speed summing differential as shown in Figure 7.5 to avoid a motor failure jamming the whole system. Such a system however would reduce the movement rate of the control surface in the case of a motor failure. An alternative design, also proposed by the same author, is the use of two separate driven jam tolerant actuators. The system is

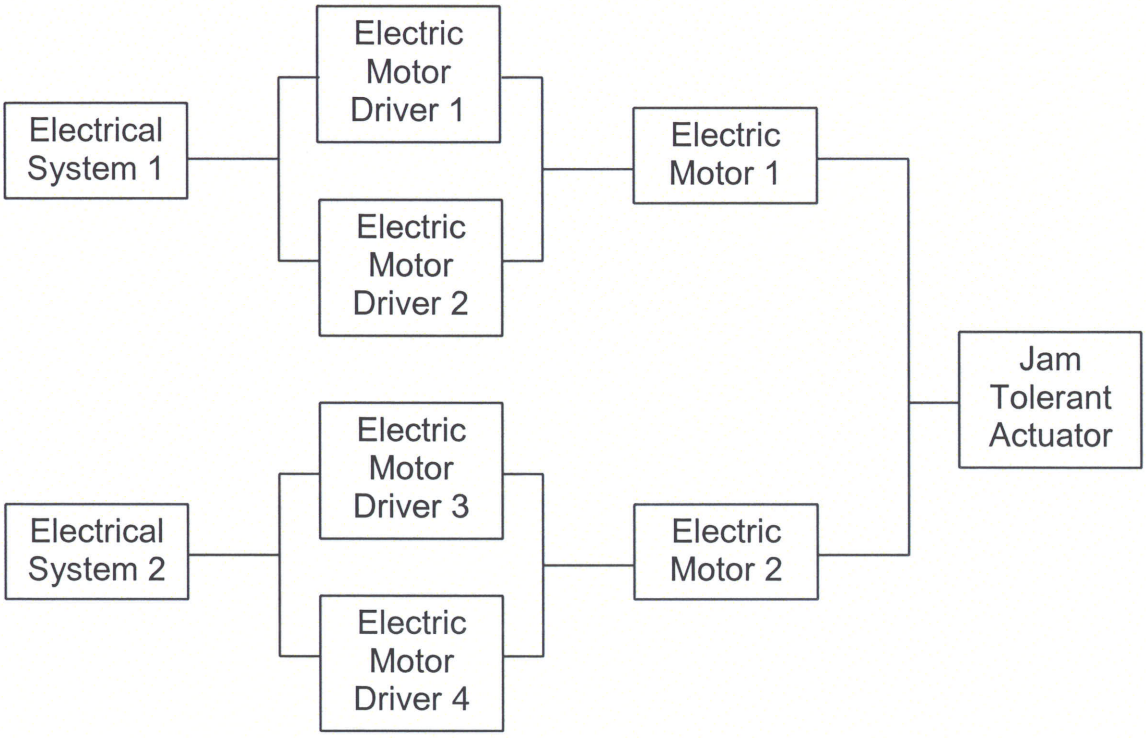


Figure 7.4: EMA Architecture with 2 Electrical Power Supplies

shown in Figure 7.6. Under normal operating conditions, each motor drives one jam tolerant actuator. The brake is used to provide a reaction of the differential gearbox. In the case of a motor jam the brake is released and the jam tolerant actuator is free to move with the surface. More information regarding both configurations and the design of jam tolerant actuators is given by Maydew [136].

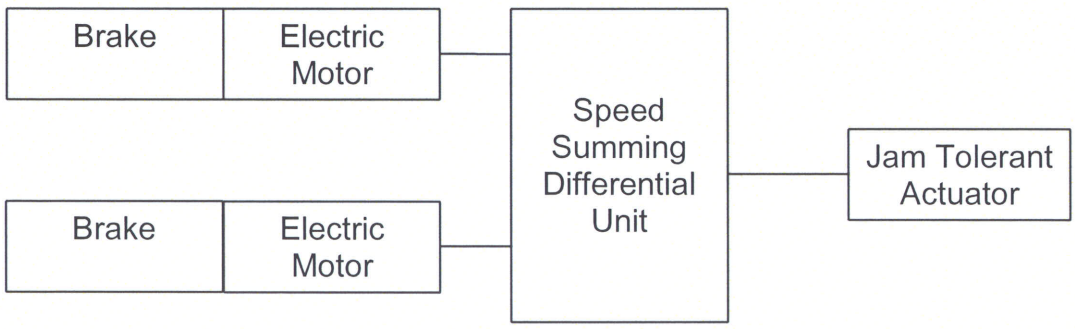


Figure 7.5: EMA Architecture Utilising a Speed Summing Differential Unit

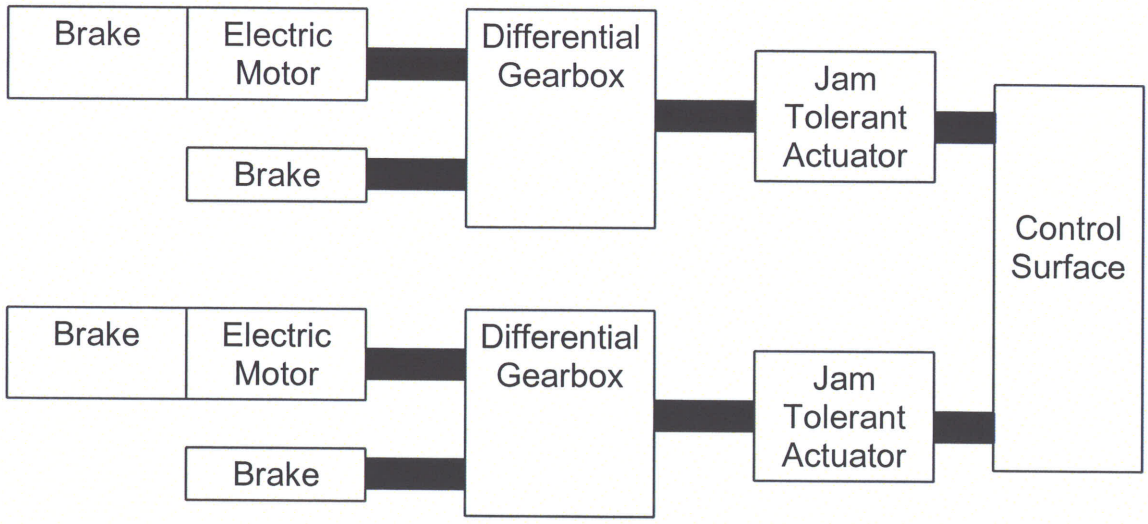


Figure 7.6: EMA Architecture Utilising Two Jam Tolerant Actuators with Differential Gearbox

7.5 Environmental Control System Requirements

The next system to be considered is the environmental control system. In the case of separate electrically driven compressors the environmental control system would be the highest load on the aircraft and therefore would have a significant impact on the design of the electrical power system. Power requirements for the system have been derived in Chapter 6. For all three baseline aircraft four separate compressors could be used each one driven by a separate motor and controller. Either two or more electrical systems can be used to supply the power required. Unlike present cooling fans that are driven at constant speed, control of the environmental system compressors may require variations in rotational speed. A typical compressor map showing the operating line of the environmental control system compressors is shown in 7.8. Pressure ratio and non-dimensional mass flow at different operating conditions are shown. Speed lines have been omitted.

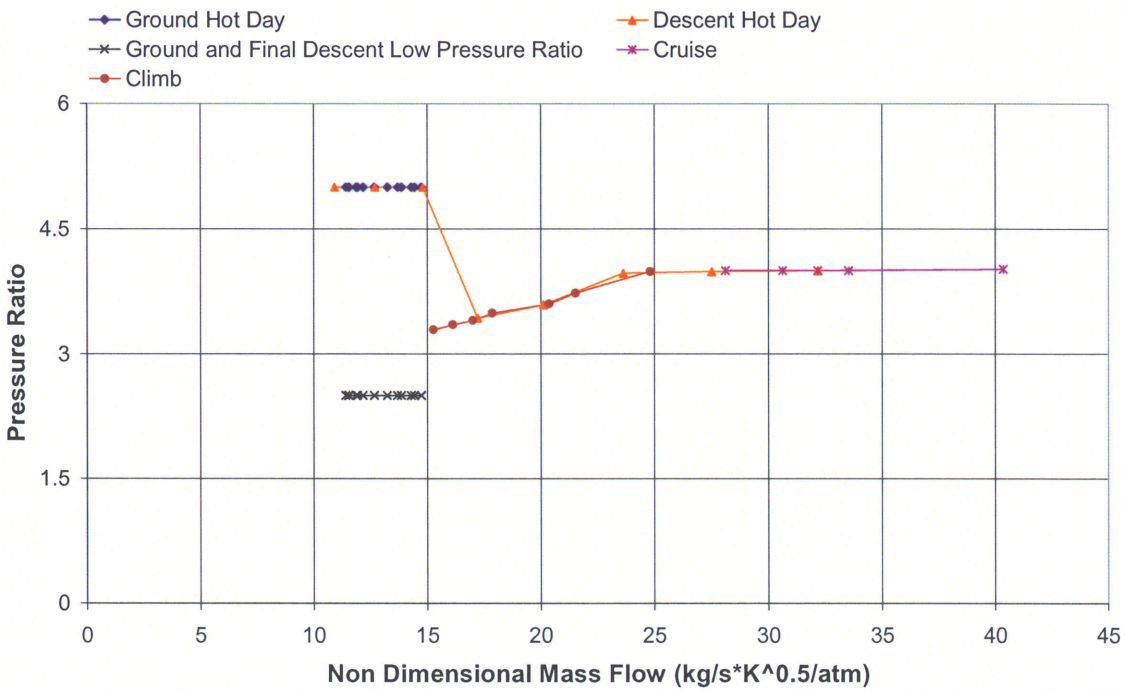


Figure 7.7: Typical Pressure Ratios and Non-Dimensional Mass Flows for ECS Compressors (Note Mass Flow is Constant and Equal to 0.699kg/s)

The pressure ratios and mass flows have been derived by considering the pressure ratio required for cooling and cabin pressurisation at different operating conditions. The operating line shown in Figure 7.7 implies that the non-dimensional speed of the compressor increases with non-dimensional mass flow and altitude. The actual rotational speed, however, could remain constant or increase slightly with non-dimensional speed and altitude. If the rotational speed of the compressor remains constant the highest torque would be required at hot day operations and while on the ground. Torque requirements would reduce at the climb and cruise section of the mission profile while the lowest torque would be required on the ground and descent during cold day operations. If the rotational speed of the compressor is allowed to fall with reducing non-dimensional mass flow then the maximum torque would increase and would occur at lower rotational speed. The minimum torque would also increase and would also occur at lower speed.

To ensure the required performance of the compressor these torque vs speed characteristics should be matched with the characteristics of the electrical motor used to drive the environmental control system compressors. The electrical power generation and distribution system would affect the characteristics of the motor and therefore these requirements must be considered.

7.6 Electric System Architectures

A number of electrical system architectures have been examined. As mentioned before the design of the electrical system would depend on a number of parameters. If a change is made to the distribution of secondary power and single type of power is adopted, then the configuration of the present electrical system may have to change. The main advantage of the conventional systems is that they provide a constant power output at constant volt and frequency. Although this is an advantage in the case of conventional electrical loads this is not true in the case of electrically driven secondary power systems. The majority of present electrical power is required for loads such as lighting, operation of cooling fans, galley and oven loads and powering avionics and communication equipment. Some of these equipments require constant voltage and frequency. Previous studies on electric power generation and distribution concluded that some of the electrics loads could be supplied with variable frequency power. For this reason they considered alternative methods of producing and distributing electric power. The alternative methods have been presented in the previous paragraphs of this work. The studies concluded that adoption of VSCF and/or variable frequency system would depend on the amount of power that needs to be regulated to standard 115V, 400 Hz supply.

In general, as the percentage of power required to be regulated reduces the mass of the electrical system also reduces. However, for reasons previously mentioned VSCF systems would always present a mass penalty even when compared with constant speed drive systems. Therefore the main advantage of using VSCF systems is the elimination of the integrated drive generator and its reliability problems. In general trade-off studies presented by Kramer [138] and Wilson [139] have shown that if the percentage of constant frequency required is less than 30% of the total then a variable speed or a hybrid system becomes more attractive.

From the existing electrical loads the galleys and some heating equipment can be supplied with variable frequency power. In the case of the A340-500 class aircraft the total power requirements are around 210-250 KW. The galleys require around 120 KW. This is almost half the load. The remaining loads are used to power fuel pumps, cooling fans, lights and avionics equipment. Avionics require around 30 KW. This power is usually converted to 28 V DC power and therefore does not depend on the frequency output of the generators. From the remaining loads the majority refer to fuel pumps and fans. These equipments are powered from AC induction motors. Induction motors have unfavourable torque and power characteristics when supplied with constant voltage variable frequency power. This means that as the supply

frequency increases the torque and power output reduce. As a result the motors need to be oversized or controllers must be employed. Both actions however would increase the mass of the system. It is these loads and the power required for internal and external lighting that currently limits the use of variable frequency or a hybrid system. In the case of the more electric aircraft the load characteristics and requirements would change and therefore the power generation and distribution system can be allowed to change as well.

If electrically driven compressors are to be used for the supply of the environmental control system then the speed of the compressor may need to be controlled to achieve the required efficiency, mass flow and pressure. The torque vs speed characteristics of an induction motor supplied with constant voltage variable frequency power are shown in Figure 7.8.

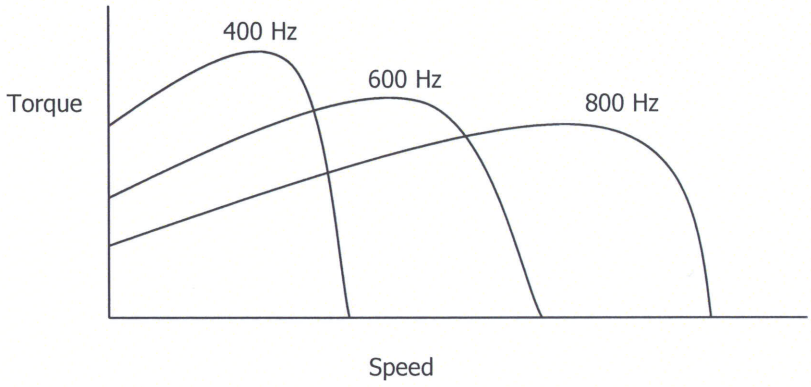


Figure 7.8: Induction Motor Characteristics for Constant Voltage and Variable Frequency Supply [5]

It is shown that as the frequency increases both the torque and the maximum power output reduce. Figure 7.9 shows the torque vs speed characteristics of an induction motor supplied with variable voltage variable frequency power. In this case the ratio of voltage to frequency remains constant. The torque of the system remains constant and the power output increases with frequency. However, for a given torque the speed also increases with frequency.

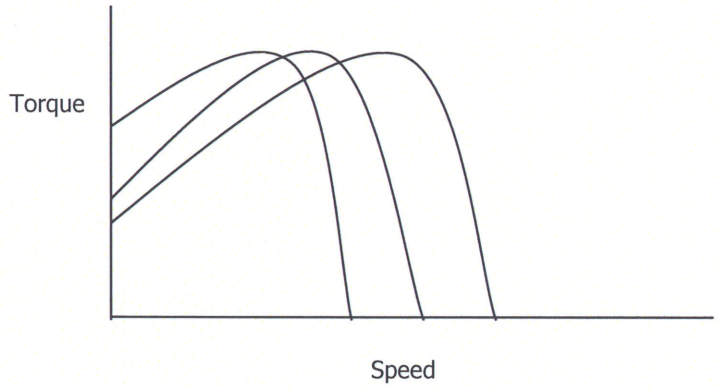


Figure 7.9: Induction Motor Characteristics for Variable Voltage and Variable Frequency Supply [5]

If a constant frequency generation and distribution system is employed then the power output would be constant. In such a case the generators would be sized for take off conditions were the maximum power requirements occur. As mentioned before the main load on the electrical system would be the power required to drive the separate compressors. If an integrated drive generator is used the generator would rotate at constant speed supplying constant frequency and voltage to the electrical system. In this case induction motors could be used to drive directly the compressors. The design point of the motors would be the hot day ground operation where the torque and power requirements are maximum. Assuming that the compressors are allowed to run at approximately constant rotational speed then the motors can be driven directly from the power supply system. However some short of speed control may be required.

In the case of a variable speed constant frequency system the generators must be sized for the power required at take-off conditions but the rotational speed of the generator would be at idle engine conditions. The size, mass and power output of a generator are proportional to the rotational speed. This would oversize the generators for no useful purpose. The problem may be alleviated with the use of a step up gearbox but it will still be present. If all the output from the variable speed generator needs to be converted to constant frequency power then the system would be heavier than the equivalent constant speed system. In the case of variable frequency power, the frequency would be determined to large effect by the rotational speed at idle conditions. This would be the minimum frequency. It is reasonable to set this frequency at the current standard which is 400 Hz and allow it to increase at the take-off, climb and cruise conditions. The frequency output at take off conditions would be close to 800 Hz. At cruise conditions the frequency output would be around 720 Hz. In such a case the torque speed characteristics of the induction motor can be matched for the ground and cruise conditions if the rotational speed of the compressor is allowed to

increase with non-dimensional mass flow. The compressor and motor characteristics, however would not match at take off conditions, unless the motor is oversized. Using the APU generators at take-off to drive the compressors could alleviate the problem. However, speed control of the motor would be required at climb and descent conditions. In this case either the load supplied to the motors must be of constant 400 Hz frequency or controllers must be employed.

In the case of variable voltage variable frequency power generation, the design point would be the power requirements on the ground and idle conditions. In the case of the A340-500 aircraft the power requirements while on the ground would be around 150 KW per engine and/or per generator. The load represents the power required to drive the environmental control system compressors on a hot day. The additional loads, i.e. lighting and other electrical equipment can be supplied by the APU generators. At take off conditions the power output of each generator would be twice the power output at idle conditions giving a total of 1200 KW. This output is enough to cover the maximum power requirements at take-off. At cruise conditions the power requirements per generator would be around 180 KW. Once again the generators would be able to meet this load. In such a case the same problems as with the variable frequency constant voltage supply are present and therefore the motors would need to be controlled. However, the controllers of the motors can be used to start the engines and eliminate the need for an air starting system.

If hydraulic actuation is replaced with electrical actuation then the actuators must be controlled to meet the power, speed and position requirements. Therefore, motor controllers are required regardless the type of motors and the type of power distributed. From the remaining equipment the fuel pumps are sensitive to the quality of power and would require some form of controllers. In the case of the all electric engine however where hydraulic pumps could be retained but need to be electrically powered then these pumps may need to be supplied with constant frequency constant voltage power. However, hydraulic pumps are usually variable displacement pumps that utilise a variable angle swash plate. The maximum torque remains roughly constant with speed and the power required is proportional to speed. If a variable frequency constant voltage is supplied to the motor then the motor has to be oversized only to provide the required acceleration. Supply of variable voltage variable frequency power would meet the requirements of the pumps and there is no need to oversize the motors. The maximum power requirements for the A340-500 class aircraft are shown in Table 7.1.

	System	Power (KW)	Comments
Conventional Electrical System	Avionics	30 KW	Sensitive
	Ventilation	10 KW	Sensitive
	Galleys	110 KW	Non-Sensitive
	Fuel Pumps	50 KW	Sensitive
	Lighting	20 KW	Sensitive
	Total	**220 KW**	**4*75 KVA**
All/More Electric Engine	Engine Pumps	150 KW	Sensitive
	Hydraulic Pumps	240 KW	Sensitive
	ECS	400-600KW	Sensitive
	Total	**1010-1210**	**4*250-300 KVA**
All More Electric Aircraft	Electric Actuation	240 KW	Sensitive
	Total	**460-1210 KW**	**4*115-300 KVA**

Table 7.1: A340-500 Electrical Power Requirements

The various power generation and distribution methods described above are shown diagrammatically in Figures 7.10 – 7.12. The mass changes of the systems for the three aircraft considered are shown in Tables 7.2-7.6.

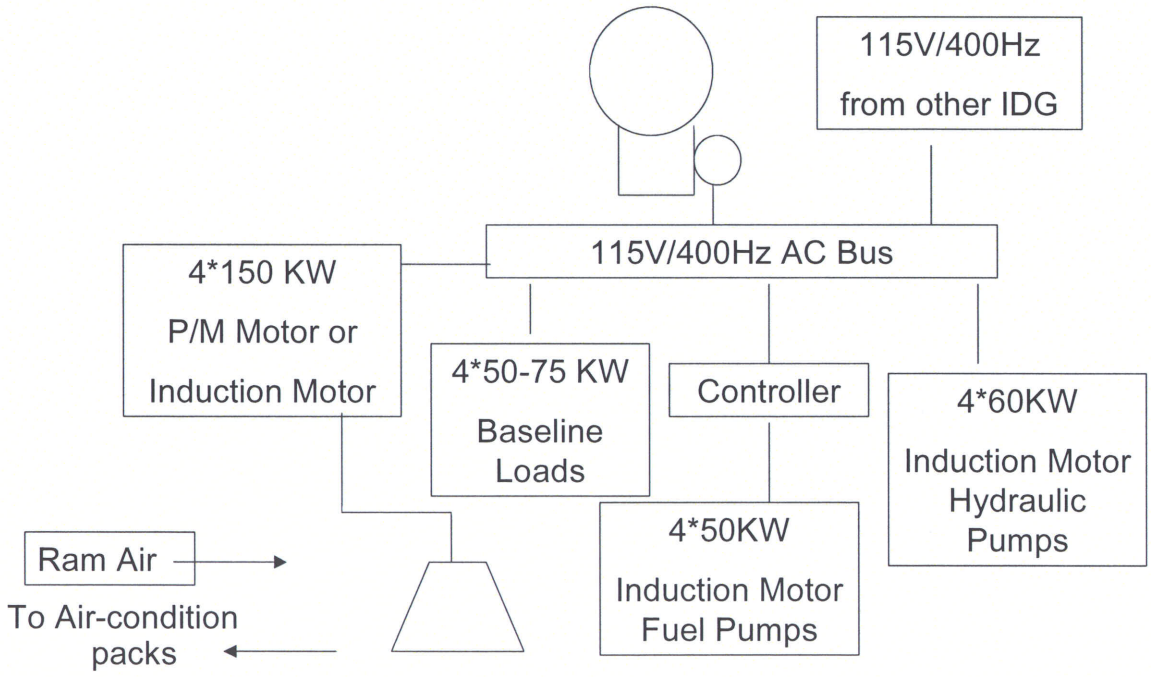


Figure 7.10: Simplified Diagram of Constant Speed Constant Frequency System

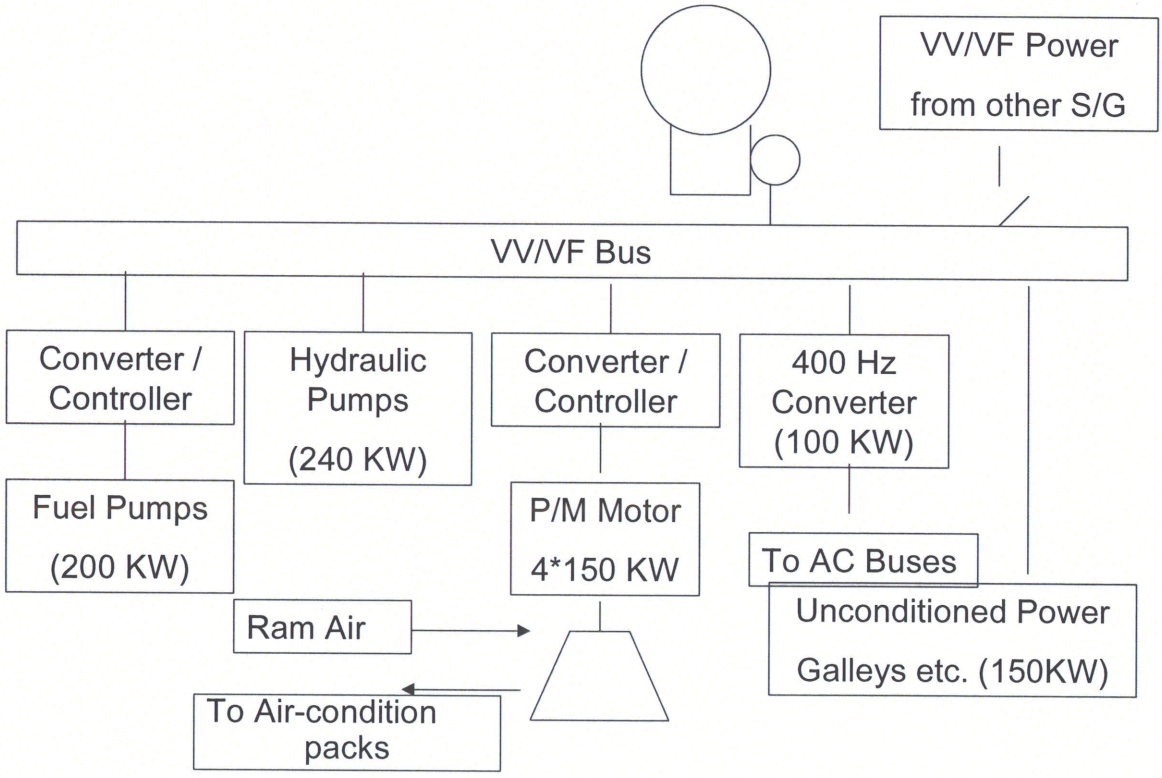


Figure 7.11: Simplified Diagram VFVV System

The figures assume that the hydraulic system is retained and the pumps are electrically driven. If electric actuators are used then the power would be supplied either by the AC 400 Hz bus or a dedicated 270V DC bus. Electric actuators are likely to be powered from permanent magnet brushless motors. The controllers of the motors would operate with DC power. Use of a dedicated 270 DC bus would simplify the system and the power could be derived by DC link inverters. Then converters could be used to reduce the voltage at the motors. If the actuators are powered from the AC bus then either permanent magnet brushless motors or induction motors could be used. The permanent magnets would require full wave rectifiers to produce the DC power necessary. In the case of a constant speed constant frequency system use of the AC bus would simplify the system and would eliminate the need of a DC link.

In the case of the constant speed drive system the generators cannot be used to start the engines. Instead the current air starter must be retained along with its associated ducting. For this reason mass savings from the elimination of the pneumatic system are considerably lower. As a result the overall mass of the system is higher than that of a variable voltage variable frequency system and a DC link system.

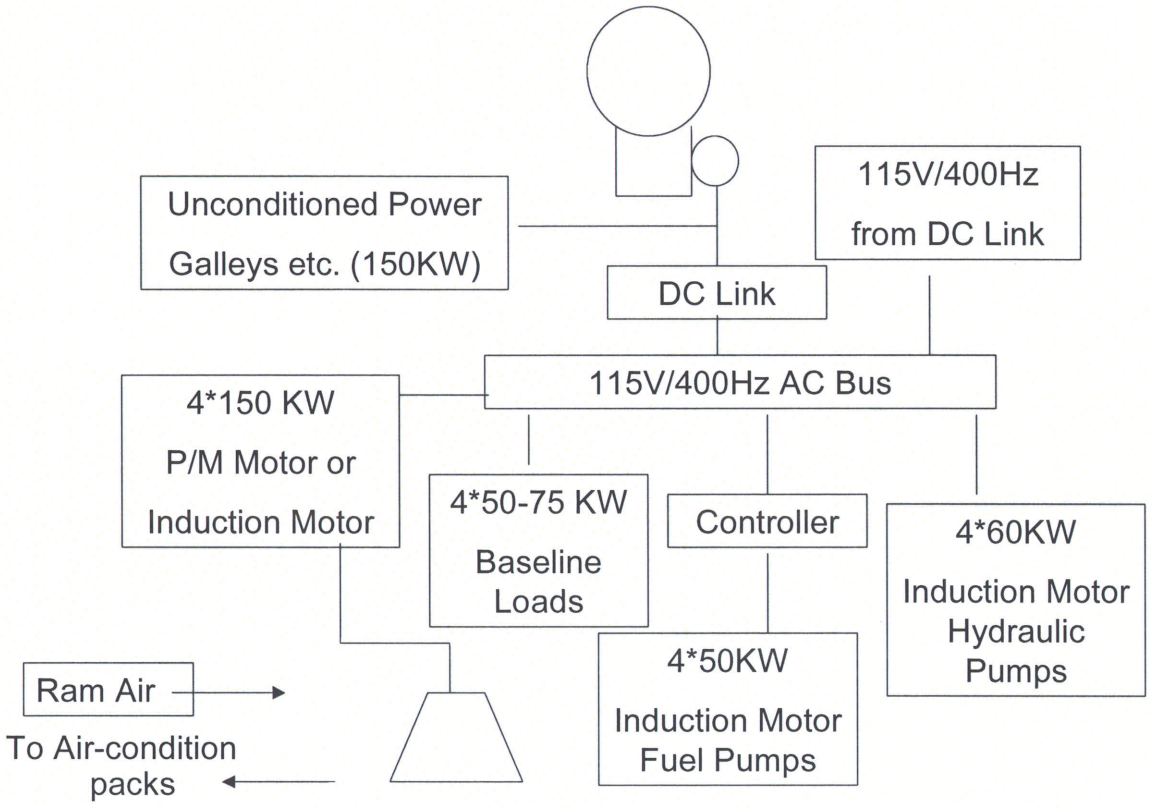


Figure 7.12: Simplified Diagram VSCF System

A comparison between the constant speed system and the variable voltage variable frequency system reveals that additional mass penalties are introduced in the case of the constant speed drive system because of the size of the generators. As previously mentioned the constant speed drive system must be sized for maximum loads at take off conditions. The variable voltage variable frequency system, however, is not a constant power system and the generators can be sized for the loads at the minimum rotational speed, provided that the power requirements at the remaining operating conditions are met. The size of the generators would be determined by the power requirements at the landing phase. At such conditions maximum hydraulic or actuation power may be required in addition to the loads of the electrical and environmental control system. The rotational speed of the engine and the generator at such conditions would be slightly higher than the idle speed. The total power requirements during the landing phase could be as high as 1200 KW. However not all of this power is required. For example a significant percentage of the electrical load can be shed. This includes the galleys and a percentage of the power used for internal landing. Similar loads can also be shed to reduce the overall power requirements. For example due to the small duration of the landing phase the environmental control system may

be allowed to operate at lower pressure ratio or even mass flow. Such an action could reduce the power requirements by another 200 KW bringing the total power requirements close to 800 KW. Therefore each generator can be sized to provide 200 KW at the landing phase. This rating would also satisfy the power requirements at the remaining flight phases. In the case of an engine failure the generators can be overpowered to supply the power required. This is common practise and the generators designed to be able to provide 1.5 times their rated output for 5 minutes. In the case of the A340-500 class aircraft each generator would be able to supply 300 KW. This power output exceeds the power requirements at the landing phase.

Equipment	All Electric Aircraft	More Electric Engine
ECS Removed		
Ducts, Precooler, Valves, etc	-293 Kg	-293 Kg
ECS Added		
4 Compressors	+104 Kg	+104 Kg
4*155KW Motors (0.3-0.5 Kg/KW)	+186 Kg (+310Kg)	+186 Kg (+310Kg)
Power Generation Removed		
4*75 KVA IDG (0.5-1 kg/KW)	-150 Kg (-300Kg)	-150 Kg (-300Kg)
APU Gen 125KVA (0.27 Kg/KVA)	-32 Kg	-32 Kg
Power Generation Added		
4*250 KVA IDG	+500 Kg (+1000Kg)	+500 Kg (+1000Kg)
2*180 KVA APU Generators	+97 Kg	+97 Kg
Elimination of Hydraulics		
Net Mass Saving	-750 Kg	
Electro Impulse De-Icing	-144 Kg	-144 Kg
4*60 KVA Electric Hydraulic Pumps		+72 Kg (+120 Kg)
4 * 50 KVA Electric Fuel Pumps	+60 Kg (+100 Kg)	+60 Kg (+100 Kg)
Overall Mass Changes	-422Kg (+84)	+400 (+962 Kg)

Table 7.2: Mass Changes for a Constant Speed System

In the case of the variable voltage variable frequency system significant mass penalties are introduced due to the need of dual controllers-converters. These are used to start the engine and control the motors for the operation of the environmental control system compressors. As a result the benefits of removing the current air starting system are limited and most of the savings relative to the constant speed drive system are due to the reduced size of the generators and the advantages of the variable power produced.

Equipment	All Electric Aircraft	More Electric Aircraft/Engine
ECS Removed		
Ducts, Precooler, Valves, air starting	-783 Kg	-783 Kg
ECS Added		
4 Compressors	+104 Kg	+104 Kg
4*155KVA Motors (0.3-0.5 Kg/KW)	+186 Kg (+310Kg)	+186 Kg (+310Kg)
4* 155 KVA Converters Controllers (0.68-1 Kg/KVA)	+421 Kg (+620 Kg)	+421 Kg (+620 Kg)
Power Generation Removed		
4*75 KVA IDG (0.5-1 kg/KW)	-150 Kg (-300Kg)	-150 Kg (-300Kg)
APU Gen 125KVA (0.27 Kg/KVA)	-32 Kg	-32 Kg
Power Generation Added		
4*200 KVA PMG (0.3 Kg/KVA)	+240 Kg	+240 Kg
2*180 KVA APU Generators	+97 Kg	+97 Kg
4*40 KVA 400Hz DC Link (0.68-1 Kg/KVA)	108 Kg +(160Kg)	108 Kg (+160Kg)
Elimination of Hydraulics		
Net Mass Saving	-750 Kg	
Electro Impulse De-Icing	-144 Kg	-144 Kg
4*60 KVA Electric Hydraulic Pumps		+72 Kg (+120 Kg)
4 * 50 KVA Electric Fuel Pumps	+60 Kg (+100 Kg)	+60 Kg (+100 Kg)
Overall Mass Changes	-643 Kg (-346Kg)	+179 Kg (+492 Kg)

Table 7.3: Mass Changes for a VVVF System

A comparison between the constant speed drive system and the hybrid variable speed constant frequency system reveals that although the mass of the former system is lower the total savings reduce and/or the total penalties increase. Once again this is due to the mass of the starting system. The findings in terms of system mass however are in good agreement with those presented in by Wilson [139]

It is noted that in all cases considered the APU generators have been sized to provide the power required for the environmental control system at sea level cold day operations. It is also assumed that the power output is sufficient to start the engines. If the APU generators are required to supply the power required for sea level hot day operations then the power output should be around 600 KW. This however is impractical and the system would require 3-4 generators driven by the APU.

Equipment	All Electric Aircraft	More Electric Aircraft/Engine
ECS Removed		
Ducts, Precooler, Valves, air starting	-783	-783
ECS Added		
4 Compressors	+104 Kg	+104 Kg
4*155KVA Motors (0.3-0.5 Kg/KW)	+186 Kg (+310Kg)	+186 Kg (+310Kg)
Power Generation Removed		
4*75 KVA IDG (0.5-1 kg/KW)	-150 Kg (-300Kg)	-150 Kg (-300Kg)
APU Gen 125KVA (0.27 Kg/KVA)	-32 Kg	-32 Kg
Power Generation Added		
4*250 KVA PMG (0.3 Kg/KVA)	+300 Kg	+300Kg
2*180 KVA APU Generators	+97 Kg	+97 Kg
4*200 KVA 400Hz DC Link (0.68-1 Kg/KVA)	+544 Kg (+800Kg)	
4*160 KVA 400 Hz DC Link		+435Kg (+640Kg)
Elimination of Hydraulics		
Net Mass Saving	-750 Kg	
Electro Impulse De-Icing	-144 Kg	-144 Kg
4*60 KVA Electric Hydraulic Pumps		+72 Kg (+120 Kg)
4 * 50 KVA Electric Fuel Pumps	+60 Kg (+100 Kg)	+60 Kg (+100 Kg)
Overall Mass Changes	-568 Kg (-298 Kg)	+145 Kg (+412Kg)

Table 7.4: Mass Changes for a Hybrid VSCF System

The systems discussed above assume the use of permanent magnet generators for both the variable voltage variable frequency and the hybrid system.

If switched reluctance generators are used, then the controllers need to operate from a direct current, preferably high voltage, supply. This type of power, i.e. 270 V DC, could be distributed to all electrical power equipment but alternations would be required in a number of devices. The environmental control system compressors could be powered from this power output and can incorporate either switched reluctance motors or permanent magnet brushless motors. In both cases separate motor controllers would be required. Fuel and hydraulic pumps could be driven directly from the 270 DC power supply. A simplified diagram of the system is shown in Figure 7.13. The mass changes for the A340-500 class aircraft are shown in Table 7.5

Equipment	All Electric Aircraft	More Electric Aircraft/Engine
ECS Removed		
Ducts, Precooler, Valves, air starting	-783 Kg	-783 Kg
ECS Added		
4 Compressors	+104 Kg	+104 Kg
4*155KVA Motors (0.4-0.5 Kg/KW)	+248 Kg (+310Kg)	+248 Kg (+310Kg)
4* 155 KVA Controllers		
(0.68 Kg/KVA)	+421 Kg	+421 Kg
Power Generation Removed		
4*75 KVA IDG (0.5-1 kg/KW)	-150 Kg (-300Kg)	-150 Kg (-300Kg)
APU Gen 125KVA (0.27 Kg/KVA)	-32 Kg	-32 Kg
Power Generation Added		
4*200 KVA PMG (0.35-0.45 Kg/KVA)	+280 Kg (360Kg)	+280 Kg (+360Kg)
2*180 KVA APU Generators	+126 Kg (+162Kg)	+126 Kg (+162Kg)
200 KVA Inverter (0.25 Kg/KVA)	+50 Kg	50 Kg (+160Kg)
Elimination of Hydraulics		
Net Mass Saving	-750 Kg	
Electro Impulse De-Icing	-144 Kg	-144 Kg
4*60 KVA Electric Hydraulic Pumps and Controllers		+259 Kg (+283 Kg)
4 * 50 KVA Electric Fuel Pumps and Controllers	+216Kg (+236 Kg)	+216Kg (+236 Kg)
Overall Mass Changes	-414 Kg (-416Kg)	+595 Kg (+777Kg)

Table 7.5: Mass Changes for Switched Reluctance Generators and 270V DC System

The main disadvantage of the 270 V DC power is that all the motors require a separate controller leading to increased mass penalties. Despite the mass penalties, however and the disadvantages of high voltage DC supply the switched reluctance machines are better suited to the all electric engine concept and can tolerate higher temperatures than permanent magnet machines. For this reasons they may be preferred if the generators are to be located within the core of the engine.

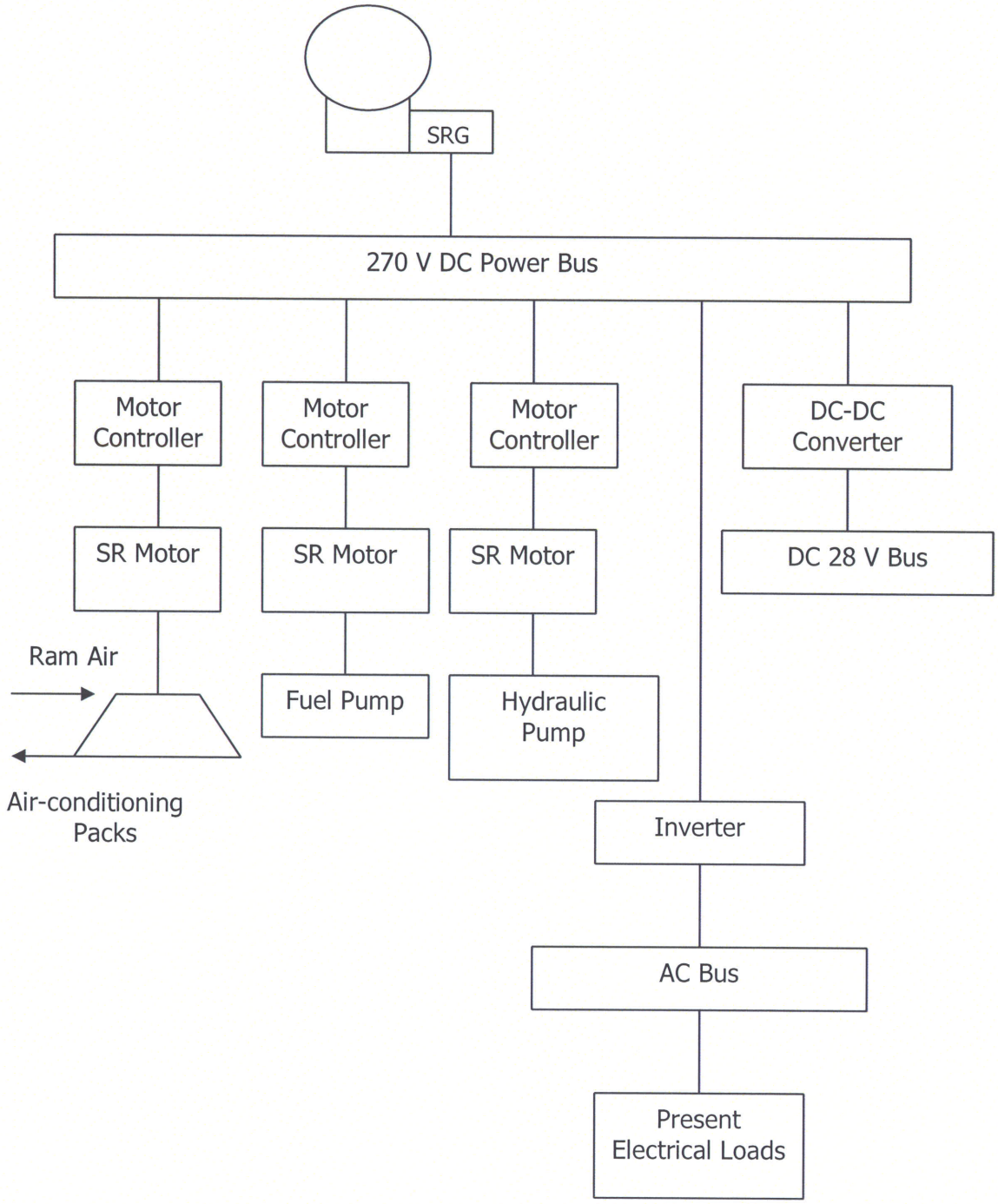


Figure 7.13: Simplified Diagram of 270 V DC System

The power requirements for the A330-200 and A380-100 class aircraft are shown in Table 7.6. Mass changes for these aircraft are summarised and presented in Table 7.7.

System	A330-200	A380-100
ECS	500 KW	1100 KW
Electrical System	150KW	450 KW
Hydraulic System	160 KW	240 KW
Fuel System	100 KW	200KW
Total	910 KW	1990 KW

Table 7.6: Power Requirements for A330-200 and A380-100 Class Aircraft

From the results it is noted that in all cases mass savings can only be obtained if the all electric engine concept is adopted and the hydraulic system is eliminated. Adoption of more electric technologies would increase the weight of the electrical equipment and as a result the net mass of the aircraft would increase. The mass penalties, however, depend largely on the configuration of the power generation and distribution system and the type of aircraft. In all cases examined use of integrated drive generators and constant speed drive systems led to higher mass penalties. This is mainly for two reasons. Firstly, such a system is a constant power system and as a result the generators must be oversized. Secondly large mass penalties are introduced due to the mass of the constant speed drive system.

System	A330-200		A380-100	
	AEA	MEE/A	AEA	MEE/A
IDG	-68 Kg (+544Kg)	+554 Kg (1214Kg)	+600Kg (+900 Kg)	+1720Kg (1920Kg)
VVVF	-407 Kg (-227Kg)	+165Kg (+323Kg)	-660Kg(-382 Kg)	+90Kg (+370Kg)
Hybrid (VSCF)	-203 Kg (212Kg)	+398Kg (+832 Kg)	-540 Kg (-262Kg)	+210Kg (+488Kg)
270 V DC	-70 Kg (-50Kg)	+555Kg (+600 Kg)	-319 Kg (21 Kg)	+ 690Kg(1257 Kg)

Table 7.7: Mass Changes for A330-200and A380-100 Class Aircraft

The penalties are more noticeable in the case of the A330-200 and A380-100 class of aircraft. Around fifty percent of electrical power is required for the operation of the environmental control system. Both the A330-200 and A380-100 class aircraft carry more passengers relative their maximum take-off weight. As a result the power requirements for the environmental control system are higher leading to larger generators and motors. As a result the mass penalties also increase. In addition, the A330-200 aircraft is a two engine aircraft. To achieve the required standards of safety the generators must be oversized to cater for an engine failure. In the case of the A380-100 class aircraft the conventional configuration assumes use of variable frequency generators and as a result the mass of the baseline generators removed is

considerably lower than the mass of integrated generators used on other aircraft. This effect leads to large mass penalties and is noted that such a system would impose considerable mass penalties even if hydraulics were to be replaced.

In all cases use of variable frequency variable voltage generators led to higher mass savings in the case of the all electric aircraft and lower mass penalties in the case of the more electric aircraft/engine concepts. This is mainly due to the fact that the power delivered by such a system increases with rotational speed and the generators can be sized accordingly for the descent case where the power requirements are not maximum. Although the rotational speed of the generator would be low at such conditions, the load requirements also reduce and as a result the generator can be smaller than in the case of a constant power system. In theory a hybrid variable speed constant frequency system shares the advantages of both variable voltage variable frequency and constant frequency systems. Therefore it is not necessary to incorporate controllers for the environmental control system. However, the mass of the DC link system would eliminate any possible mass savings. Once again the savings are lower and the penalties are increased due to the fact that the variable speed constant frequency system is a constant power system and the generators need to be oversized. In most cases use of switched reluctance generators and 270V systems led to higher mass penalties than the variable hybrid and variable voltage systems. The main disadvantage of the system is the mass of the motor controllers.

In general the mass savings were higher in the case of the A340-500 class aircraft. Similar mass savings were obtained in the case of the A380-100 class aircraft while savings reduced for the A330-200 class aircraft. The findings represent the combined effect of three factors. The type of aircraft considered, number of engines and the level of baseline technology. Mass savings reduced in the case of the A330-200 class aircraft because of the high power requirements for the environmental control system and the fact that because this is a twin-engine aircraft, several components should be oversized and duplicated to ensure the required levels of safety. The A380-100 class aircraft also presents large power requirements for the operation of the environmental control system but it is a four engine aircraft and the components do not need to be oversized or duplicated. Finally the aircraft already incorporates a variable frequency system and as a result the baseline system can be easily converted to a variable voltage variable frequency system. This fact counteracts to some extent the detrimental effects of increased power requirements and as a result savings and/or penalties become similar to the A340-500 class aircraft. Possible system architectures and the implications of number of engines are discussed in the next chapter of this work.

Chapter 8: Discussion

8.1 Introduction

Implementation of all or more electric technologies has the potential of improving the performance of a given aircraft. Savings can manifest through improved engine performance, reduced equipment weight and improved secondary power and systems utilization. Despite the potential benefits and the amount of research invested the concept has not been implemented yet and its impact on the overall performance of the aircraft remains unclear.

Adoption of a single form of secondary power and implementation of more or all electric technologies is claimed to offer a number of distinct advantages over the conventional secondary power systems. These claims include the following:

1. Improved engine performance through the optimization of off-takes and elimination of bleed air. These issues is said to become critical as the bypass ratio of the engine increases and the size of the core reduces. However, despite concerns over the capability of high bypass ratio to provide secondary power requirements of future aircraft the claims remain unproven and it is generally believed that off-take penalties are not dependent on the configuration of the engine and can be kept to a minimum if the engine is designed to provide them.

2. Improved system utilization and implementation of more efficient power units. Conventional secondary power systems developed during the years that fuel was relatively inexpensive and have not been optimized for maximum efficiency. Bleeding air from the engine is largely inefficient and a large amount of power is wasted for no useful purpose. In addition, hydraulics are often sized for loads that are small in duration thus increasing the mass and power requirements of the system.

3. Elimination of conventional hydraulic and bleed air systems could lead to a significant mass reduction. However, the mass and size of the electrical equipment would also increase and these changes must be accounted for.

4. Additional benefits include reduced single type maintenance, improved reliability and life cycle costs.

Since a number of technical challenges need to be solved an accurate assessment of the benefits is difficult and prevents the application of such systems. Previous studies performed by NASA, Boeing and Lockhead highlighted the potential benefits but underestimated the difficulties and associated penalties. A more careful examination revealed that the benefits claimed were not justified and as a result they still remain unclear. Subsequent studies on the subject resulted in moderate savings, both in terms of fuel and overall mass reductions, which did not justify a change in the form that secondary power is extracted and distributed around the aircraft. Despite these findings and the uncertainties involved an active interest on the AEA systems has been maintained with subsequent studies concentrating on the development of individual and enabling technologies such as advanced generation systems, power distribution and control, electric actuation, etc.

The present study focused on assessing the potential benefits of adopting the all or more electric aircraft concepts in the case of high capacity long-range aircraft. Previous studies concentrated on low to medium range aircraft powered by medium bypass bass ratio engines. As a result the moderate power requirements and the more important weight considerations, for the class of aircraft reviewed, reduced the potential benefits and led to retrofitted designs with low savings. It is thought that high capacity long range aircraft as those examined by the present study are more suited to the all electric concepts and could benefit more.

The study was aimed to address the first three claims made and also provide a better understanding of the issues involved. As a result the following topics have been considered.

1. Future trends in aircraft and engine design. Since previous studies on the subject of the AEA concept concluded that the performance benefits alone would not justify a change in the pattern of SPS it was decided to review the general trends and requirements of future aircraft and determine how the All/More Electric systems fit with the overall design approach. The findings suggest that despite the advances in all disciplines involved in aircraft design, i.e propulsion, aerodynamics, structures, etc., performance improvements are approaching their limits and new practices to enhance the performance of a given aircraft must be sought. In the case of engine efficiency limitations in maximum attainable turbine entry temperature, compressor pressure ratio and environmental concerns limit the potential of increasing the already high thermal efficiency. Propulsive efficiency, on the other hand is constrained by mass drag and integration implications associated with increased bypass ratio.

Environmental concerns may also limit the increase in bypass ratio and may drive the design of the engine away from its optimum for minimum sfc. In theory, adoption of all electric technologies and elimination of bleed air could alleviate or even eliminate some of these problems. Current design practices and adoption of evolutionary rather than revolutionary concepts also limit potential improvements in aircraft design. In the case of large aircraft, the aspect ratio of the wing is limited by the 80 meters box constrain and this in turn limits reductions in induced drag. Incremental improvements in structural and aerodynamic efficiency may be overshadowed by the increased weight of the propulsion system resulting from increased bypass ratio and reduced specific thrust. Noise restrictions may also require an increase in bypass ratio for a given take-off thrust output. Obviously there is a conflict between increased bypass ratio and its effect on sfc, turbine entry temperature and NOx emissions, mass and noise requirements. Extraction of bleed air and its inefficiencies can only aggravate these problems. For a fixed thrust output elimination of bleed air and optimization of off-takes could reduce the size and mass of the engine, turbine entry temperature and finally sfc. Elimination of bleed ducts and associated equipment would minimize pressure losses within the bypass duct and allow for the design of a smaller, lighter and aerodynamically efficient nacelle. Adoption of all electric technologies could reduce the operating empty weight of the aircraft. This would have a snowball effect and could lead to a reduction in fuel load, thrust required (hence mass, size and drag of the engine), maximum take-off weight, etc. Reductions in maximum take-off weight could reduce the wing area, which in turn could allow for reduced wing span and increased aspect ratio, thus improving the aerodynamic efficiency of the aircraft. Reductions in thrust requirements and/or turbine entry temperature could reduce jet velocity and noise as well as NOx emissions. Finally reduced sfc, aircraft mass and improved aerodynamic efficiency would lead to reduced fuel consumption thus reducing CO_2 emissions and aircraft fuel costs. Therefore, even if the direct benefits of adopting the all electric concepts are not large enough to justify such a change the concept should be treated as an enabling technology which could increase the design envelope of present aircraft and engines and provide increased design flexibility to both engine and aircraft designers/manufacturers. Innovative designs and advanced concepts also favor adoption of all or more electric technologies and elimination of bleed air in particular. Implementation of laminar flow control, for example of the wing of a given aircraft could reduce cruise drag and therefore cruise thrust requirements by 10%. However, such a system would increase the power off-takes from the engine and would have a detrimental effect on its

efficiency. Although drag reduction will dominate leading to increased fuel savings, overall efficiency could be further improved. A laminar flow control system would have an effect at cruise conditions and take-off thrust requirements are likely to remain unchanged. As a result the engine would be designed for either take-off conditions or/and top of climb requirements. Such a case would impose additional difficulties in matching the performance of the engine for minimum sfc at the cruise section of the mission profile. Increased off-takes would aggravate these effects. In the case of the blended wing body the increased passenger capacity and aerodynamic efficiency of the aircraft coupled with the reduced number of engines (most designs propose use of three engines, would impose additional off-take requirements that would have a detrimental effect on the performance of the engine. Once again it is more likely that the improved aerodynamic efficiency and mass of such an aircraft would predominate but nonetheless implementation of all electric technologies could improve the overall performance of the aircraft and could contribute in sharing and/or minimizing life cycle costs.

2. Effects of off-takes on engine design and performance. It is well known that off-takes impose a significant penalty on the performance of aero-engines. There has been a lot of debate in the open literature regarding the penalties imposed by extraction of secondary power and whether these penalties increase with bypass ratio and affect the design of the engine. For these reasons and since a large proportion of the benefits offered by the all electric aircraft concept is due to the elimination of bleed air a detailed analysis concentrating on the effect of off takes was undertaken. Although the detrimental effects of extracting secondary power from the engines are widely recognized, no such work has been found in the open literature that addresses the exact implications involved in either a qualitative or quantitative manner. This was the first time that such an analysis was undertaken and considered the effect of engine design and parameters on penalties associated with extraction of off-takes. The analysis resulted into two methods of evaluating engine performance penalties. The first is based on the percentage of bleed air or shaft power extracted. The second considers the extraction of shaft power alone. Both methods can be used at the initial design stages to assess the performance penalties of the engine operating with extraction of off-takes.

3. Consideration of all electric concepts in the case of military aircraft.

4. Aircraft/engine performance. The effects of conventional off-takes on the performance of both the engine and the aircraft have been investigated. All electric variants have been derived and considered whilst comparisons were made against the conventional baseline aircraft. The effects of the all electric aircraft concept on the mass and performance of the aircraft have been assessed. Figures of merit included engine sfc, turbine entry temperature, overall fuel usage and changes in aircraft mass. Such a rigorous analysis helped to quantify the benefits claimed in the past and by other studies concerned with the development and design of electric technologies. The work forms the basis for more detailed studies, which should consider all the implications involved in adopting the all electric aircraft/engine concepts, i.e. cost, reliability and tradeoff studies. A sensitivity analysis is also included to allow for future developments and different design approaches as well as to minimize the uncertainties and the simplifications of the present study. Since the origins of the all electric aircraft and the initial studies performed during the mid and late eighties, no such work has been published in the open literature that quantifies the benefits of adopting all or more electric technologies on an aircraft engine level. Other studies and research projects on the subject emphasised the fact that advances in a detailed level and enabling technologies should be incorporated into a detailed study covering the overall performance of the aircraft. This, however, is a huge and lengthy task that requires a lot of unavailable information and most of all, expertise from a very diverge background. The present work forms the basis of such an analysis and provides some indications and preliminary results. Implementation of all electric technologies would require close and strong collaboration between the aircraft, engine and systems engineers not always necessary in the design process of present conventional aircraft. It is the purpose of this work to identify such areas of work and provide a common background for all disciplines required to be involved. Although the work presented relies on a number of assumptions and simplifications it provides an independent and unbiased view on the prospects of all or more electric technologies and presents a method for undertaking the task of assessing the potential benefits and implications involved from data available in the open literature. Finally the impact of aircraft and engine design parameters on the benefits/implications of the all electric aircraft concept is also considered and presented. Once again this is the first time that such issues have been reported in the open literature.

5. At the start of the study the claim was made that the configuration and design parameters of the aircraft/engine system would have an effect on the impact of

all electric technologies. This claim has been justified and it is shown that the configuration of the aircraft would have a multiple effect on both the performance of the all-electric aircraft, its systems architecture and design point as well as the overall mass changes. The main parameters that should be considered are number of engines, bypass ratio, engine pressure ratio and turbine entry temperature, number of passengers, range and finally size of the aircraft. It is shown that these parameters could have an opposing effect on performance of the engine and the aircraft and mass changes.

6. Consideration of present secondary power systems and their limitations. The configuration of present secondary power systems has been studied and presented along their limitations. Alternative systems were also reviewed. Particular emphasis was placed on the inefficiencies and limitations presented by the present bleed and environmental control system. Bleeding air from the engines is highly inefficient and has a detrimental effect on the performance of the aircraft without any real benefits. Bleed air is usually over compressed for no useful purpose. At most operating conditions the bleed air is extracted at pressures and temperatures higher than those needed. The position of bleed air ports are dictated by the requirements of the bleed air thermal anti-icing system and as a result twice the power required is extracted. As a result, the pressure and temperature of the bleed air must be reduced before entering the airframe and thus further performance, mass and aerodynamic penalties are introduced. These penalties have a more pronounced effect in the case of bypass ratio engine and as a result may prevent their application, at least that is in some particular applications such as twin engine aircraft. Inefficiencies of current bleed air systems can be improved if the energy content of the air extracted could be used in a more efficient manner. However, since bleed air is presently used for both air-conditioning/pressurization and ice protection functions both the pressure and thermal energy content are required, at different operating conditions. Over the years, various systems have been proposed for alleviating the problem, all of them present additional difficulties and they merely address the problem. Elimination of the bleed air ice protection system could improve the efficiency of the system and the performance of the propulsion system, since air for air-conditioning and pressurization would be extracted at the required pressure. This could half the levels of power currently extracted but a more efficient method, in terms of engine performance, would be the use of separate driven compressors. Analysis of off-takes penalties confirmed that for the same amount of bleed and shaft

power, turbine entry temperature, sfc and thrust penalties are higher in the case of bleed air extraction.

7. Possible systems architectures suitable for the all or more electric aircraft have been reviewed. The requirements for each system have been considered separately and different architectures have been presented. Possible electrical system architectures were then presented. The implications of component requirements and design have been reviewed and overall conclusions were drawn. Adoption of all or more electric engine technologies could improve system utilization and use of a smaller and more efficient power unit. This is true particular in the case of a variable voltage variable frequency system. In such a case the power generation and distribution system can be designed to produce the power required at each flight phase of the mission profile without the need to oversize and/or overload the generators. The design point in such a case would depend on the configuration of the aircraft and the number of engines. Adoption of all the all-electric aircraft or all electric engine would also have an influence on the configuration and overall architecture of the system. In this case additional factors such as system generator suitability and safety would also affect the final decision.

8.2 Effects of Off-takes on Engine Performance

A detailed analysis focusing on the effects of off-takes on the performance of the engine has been performed. It is claimed that most of the benefits of the all-electric aircraft concept would result from the elimination of bleed air and extraction of shaft power. As a result, analysis of off-takes penalties is an important section of this study. Both, effects of bleed air and shaft power off-takes have been considered. Although both forms of off-takes have similar effects on the performance of the engine and both of them are supplied by the hot section of the engine their magnitude vary considerably and they present some distinct differences. For these reasons, they should be treated differently.

Extraction of secondary power would reduce the power output of the engine in terms of thrust and its efficiency. As a result, both sfc and turbine entry temperature would increase. Shaft power and bleed air extraction would also affect the characteristics and the operation of the compressor. In the case of shaft power extraction, the compressor running line would move towards surge thus increasing pressure ratio. Bleed air on the other hand has the opposite effect and the running line would move away from surge

reducing compressor pressure ratio. The movement of the running line would affect the efficiency of the compressor hence increasing performance penalties.

The operating characteristics and efficiency of other components may be also affected, though, their effect on engine performance is usually less important. Extraction of secondary power was studied under different engine modes of operations. Both constant amount and percentage off-takes were considered. It was thought that extraction of constant percentage of bleed air and/or shaft power would isolate the effect of engine parameters, while extraction of constant amount of power could account for off-take size.

Reductions in the TET and/or TET to compressor inlet temperature ratio at given flight conditions increased the adverse effects of bleed and power extraction. As far as changes in altitude are concerned the analysis showed that for a fixed percentage of off-take the thrust sfc penalties reduced with altitude. Penalties remained constant for altitudes higher than 11.5 Km. Extraction of constant amount of shaft power led to increased engine performance penalties with altitude. This effect is attributed to the reduced density of the air with increasing altitude. Finally, for fixed percentage of off-takes, the detrimental effects on thrust and sfc increased with Mach number. In most cases extraction of constant amount of off-take led to lower engine performance penalties with Mach number.

A more detailed analysis was undertaken and concentrated on the effects of engine overall configuration and performance penalties associated with extraction of secondary power. The aim of the analysis was to isolate the individual effects of engine parameters and to develop a general method for estimating the performance penalties of different off-takes on various engine configurations. Two different approaches were possible and both of them are presented and discussed in turn in the following paragraphs. The first method deals with the effects of fixed percentages of off-takes on thrust loss, sfc, TET and compressor characteristics. The method is derived by expressing the performance of the engine and its matching procedure in a non-dimensional form. The second approach is based on a simplified form of the pi-theorem and examines the effects of different magnitudes of shaft power off-takes on various engine configurations. Both constant TET and thrust conditions were examined. The method has been derived for turbojet and turbofan engines.

In the case of constant TET conditions, thrust penalties depend on compressor pressure ratio and percentage of off-take relative to the power recirculated into the core of the engine. Movement of compressor operating line with extraction of either

bleed air or shaft power was established. Specific fuel consumption penalties found to be a function of engine pressure ratio, turbine entry temperature and percentage of off-take relative to the power recirculated in the core. Finally thrust penalties depend on the percentage of off-take and compressor pressure ratio. For fixed percentage of off-take, engine performance penalties are not dependent on bypass ratio.

For constant thrust operations, sfc penalties depend on compressor pressure ratio and TET. Penalties increase with pressure ratio and reduce with TET. TET penalties depend on the percentage of off-take alone. Movement of compressor running line was also established for constant thrust operations. Specific fuel consumption penalties were expressed by the following relationship relating the fuel to air ratio of the clean engine to that of the engine operating with extraction of off-takes.

$$\frac{sfc_{BLEED}}{sfc_{CLEAN}} = \frac{(T_4 - T_3)_B}{(T_4 - T_3)}$$

Although there have been a number of studies on the subject of off-takes none of them examined the effects of engine parameters on performance penalties. Furthermore, only semi-empirical relationships are provided for estimating sfc penalties. The accuracy and application of such techniques, however, is limited and does not consider all the parameters affecting the performance of the engine. A comprehensive literature review revealed that there are not such relationships for modelling the thrust and TET penalties associated with extraction of bleed air and/ shaft power. Furthermore, the effect of off-takes on the operating line is treated qualitatively. It is worth mentioning that all the references obtained on the subject of off-takes suggest that performance penalties depend only on the percentage of the off-take relative to the power of the engine. However, different opinions have been expressed in a number of references regarding the percentage of an off take and its effect on the performance of the engine. For example, Jones [58] suggests that off-takes should be related to the power recirculated within the core of the engine and thus for the same amount of off-take engine performance penalties would increase with bypass ratio. Ruffles[19], however, argues that an off-take should be related to the total power, hence thrust, delivered by the engine and not the core power. This view is also supported by Peackock[53] who claims that there is nothing fundamental on why high bypass ratio engines should suffer greater sfc penalties provided that the engine has been designed to provide them.

On the contrary, the present analysis suggests that the off-take should be considered relative to the core power. Moreover, although the percentage of bleed air is usually the dominant factor, compressor pressure ratio and TET have an important second order effect. Indeed, in the case of constant thrust operation, the effect of engine parameters may be more pronounced than that of the size of the off-take. The latter is shown in figure 8.1 where sfc penalties derived from the analysis have been expressed as a function of fuel to air ratio and percentage of bleed air. In each case, TET was varied from 1000 K to 1800 K and the pressure ratio of the compressor was varied from 10:1 to 50:1

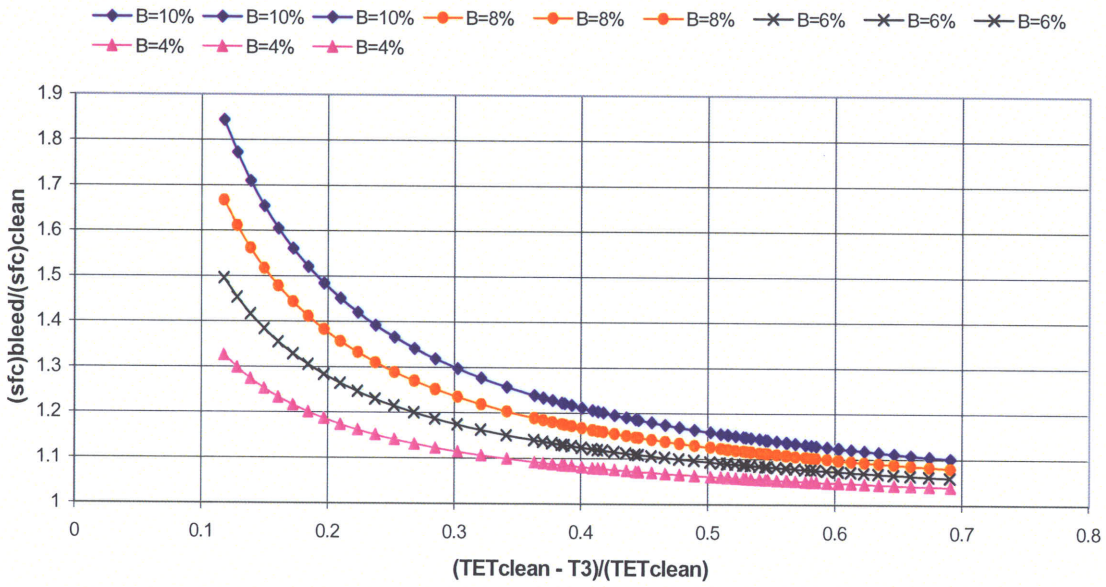


Figure 8.1:Effect of Engine Parameters and Off-take Size on sfc Penalties

The accuracy of the method was validated against Turbomatch results for a two spool turbofan engine operating at 10000 meters and 0.8 Mach. The pressure ratio of the engine at the design point was set at 30:1, TET at 1500 and the bypass ratio at 10:1. TET and sfc penalties are shown for bleed air percentages varying from 1%-10%. The results are shown in Figure 8.2.

The findings suggest that there is a strong influence of compressor pressure ratio and TET on performance penalties associated with extraction of off-takes. This is important since the effects of off-takes can be considered at the early stages of the design process of the engine and pressure ratio bypass ration and TET can be selected accordingly for minimum sfc with the off-takes. The effect of these parameters has been overlooked by the references mentioned above and only the size of the off-take was considered.

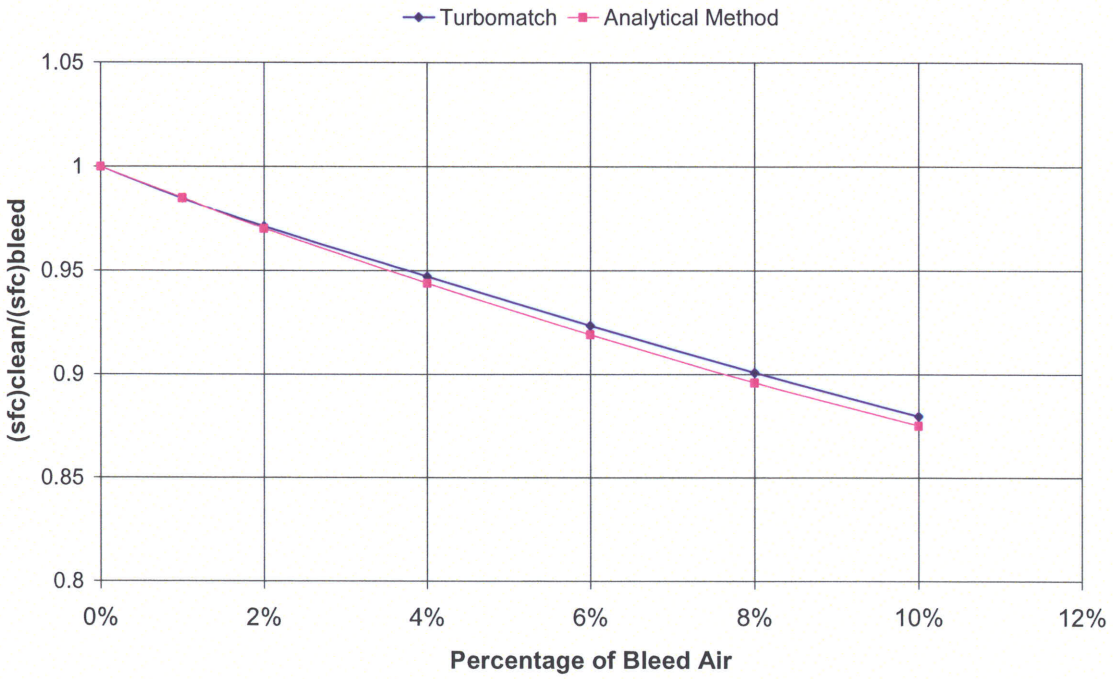


Figure 8.2: sfc Penalties

The second method relied on deriving a non-dimensional factor that would contain all the parameters affecting engine performance penalties with extraction of shaft power. In this case, the power off-take was expressed as a function of the energy flow rate supplied to the engine with the addition of fuel. The factor is shown below:

$$\beta = \frac{Fuel\ \ Flow \times ST \times PR}{\frac{Shaft\ \ Power}{P_1\sqrt{T_1}} \times \sqrt{TET}} \quad or \quad \beta = \frac{(TET - T_3) \times PR \times \frac{Thrust}{(1+BPR)}}{Shaft\ \ Power \times \sqrt{TET}}$$

Figure 8.3 plots the sfc ratio against the quasi-dimensional relationship for several engines operating at various conditions and with extraction of different amounts of shaft power off-takes.

Three areas of interest are identified. At relatively high values of power factor, i.e. low off-take and bypass ratio and/or high TET, pressure ratio and thrust, sfc penalties increase linearly and remain relatively low. Continuous reduction of power factor causes an adverse, non-linear sfc penalty. Further reduction of power factor causes penalties to increase rapidly.

Peackock [53] claimed that future turbofan engines would experience a 5% sfc penalty due to increased off-takes. Peackock[53] also mentioned that penalties could be reduced to 2% by optimising the method of extracting secondary power. From Figure 8.4, it is seen that these sfc penalties correspond to low power factor and lie in *design areas* where sfc penalties could increase rapidly.

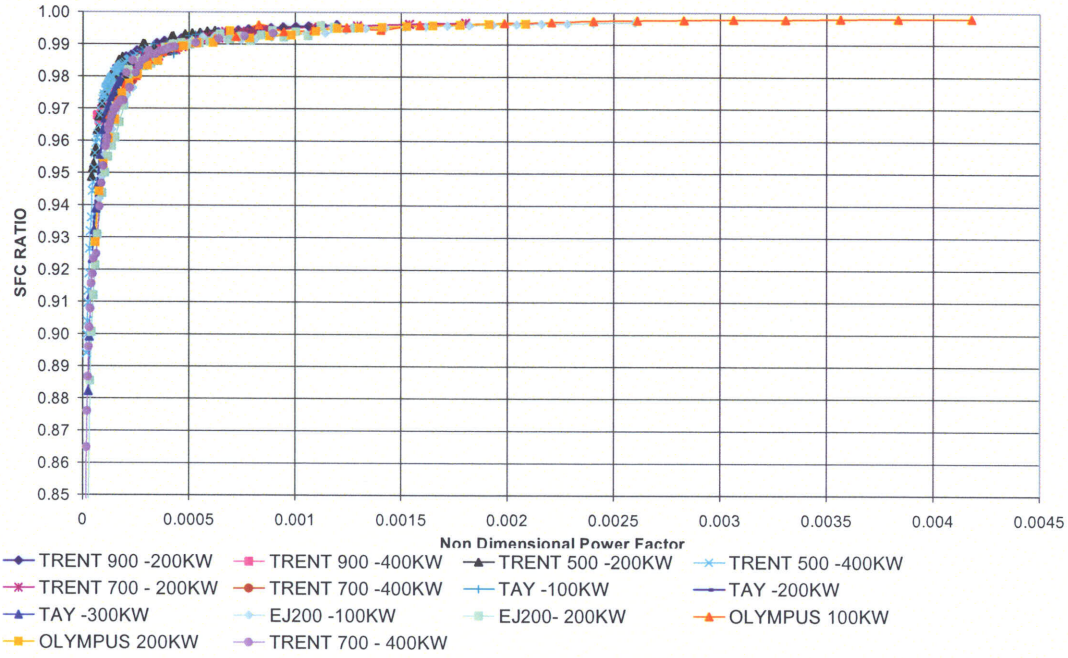


Figure 8.3: sfc ratio Vs Noni-Dimensional Power Factor

The factor derived contains all the parameters used in the engine design point map and therefore can be used at the early stages of the design process to optimize for minimum sfc with shaft power off-takes. The figure derived suggests that future engines may experience high sfc penalties due to the extraction of off-takes. Future engines are likely to have increased bypass ratio and as a result for constant thrust, TET and shaft power the non-dimensional power factor would reduce due to the lower specific thrust and fuel flow. As explained before current designs are already in the region of high penalties and further reduction of these two parameters could impose increased sfc penalties. Increase in bypass ratio is likely to be accompanied by an increase in TET to compensate the reduction in jet velocity and thermal efficiency. It was mentioned before that increased TET has a positive effect of performance penalties with extraction of off-takes. In this case however, increase in TET would result in a smaller core and reduced fuel flow. The effect of a smaller core due to the increase in TET is included in the non-dimensional factor by the square root term in the denominator.

8.3 All Electric Aircraft Performance Models

To be able to study the effect of all electric concepts on the baseline aircraft an aircraft performance model was developed. The model was used to compute the aerodynamic characteristic of the aircraft based on geometric details found in the open literature. Typical flight profiles were considered. These included the range covered during a diversion and the modeling of contingency fuel as required by the appropriate regulating authorities such as the FAA and the JAA. Although the main impact of the all electric concepts would be elimination of bleed air at the cruise section of the mission profile, fuel required for diversion must be loaded on the aircraft and is carried throughout the mission. Any reduction in mass could therefore have a significant effect on the overall performance of the aircraft. In addition, the general case of an accelerated climb and stepped cruise were considered. The initial cruise altitude and subsequent changes were allowed to vary for each aircraft to achieve maximum specific range and aerodynamic efficiency.

The accuracy of the model was validated against published data by using the payload range diagrams. The range computed was within 1% of the range given by the aircraft manufacturer. The model was integrated with an existing engine performance model developed at Cranfield University. Engine modeling allowed complete freedom to change the off-takes as required for the purpose of the study. The tool developed was vital in determining the impact of all electric concepts since it allowed for the modeling of different off-takes and changes in aircraft mass. The effect of engine parameters and aircraft type and configuration were also studied. The range of the aircraft was computed through an iterative process and use of a variation of the integrated range method.

Information on the system specifications and power requirements for the three aircraft modeled were collect from the open literature. Unavailable data were scaled from published data on existing aircraft. Airworthiness regulation and published data were also consulted in determining the mass flow rates for the operation of the environmental control system. Shaft power off-takes were determined by considering the range and number of passengers for each aircraft and scaling published data. Three generic scaling methods were used to validate the power requirements derived.

The computation allowed variations in TET and other aircraft/engine parameters to be recorded at the various mission phases with various off-takes. This is a significant feature of the engine performance model, which allowed for a more detailed and in depth analysis on the effects of off-takes.

The performance of each aircraft at the cruise section was expressed in terms of aerodynamic efficiency and engine sfc. Reasonable variations in both parameters were obtained increasing the confidence on the tool. The next step involved the analysis of the baseline aircraft performance and the effect of conventional secondary power systems. In the case of the A340-500 class aircraft, extraction of secondary power in the conventional form introduced an average sfc penalty of 2-3% relative to the clean case. The values are in good agreement with those presented by Peackock[53]. It is noted that performance penalties increased with time and altitude. As the aircraft spends time at cruise the engine power setting is reduced to compensate for the reduction in fuel and aircraft mass. As a result, the mass flow through the engine reduces and a constant amount of off-take implies increased sfc penalties. When the aircraft climbs to a higher altitude the density of the air reduces and once again the mass flow through the engine reduces leading to increase sfc penalties. TET penalties also increased with time and altitude.

The analysis revealed that the power extracted in the form of bleed air for the operation of the environmental control system was twice the power actually required. Therefore, performance penalties could be optimized by introducing a third bleed air port in the core of the engine that would supply air at temperatures and pressures required at the cruise section. However, a more efficient method of extracting the power required would be in the form of shaft power that could be used to drive separate compressors. Despite the inefficiencies of such a system the fact that extraction of shaft power introduces lower sfc and TET penalties than equivalent amounts of bleed air power led to lower engine performance penalties. Optimisation of off-takes and elimination of bleed air in particular led to a 15-20 degrees reduction in engine TET. Reductions in TET increased with altitude and time.

8.4 Sensitivity Studies

To allow for future developments in electrical technologies and also account for the simplifications made in this work, a sensitivity analysis has been included. So called carpet plots have been produced for the three aircraft examined. Previous studies noted that findings were not particularly sensitive to changes in mass but the uncertainty and fuel savings found to be sensitive to the amount of bleed air replaced. For this reason both effects have been considered and are included in the results. The findings are shown in Figures 8.4-8.6.

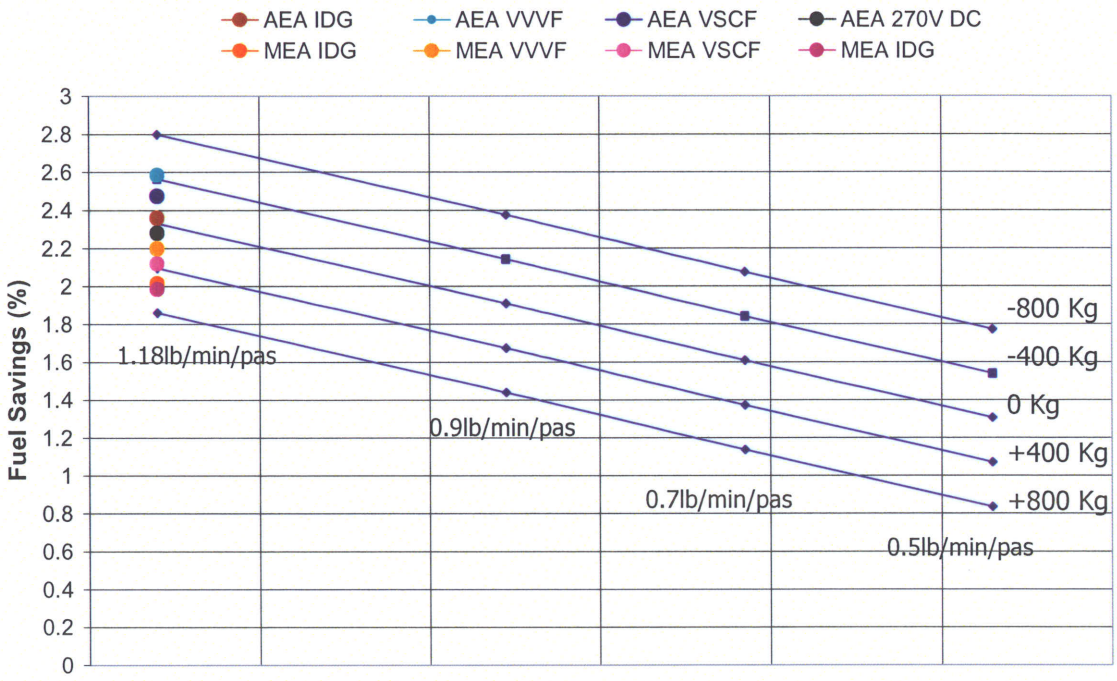


Figure 8.4: Sensitivity Analysis for A330-200 Class Aircraft

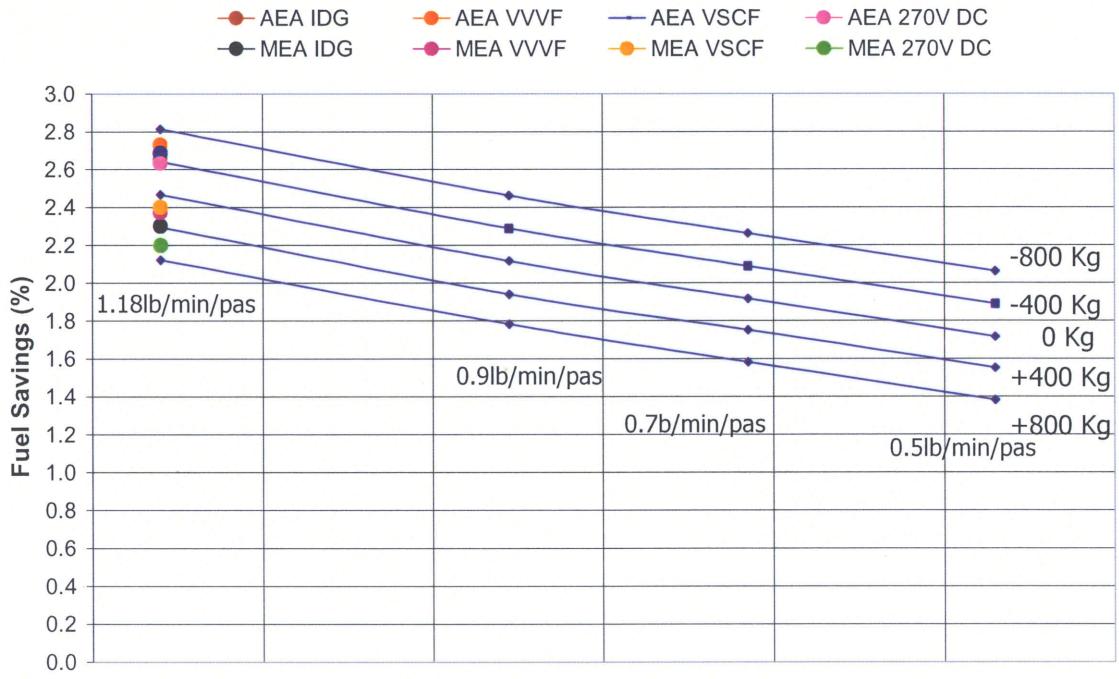


Figure 8.5: Sensitivity Analysis for A340-500 Class Aircraft

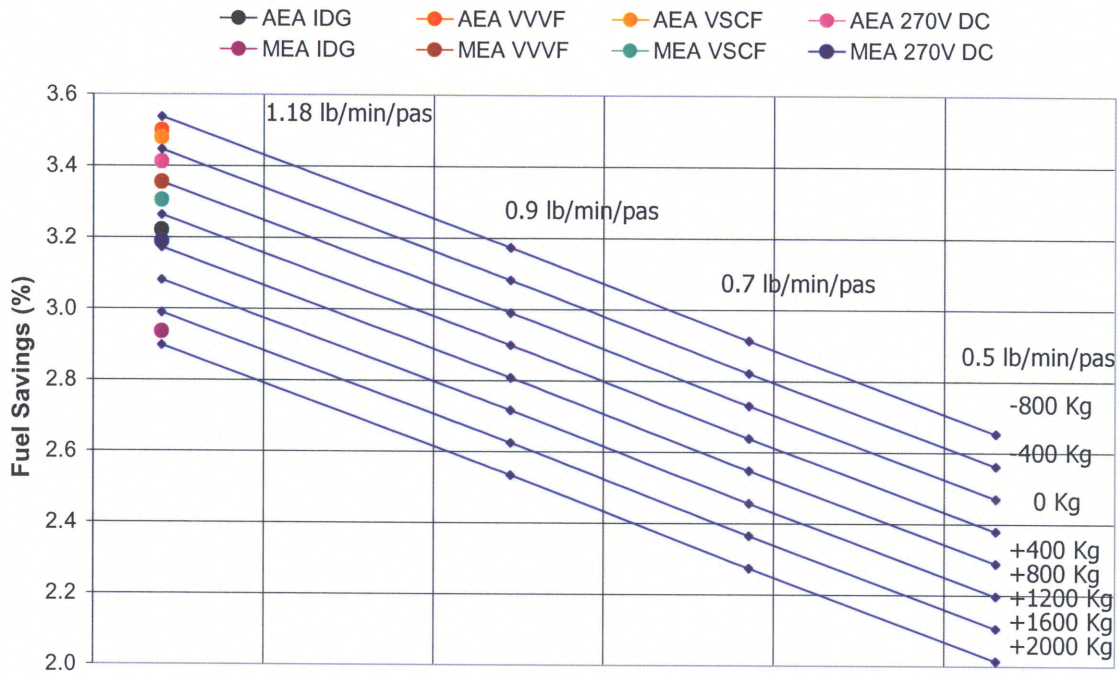


Figure 8.6: Sensitivity Analysis for A380-100 Class Aircraft

The amount of bleed air extracted for each aircraft was based on published data on the A330-200 class aircraft. The bleed air corresponds to 1.18 lb/minute/passenger. This is considerably higher than the bleed air requirements defined by the airworthiness authorities. However, because of the large number of passengers and the long cruise of all three aircraft these values considered to be appropriate for maintaining a comfortable cabin environment. Future advances, in the design of the environmental control system may reduce the bleed airflow rates and for this reason the amount of bleed air replaced has been reduced to 0.5 lb/min/passenger. It is seen that as the amount of bleed air replaced is reduced fuel savings also reduce. In reality the mass of the generators, motors, controllers and compressors would also reduce but the effect of power extraction on fuel savings is more pronounced and the end result would be a reduction in potential savings.

It is also noted that the A330-200 and A380-100 class aircraft are more sensitive to the amount of bleed air replaced. This is mainly due to the high amount of bleed air currently extracted and the increased sfc penalties of the conventional bleed air system, as discussed in Chapter 5. It is also noted that in the case of the A380-100 class aircraft the reduction in potential savings with replacement of bleed air is counteracted to some degree by the increased range of the aircraft and as a result replacement of lower bleed airflows would result to higher saving than in the case of the A330-200 class aircraft. The A340-500 seems to be less sensitive to the amount of

bleed air replaced. This is due to the long range of the aircraft which counteracts to a larger extent the reduction of bleed airflows replaced and therefore the reduction in fuel savings are lower than in the case of the two other aircraft.

In terms of mass penalties, the A330-200 class aircraft found to be more sensitive. This is because the mass changes considered constitute a larger percentage of the maximum take-off weight and more importantly the operating empty weight of the aircraft. From the three aircraft modelled the A380-100 was the less sensitive in terms of mass changes.

8.5 Effect of More Electric Technologies on Aircraft-Engine Design and Resizing Potential

Adoption of all or more electric technologies could have an impact on the design of the aircraft and the engine. As a result current limitations imposed on the deign envelop could be alleviated leading to increased overall savings. Examination of the results and the parameters of the three aircraft considered identified two cases where adoption of more electric technologies could have an impact on the overall configuration and the design of the aircraft/engine system. The first case refers to the A380-100 class aircraft and the limitations imposed on the current design by the 80-meter box constrain. Existing regulations limit the span of the aircraft to a maximum of 80 meters and as a result the aspect ratio of the wing is reduced to a value around 8.2. Elimination of bleed air could reduce the fuel load and maximum take-off weight by 7670 Kg. Adoption of all electric technologies could lead to an overall mass reduction of 8600 Kg. Assuming that the wing loading would remain the same then the wing area could reduce. The wing area of the A380-100 class aircraft is around 780m^2. Implementation of all electric technologies could lead to a wing area of 768m^2. Keeping the span of the aircraft at 80 m, then the resulting aspect ratio is 8.33.

The mass of the wing was approximated using the following equation found in Jenkinson [127]:

$$M_W = 0.021265*(MTOM*NUTL)^{0.4843}SREF^{0.7819}ARW^{0.993}(1+TRW)^{0.4}(1-R/MTOM)^{0.4}/(WSWEEP\ TCW^{0.4})$$

Where:
MTOM = aircraft maximum take-off mass (kg)
NULT = aircraft maximum load factor (typically 3.75)
SREF = wing area (m^2)
ARW = wing aspect ratio
TRW = wing taper ratio

TCW = wing average thickness to chord ratio
WSWEEP = wing quarter chord sweepback angle (degrees)

R is the effect of inertia relief to the wing root bending moment given by

$$R= (Mw + Mf + ((2 Meng Bie)/0.4B)+((2*Meng*Boe)/0.4B))$$

Mw = Wing mass estimated from previous equation and iterated (kg)
Mf = fuel load (kg)
Meng = mass of individual engine (kg)
Bie = distance between inboard engines (m)
Boe = distance between outboard engines
B = wing span

The mass of each engine was calculated using the following expression found in the same reference:

$$Meng= 1.43 T (8.7+(1.14BPR)$$

where:
T = sea level static thrust (KN)
BPR = engine bypass ratio

The mass of the baseline and new wing were found to be almost the same. This is because the mass of the wing with increased aspect ratio increases due to the reduced bending moment relief (resulting from the reduced fuel load) and the increase in aspect ratio. The increase in mass is counteracted by the reduction in wing area and MTOM. Using the new values for wing area and aspect ratio the total drag coefficient was calculated. Figure 8.7 shows a plot of the aerodynamic efficiency of both the baseline and all electric aircraft with increased aspect ratio. It is seen that adoption of all electric technologies and increased aspect ratio led to improved aerodynamic efficiency.

Overall fuel savings increased by 2000 kg, leading to a total reduction of 10324 Kg. The figure represents a 4.6% reduction in total fuel load compared to the baseline aircraft. The maximum take-off mass of the aircraft could reduce by 1.96%. The reduction in maximum take-off mass could be used to resize and re-optimise the aircraft leading to further reductions in maximum take-off mass, take-off thrust

requirements and engine mass and finally aircraft drag. The findings are summarized in Figures 8.8 and 8.9.

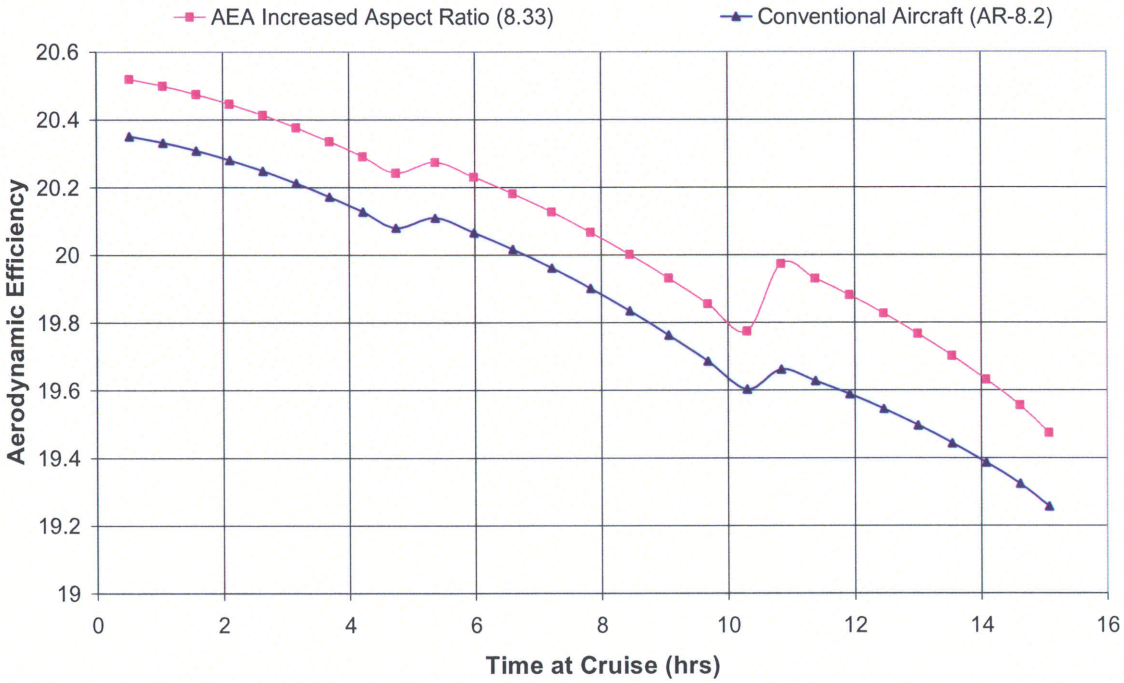


Figure 8.7: Aerodynamic Efficiency at Cruise for Resized A380-100 Class Aircraft

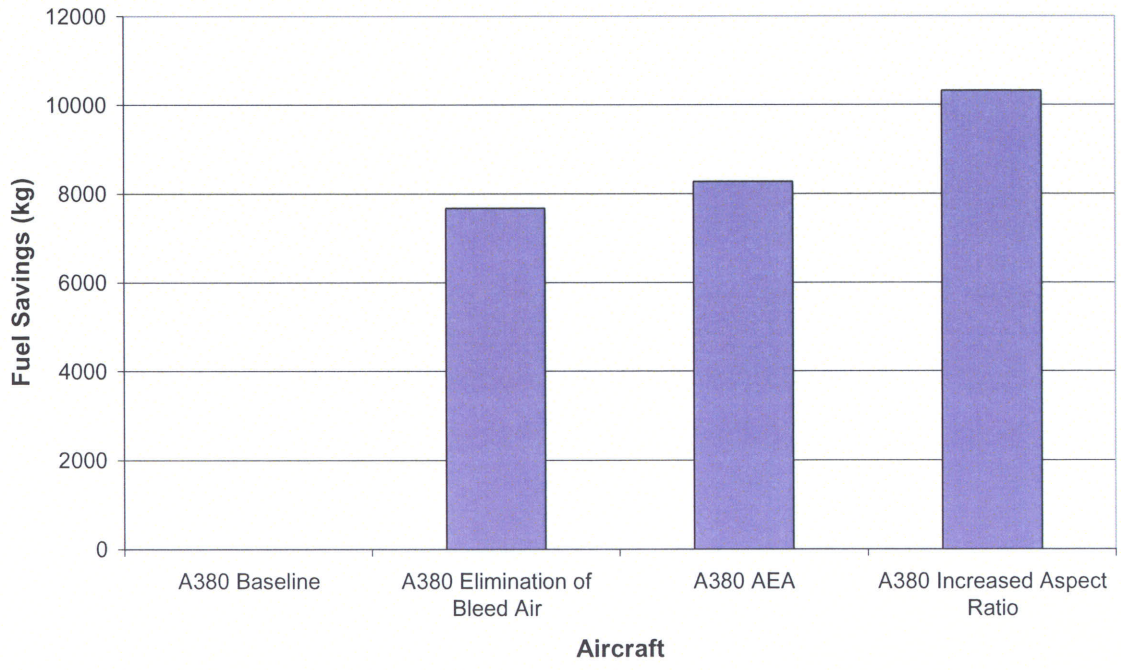


Figure 8.8: Fuel Savings for A380-100 Class Aircraft

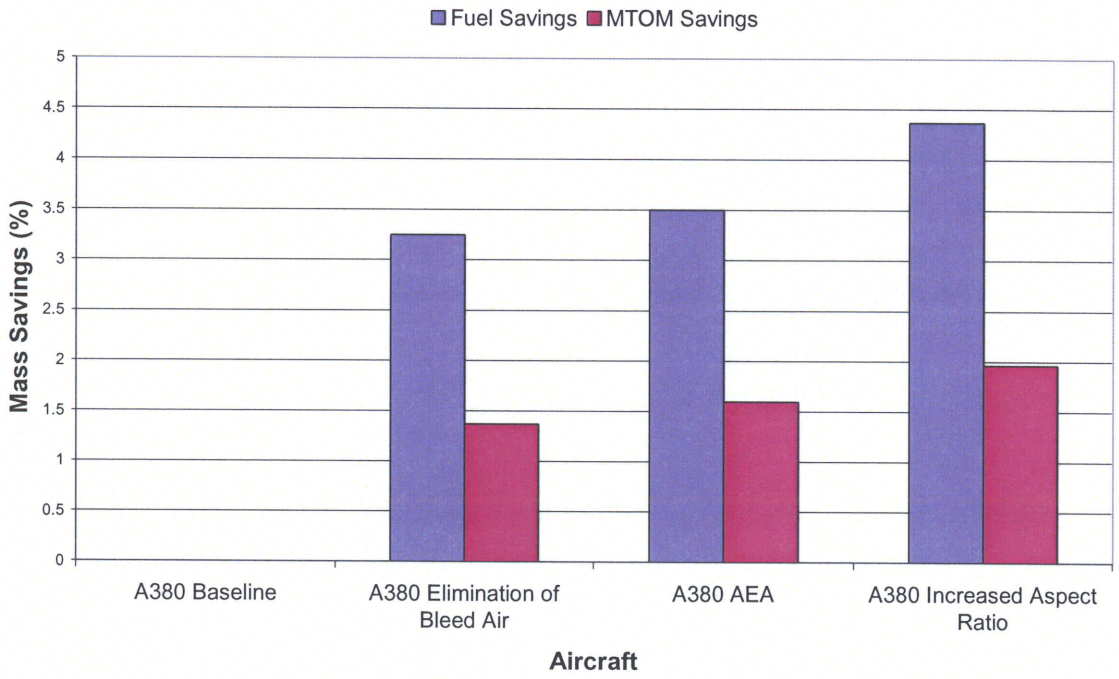


Figure 8.9: Percentage Mass Savings for A380-100 Class Aircraft

The second case identified refers to the design point of the A330-200 class aircraft. As mentioned before, this is a twin-engine aircraft and the bypass ratio of the engines is restricted by mass and off take penalties. Elimination of bleed air could and adoption of the all electric aircraft could result in an overall mass reduction allowing increased engine bypass ratio.

8.6 All Electric Systems and Architectures

Despite its advantages the conventional constant speed constant frequency system has a number of drawbacks including mass penalties, and reliability/maintainability problems of the constant speed drive unit. In addition, although constant power output through the mission profile favors conventional electrical loads this is not true in the case of an all electric aircraft. Present electrical loads remain almost constant throughout the mission profile and therefore a constant speed system is sized, accordingly. In the case of the all electric aircraft or engine concepts, the loads would vary with flight conditions and therefore it is not necessary to size the generators for the maximum load. Instead, the generators can be sized to provide the required power output at low rotational speeds and allow the power to increase with rotational speed. In such a case, the design point of the generators would be the descent and landing phases where the power requirements would be relative high and the engine/generators would operate at low rotational speed. A generator sized for such

conditions would be able to provide the power output required at the remaining phases.

In all cases examined, use of a variable voltage variable frequency system increased mass savings and reduced mass penalties. The findings represent the net effect of two factors. Firstly, the size of the generators was optimized for the minimum loads and secondly the variable voltage variable frequency power improves the operational characteristics of the motors eliminating the need to oversize motors or use controllers.

It is noted, however, that mass reductions can only be realized if the all electric concept is adopted and conventional hydraulic systems are replaced with electromechanical or electro hydraulic actuators. If hydraulic systems are not replaced, the increase in electrical equipment would lead to a small increase in aircraft weight. The increase would depend on the configuration of the aircraft and the power generation and distribution system. In terms of percentages penalties, they would increase in the case of twin engine aircraft were the generators must be oversized to account for the event of an engine failure. In addition, a number of other equipment, such as converters and motors, should also be at least duplicated to ensure the required levels of reliability.

Several power generation and distribution systems have been considered. These are shown in Figures 8.10 – 8.13. Figure 8.10 considers the use of a hybrid VVVF and VSCF system. In this case, two dedicated buses power all the essential electrical equipment. Essential equipment include airframe fuel pumps, engine fuel pumps and the power required for actuation functions. The latter can be electric actuators, dedicated/localized hydraulic systems and/or electrically driven fuel pumps for the primary flight control system and extension/retraction of the landing gear.

Most present aircraft incorporate a single essential AC bus, however due to the high loads of the all/more electric aircraft/engine designs two such buses are required to avoid oversizing the generators. Each bus is powered from a separate generator through a DC link. The DC link units can be designed to operate either in parallel, i.e. sharing the load, or in series. The latter implies that under normal operating conditions only two DC link units operate and the remaining units are used for redundancy purposes. The simplified diagram shown in Figure 8.10 assumes parallel operation. The VVVF buses provide power to the remaining equipment including the environmental control system motors. Since DC loads would remain the same a single Essential DC Bus has been incorporated. The DC system is similar to that used on the Airbus A340-500 aircraft and incorporates two DC buses, each one powered by a separate

generator, and a single battery bus. In the case of a TRU failure both DC buses would be connected to the battery bus and a single generator would supply the power required. The two separate buses or the batteries can be used to supply power to the essential DC bus in the event of the two essential TRUs failing. For safety reasons, twin-engine aircraft, such as the A330-200 class aircraft, would incorporate four generators. This implies that each engine would drive two separate generators. Therefore the system shown in Figure 8.10 can be used with all three aircraft considered.

In the event of generator 2 failure, then generator 1 would be connected to the VVVF Bus 2 and the left wing essential bus would be powered from generator 1. In the event of two generators failing (i.e generators 1 and 2) or an engine failure (in the case of the A330-200 class aircraft), then generators 3 and 4 would supply the power required to VVVF Bus 1 and 2 and also the left wing essential bus. In such a case, the left wing essential bus can be connected to the right wing essential bus and generator 3 would provide the power required. Generator 4 can be connected to VVVF bus 1 and 2. This also allows generator 4 to power the left wing essential bus through the VVVF 2 bus. Finally the APU generators can also be connected to power the VVVF 1 and 2 busses and/or the left wing essential bus.

The system shown in Figure 8.10 has a number of disadvantages. In the case of the A340-500 class aircraft, the maximum power requirements would be around 1210 KW. These power requirements assume maximum hydraulic loads (or actuator loads in the case of an all electric aircraft) maximum electrical loads with the galleys operating, maximum engine fuel pump loads (i.e. take-off conditions) and maximum environmental control system loads (i.e. ground level hot day operations). If all these loads are to be powered simultaneously then four generators sized at 300 KVA are required. However, some of the electrical loads could be shed at take-off conditions reducing the total power requirements by 100 KW. In addition, due to the small duration of the take-off the loads of the environmental control system could be reduced by 200 KW. In such a case the required power output of each generator reduces to 225-250 KVA. During landing the power requirements for the engine fuel pumps would also reduce and therefore each generator could be sized to provide around 200 KVA at landing conditions.

The essential AC power requirements would vary from 540 KW at take-off to 340 KW at landing conditions. Therefore each generator must be able to supply 270 KW at take-off and 170 KW at landing. Each generator should also provide the power required to drive one of the separate compressors. Therefore the total output power

increases to 370 KW and 270 KW respectively. This load however is higher than the 200 KVA load previously mentioned and as a result generators 2 and 3 may need to be oversized to cover the loads. Possible solutions to this problem may include the overrating of the generators at landing. Usually, generators are sized to provide 1.5 times their rated output for 5 minutes. A generator rated at 200 KVA would therefore be able to supply 300 KVA. Although such an action would allow the use of smaller generators, the frequent overrating would affect the reliability of the system and would increase maintenance costs. Allowing the remaining generators to supply all the power required for driving the environmental control system compressors may alleviate the problem. Such an action would require complex switching procedures increasing the complexity of the system. Finally, the system can be designed so that two generators supply the power to each essential AC bus. In such a case DC link 1 and DC link 2, shown in figure 8.11 would be connected to generator 1 and generator 2, respectively.

Another disadvantage of the system shown in Figure 8.10 is that some of the loads powered from the essential busses would require uninterrupted power supply. By definition a VVVF system cannot comply with such requirements. Use of DC link units however can control the frequency output and could allow for such a requirement. The latter implies that in the case of a generator failure (either generator 2 or generator 1) the remaining generator may have to be overrated until generator 1 or generator 2 is connected to the system.

The power requirements for the operation of the environmental control system while on the ground would be around 600 KW. Present APUs are designed to start the engines, supply the environmental control system and also provide some electrical power. Due to the high power requirements in the case of the all-electric aircraft it would not be practical to have APU generators driving the separate compressors. The APU generators would be used to start the engines, supply some of the electrical power required and also provide a proportion of the load required to drive the compressors. Therefore two APU generators could be used each one rated at around 200 KVA. The total power output is enough to supply the power required for the environmental control system under most operating conditions or start the engines.

Figure 8.12 shows a simplified diagram of a VVVF electrical system. One again four engine driven generators are used to supply power to four separate VVVF buses. In this case the essential loads are powered from two separate VVVF buses, each one supplied from an independent generators. The system is similar to the hybrid system with the most noticeable difference being the supply of VVVF power to the majority of

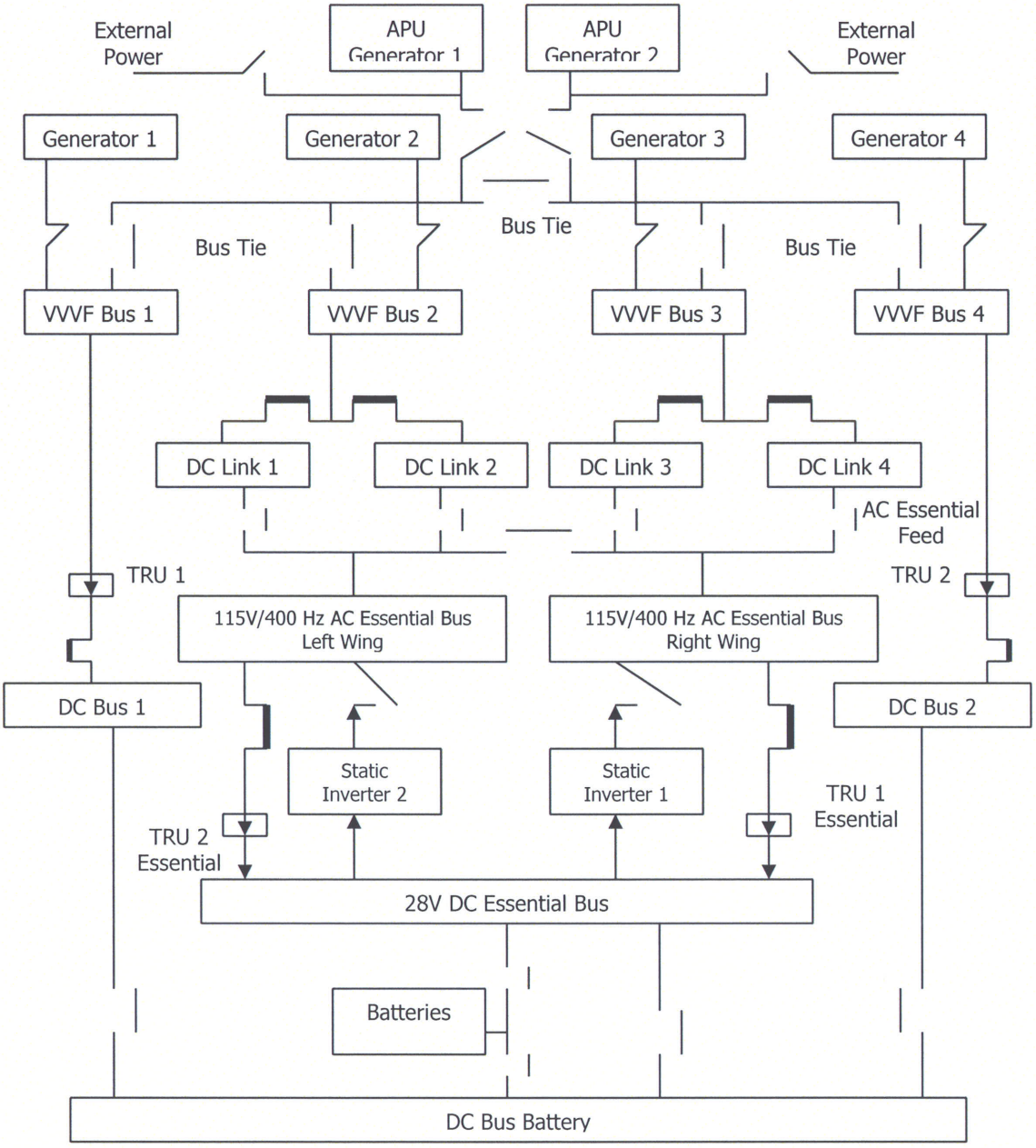


Figure 8.10: Simplified Hybrid VVVF/VSCF Electrical System Diagram

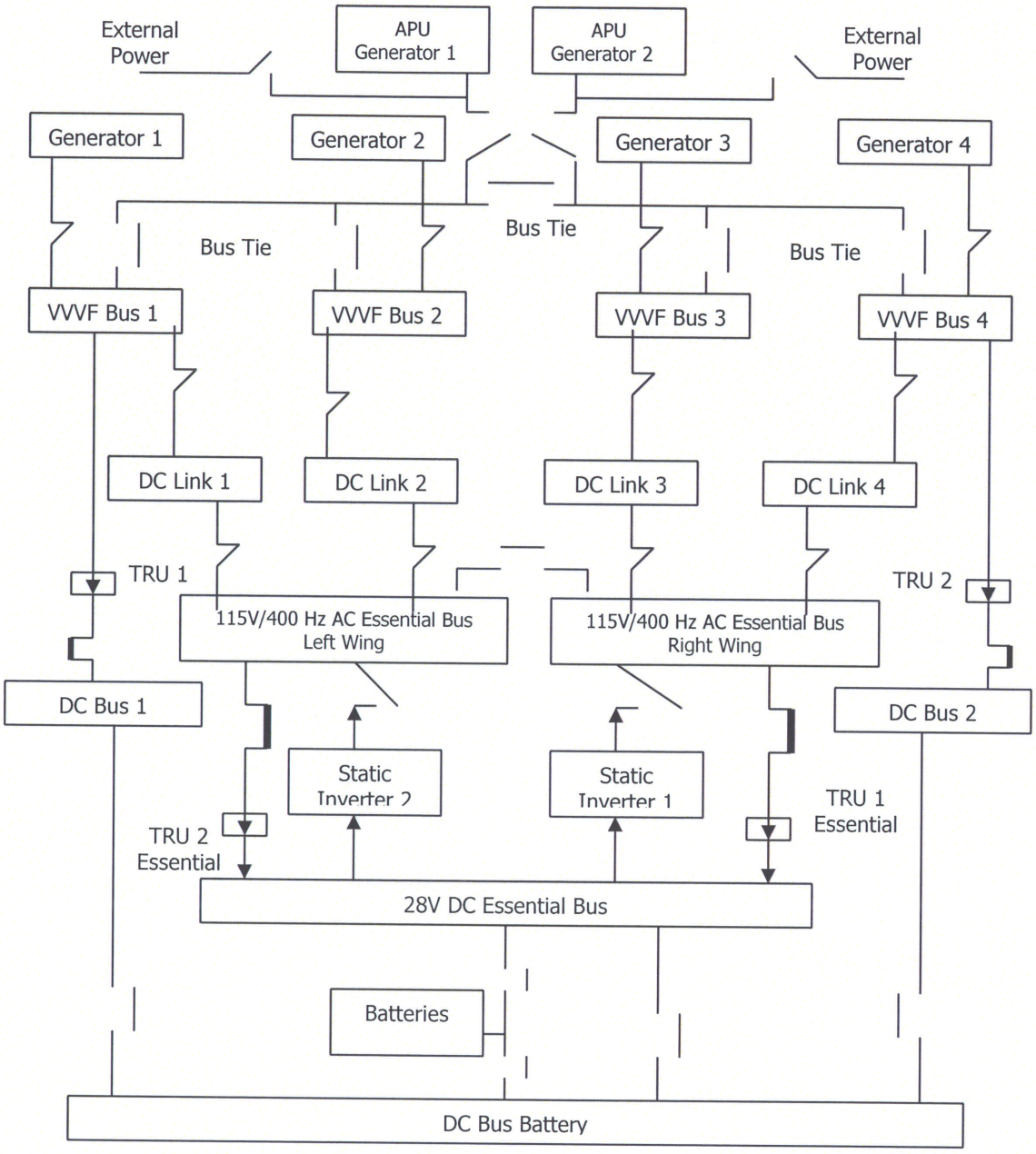


Figure 8.11: Simplified Hybrid VVVF/VSCF Electrical System Diagram Operating in Parallel

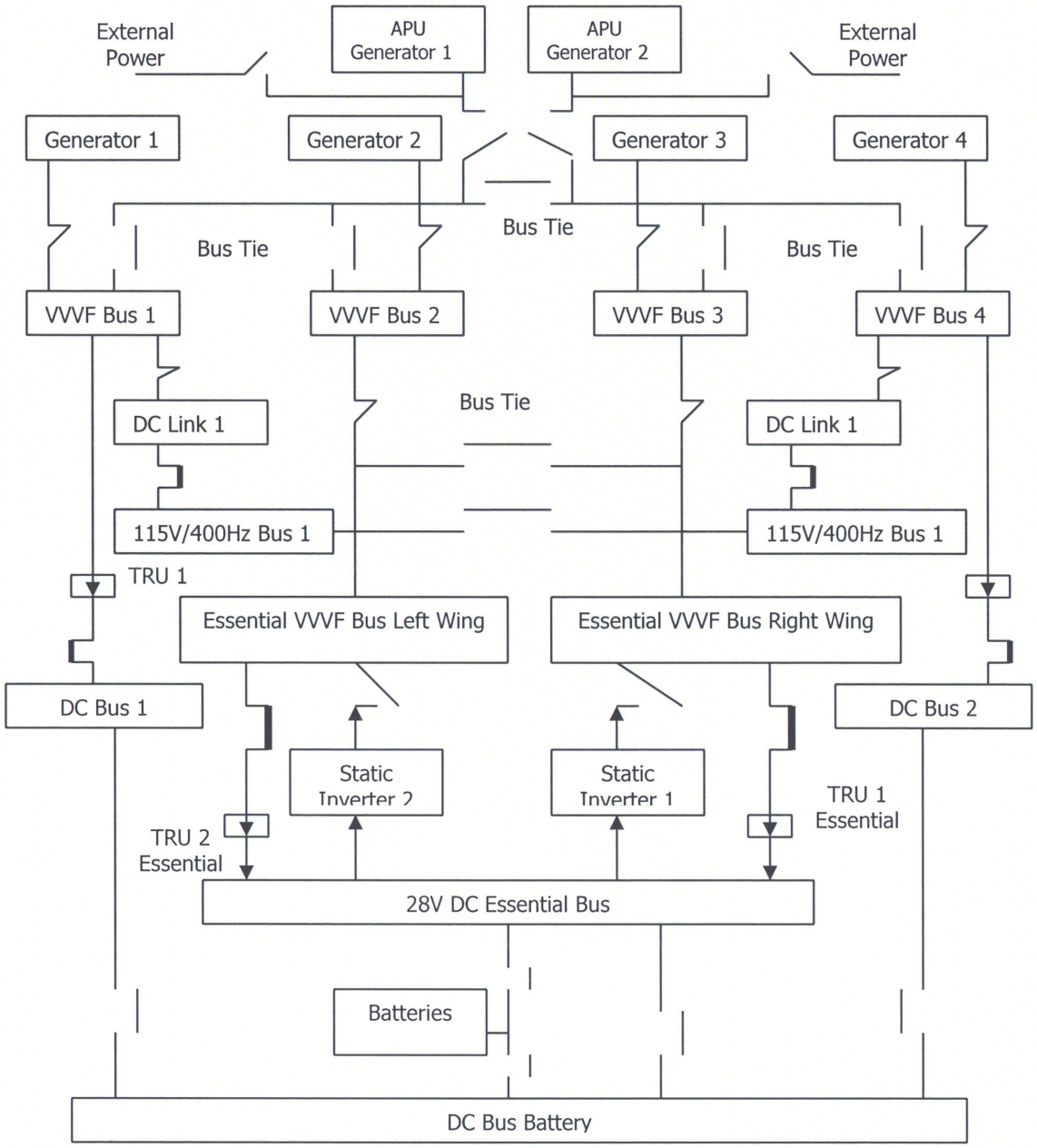


Figure 8.12: Simplified VVVF Electrical System Diagram With Two Essential VVVF Buses Leading to Power Interruption

the equipment. The main disadvantage of such a system is the fact that the use of VVVF power for the essential equipment does not allow uninterrupted power supply in the case of an engine or a generator failure. Although flight control computers and avionics can be powered from the essential DC bus equipment such as fuel pumps and hydraulic pumps would experience a short power cut. The problem can be alleviated through the use of a hybrid system, shown in figures 8.10 and 8.11. Such a configuration, however, would increase the mass of the system due to the use of DC

links, that have to be sized according to their power output. An alternative may be the use of four independent essential VVVF buses. The system is shown in Figure 8.13. The loads supplied by the essential VVVF bus would include engine fuel pumps, hydraulic pumps, and or motors for electro-mechanical and electro hydraulic actuators.

To meet the reliability targets, at least two motors and three controllers are required. For example, as shown in Chapter 7, in the case of electro-hydraulic actuators, three independent systems are required. These include three actuators, three motors and three controllers. Either three or two electrical power supply systems can be used. If each controller and motors are driven by a separate electrical supply system then the overall configuration would ensure uninterrupted power supply to the control surface. In the case of a generator or an engine failure, the bypass or spool valve of the actuator connected to the failed electrical system would open to allow the actuator to move with the control surface. When the power supply is restored, the bypass or spool valve would move to the position that would allow control of the actuator.

In the case of electromechanically actuated primary flight control surfaces two load paths would be required. Each load path would be driven by a separate electrical system and would employ one motor and two controllers. In the case of two motors driving a single jam tolerant actuator, as proposed by Maydew[134], a speed summing differential would be incorporated to prevent a motor failure from jamming the whole system. Therefore loss of power in one electrical power supply system would not affect the operation of the actuator and/or the control surface. If two jam tolerant actuators are used then a separate motor would drive each one. Two controllers would control each motor. A differential gearbox and a brake would be required to prevent a motor failure from jamming the system. Therefore, if each motor and each controller are driven from a separate system then the system could afford loss of a generator.

The same arguments also hold true in the case of hydraulic pumps and electrically driven fuel pumps. In both cases, either a speed summing differential or an epicyclic differential gearbox must be incorporated to ensure that an engine failure would not jam the system. As a result if the motors are driven from separate electric power supply systems the integrity of the system is not affected by the loss of one or more generators. The main difference of the system shown in Figure 8.13 and that shown in Figure 8.12 is that although in all cases the number of power sources available is the same, the system shown in Figure 8.13 incorporates more load paths. In terms of reliability the system shown in Figure 8.12 has four generators connected in parallel and two VVVF buses also connected in parallel. The configuration shown in Figure 8.13 has four generators connected in parallel, in terms of reliability, and four VVVF busses

also connected in parallel. However, in both cases each generator and bus is operated separately and the loads are shared.

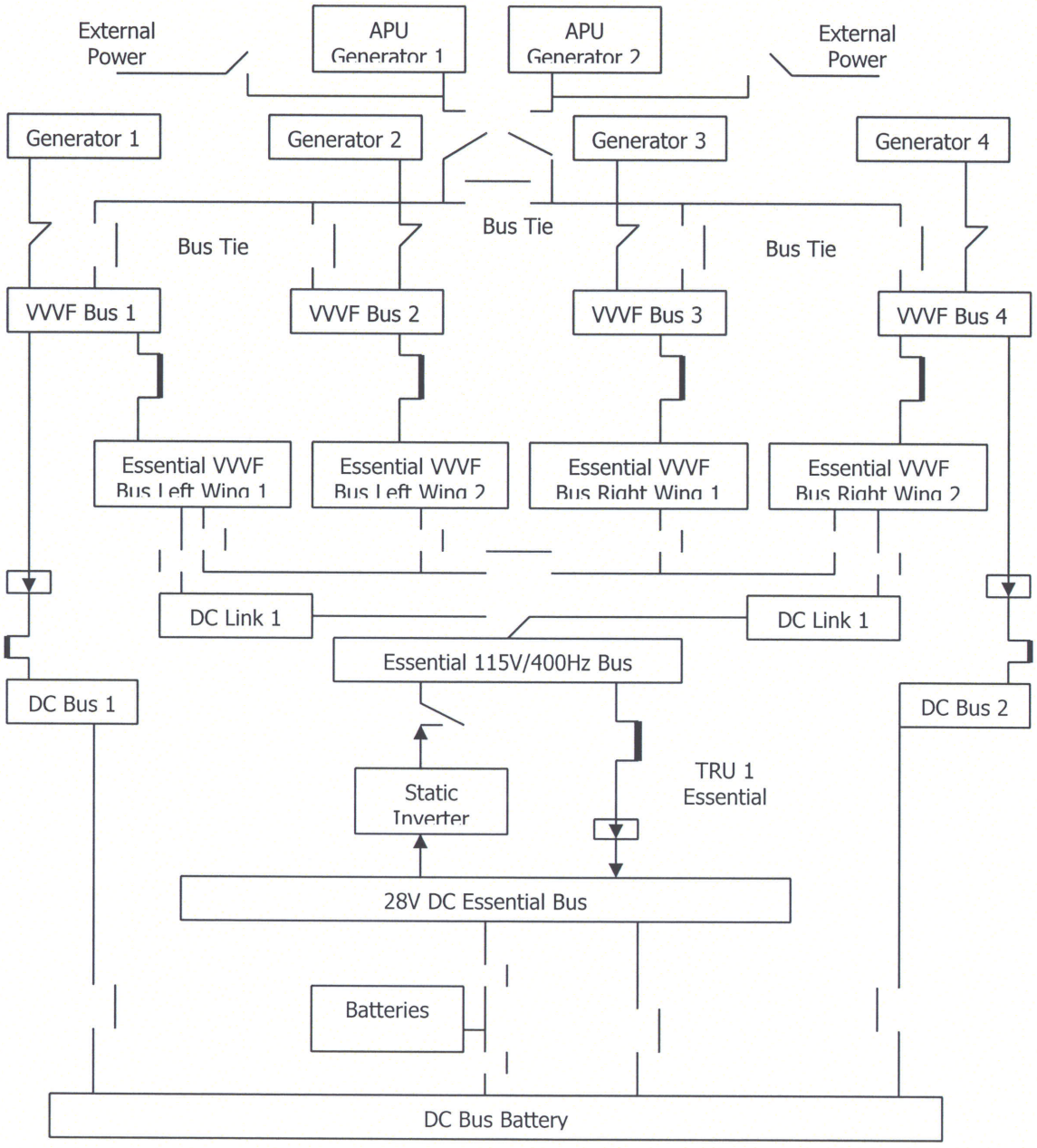


Figure 8.13: Simplified VVVF Electrical System Diagram With Four Essential VVVF Buses

Implementation of all electric technologies can allow for the elimination of the accessory gearbox and this can have a major effect on the design of the engine and the nacelle. Although eliminating the gearbox would allow for reduced weight and frontal area, it also removes the option of mechanically driven engine accessories. This

may prove to be a major problem for critical accessories such as engine fuel pumps. Within the scope of this study it is assumed that the failure rate of the power input to each engine fuel pump should be lower than $1.0*10^{-9}$. The reliability block diagram of an electric drive system is shown in Figure 8.14.

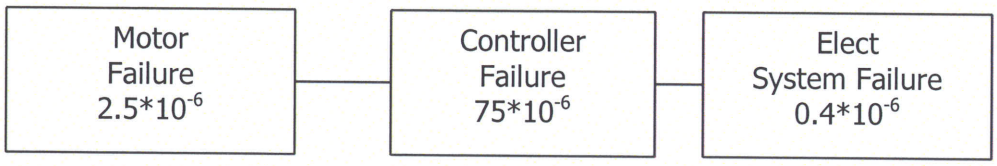


Figure 8.14: Reliability Block Diagram for Electric Drive System

The overall failure rate of such a system is $7.79*10^{-5}$. This is considerably higher than the required target. It is also noted that the individual rate of failure for each component is also higher than the required target. Eliminating the controller and using an induction motor could improve the reliability of the system. In the case of induction motors VVVF supply with constant ratio of voltage to frequency leads to constant torque and increased output power with speed and frequency. The overall efficiency of the system, however, would be lower that the target set. In the case of induction motors the system would require two independent motors powered from two separate electrical systems in such a case the rate of failure reduces to $6.41*10^{-12}$. If permanent magnet motors are used then three controllers are required. The systems are shown diagrammatically in Figures 8.15 and 8.16.

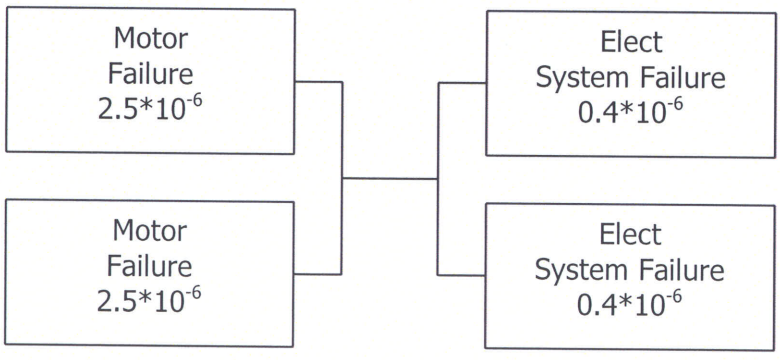


Figure 8.15: Reliability Block Diagram of an Engine Fuel Pump Electrical Drive System Incorporating Induction Motors

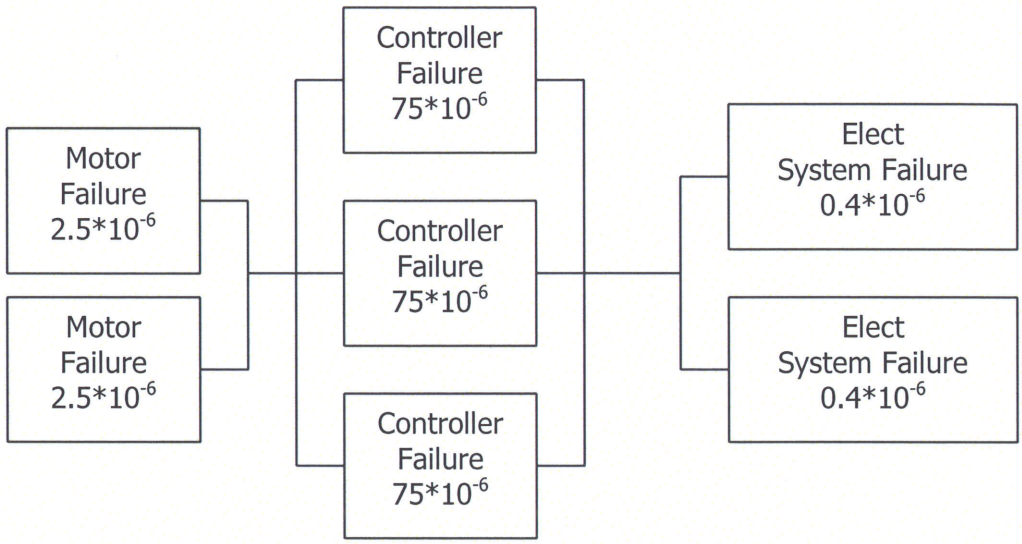


Figure 8.15: Reliability Block Diagram of an Engine Fuel Pump Electrical Drive System Incorporating Permanent Magnet Motors

Similarly to the case of jam tolerant actuators the output of the two motors must be fed into the pump so that a single motor failure would not jam the system. A summing speed differential gearbox could be used but such a solution would lead to unavoidable mass penalties.

8.7 Effect of Aircraft/Engine Characteristics

The impact of the all or more electric technologies depends on the configuration of the aircraft in question. The design specifications of the aircraft along with the number and type of engines would affect a number of parameters including, power requirements, mass changes, sfc and overall fuel usage savings. In general, fuel savings increase with number of passengers, aircraft range, engine bypass ratio and reduced number of engines. Increased number of passengers implies increased secondary power requirements and bleed air in particular. For a given sfc reduction, savings increase with aircraft range. Increased engine bypass ratio leads to increased off-take penalties and as a result elimination of bleed air would have a more pronounced effect on sfc reduction. Finally, as the number of engines reduces the power requirements placed on each engine increase leading to increased off-take penalties. The adverse effect of off-takes on engine performance becomes more critical in the case of twin engine aircraft due to the design point of the engines. The engines of a twin engine aircraft would be sized for the take-off conditions with one engine inoperative. For this reason the engines are oversized. At cruise conditions, the engines operate at significantly lower power setting and as a result the power extracted in the form of shaft power and bleed

air constitutes a larger percentage of the power recirculated with the engine. Although in the case of a four engine aircraft the engine can be sized accordingly to minimize the effect of off-takes, such an action becomes more difficult in the case of a twin engine aircraft.

The design characteristics of the aircraft and the number of engines would also affect the size and the configuration of the electrical power and distribution system. The electrical requirements for extended-range twinjet operations (as in the case of A330-200) demand the provision of sufficient electric power sources to supply services necessary to maintain normal flight and landing, to an alternate airport without requiring exceptional piloting skill, considering the loss of one engine combined with the loss of another normal source of electric power. This requires a minimum of three systems, anyone of which must be capable of powering the required loads. In the case of the all electric aircraft or engine concepts, this includes power to the essential electric and avionics equipment as well as power for the environmental control system and the flight control system. In the case of an engine failure, the minimum air supply defined by the appropriate authorities is 0.4lb/min/passenger. The existing system supplies twice as much. As a result in the event of an engine failure the power supply from the remaining engine would not increase but will remain the same. This power is around 250 KW. The essential electrical power requirements in the event of an engine failure reduce to 100 KW. The electrical power required for the supply of the engine fuel pumps would also remain constant and around 50 KW. Finally the power required for the flight control system would be around 160 KW. The total power requirements would be around 560 KW. Therefore in the case of a twin engine aircraft the design point of the electrical system would not be the descent case but the one engine inoperative condition.

Assuming, however, that the pressure ratio of the environmental control system could be reduced to the levels required at cold day operations the total power requirements would reduce to around 400 KW. If the engine generators need to supply the power at such conditions then four generators are required each one rated at 400 KW. Therefore, with an engine failure and a generator failure the remaining generator, driven by the operating engine would supply the power required. Such an option eliminates the use of an APU and despite the increased mass of the generators, could lead to increased mass savings. However if the APU is to be retained, then three generators are required. Two engine driven generators would be rated at 400 KW each and an APU driven generator rated at 400 KW are required. This would give a total of 1200 KW. Which is lower than the previous figure and reduces the mass of the generators by approximately 120 Kg. The capacity of present generators, however, is

close to 300 KW. In such a case, five generator are required each one rated at 200 KVA. Each engine would drive two generators and the remaining generatorr would be driven by the APU. The total capacity of the system is 1000 KW. However, the aircraft would be allowed to dispatch only if all five generators and the APU are operating properly. If one generator and/or the APU is not operating prior to take-off then the aircraft would be grounded since in the event engine failure and loss of one generator the aircraft would not meet the ETOPS requirements. If six generators, each rated at 200 KVA, are used then the total power output increases to 1200 KVA. Although the figure is higher than that required such a system would allow the aircraft to be dispatched even if one of the generators has failed while on ground. The remaining generators would be able to supply the power required in the event of an engine failure and a generator failure.

The findings from the sensitivity analysis were used to study the effect of mass and sfc changes on the overall performance of each aircraft. The results are shown in Figures 8.16 and 8.17.

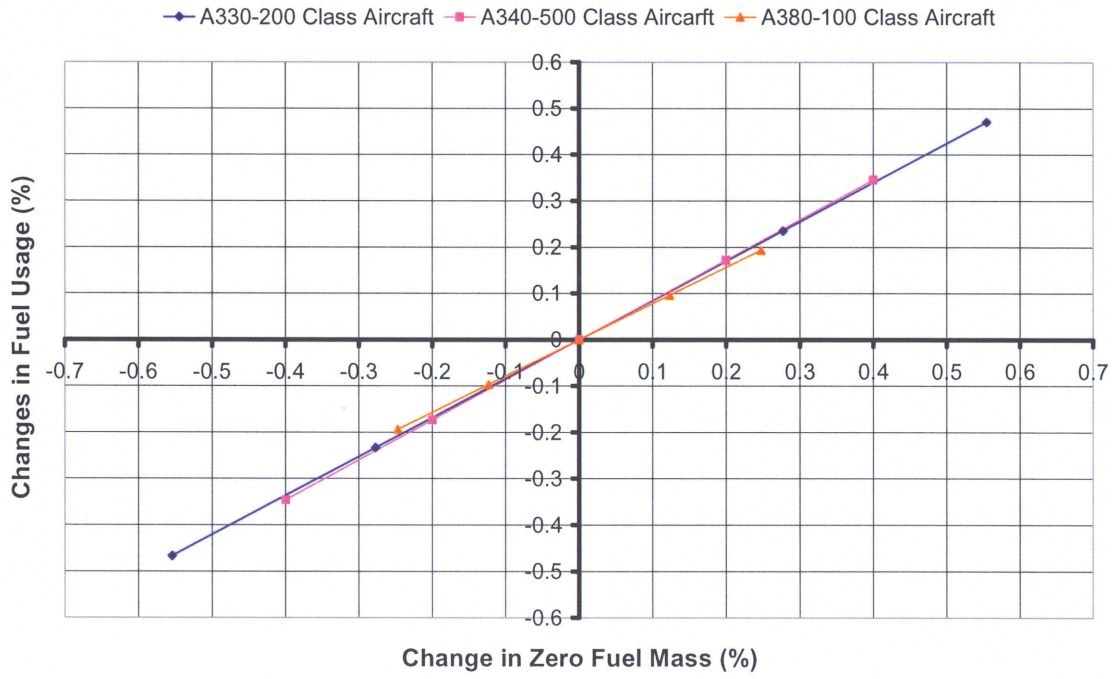


Figure 8.16: Effect of Mass Changes on Fuel Usage

All three aircraft examined present the same percentage changes in fuel usage with changes in zero fuel mass. In the case of sfc changes however, in the case of the A340-500 class aircraft changes in have a stronger influence on overall fuel usage.

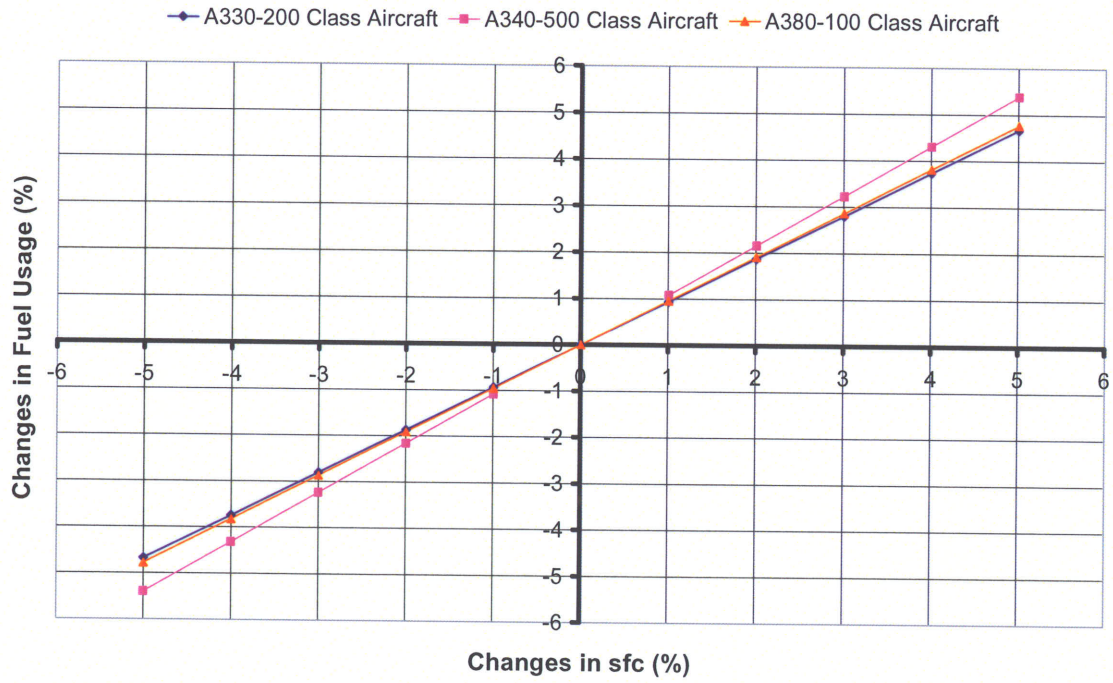


Figure 8.17: Effect of sfc Changes on Fuel Usage

The effect of relative changes in mass and sfc on fuel consumption can be studied through the following equations found in Young [132]:

$$\frac{\Delta fuel}{Zero\ Fuel\ Mass} = \left(e^{\frac{R\ sfc\ g}{E\ V}} - 1 \right) \frac{\Delta\ Zero\ Fuel\ Mass}{Zero\ Fuel\ Mass}$$

$$\frac{\Delta fuel}{Zero\ Fuel\ Mass} = \left(e^{\frac{R\ sfc\ g}{E\ V}} * \frac{R\ sfc\ g}{E\ V} \right) \frac{\Delta sfc}{sfc}$$

where:

Dfuel : increase in fuel (kg)

R: aircraft Range (m)

Sfc : engine specific fuel consumption (kg/Ns)

g : gravitational acceleration (m/s^2)

E: aerodynamic efficiency

V: true air speed (m/s)

Δsfc: increase in sfc

The accuracy of the above equations was validated against the results from the aircraft performance models and the findings were quite similar. The above equations can be used to isolate the effect of aircraft parameters on fuel savings. For a fixed percentage of zero fuel mass reduction, fuel savings increase with time and sfc and reduce with aerodynamic efficiency. Percentage fuel savings, however, depend on the fuel load of the aircraft. Despite the fact that the A330-200 class aircraft spends less time at cruise, percentage changes in fuel usage are similar to those for the A340-500 and A380-100 aircraft due to the lower fuel load of the aircraft.

In terms of sfc savings for a fixed percentage change in sfc fuel changes would also increase with time, sfc and aerodynamic efficiency. Once again the dominant effect would be the time spend at cruise and for this reason the A340-500 aircraft is more sensitive to changes in sfc.

Although implementation of all electric technologies could lead to an overall mass reductions savings would be small and the majority of the savings would result from the optimisation of off-takes and the elimination of bleed air. These changes would allow for improved engine performance and sfc reductions. Reductions in sfc have a more pronounced effect on long range aircraft and therefore adoption of all electric technologies would be more favorable in the case of long range high capacity aircraft.

Chapter 9 Conclusions

1. Effects of shaft power and bleed air off-takes can be modeled by expressing the performance of the engine and its matching procedure in a non-dimensional form. Off-takes should be related to the power re-circulated within the core of the engine. Performance penalties depend on the percentage of off-take extracted relative to the power re-circulated within the core of the engine. Engine pressure ratio and turbine entry temperature have an important second order effect. Thrust penalties depend on percentage of off-take and engine pressure ratio. Turbine entry temperature penalties depend on the percentage of power extracted. Performance penalties associated with a constant percentage of off-take do not depend on engine bypass ratio. Extraction of bleed air introduces larger performance penalties, than the equivalent amount of shaft power, leading to increased sfc and turbine entry temperature. Future engines are likely to suffer higher performance penalties due to their design parameters (i.e. increased bypass ratio, pressure and TET) and increased secondary power requirements of future aircraft. Two different methods presented in this work can be used at the early design stages of the engine to minimize performance penalties associated with extraction of secondary power. Performance penalties can be kept to a minimum provided that the engine is designed to provide them.

2. In the case of supersonic combat aircraft, the bleed air used for environmental control purposes is a lower percentage of the total accessory power load and therefore the saving in fuel consumption by introducing electrically driven compressors would be very small with comparison to the conventional system. On the contrary the system would increase the mass of the aircraft leading to increased fuel usage. In terms of engine performance implications, there is not such a strong case for not using an electrical drive for the operation of the afterburner pump. Although performance penalties would be higher than those imposed by the conventional mechanical system, the difference would be minimal and the performance of the engine will almost be unaffected. The opposite is true however when considering the use of a bleed system. In all cases higher performance penalties were experienced when comparing the bleed systems to the mechanical and electrical drives. In terms of thrust loss and increase in TET, the difference was more noticeable at high altitude low speed operations where the reduced mass flow through the engine had a direct impact on the performance penalties imposed by the bleed systems.

3. Trip fuel calculations within 1% of published data are possible using the aircraft performance model developed.

4. The power extracted in the form of bleed air for the operation of the environmental control system is twice as much the actual power required. In the case of a long-range high capacity aircraft, elimination of bleed air and use of separate electrically driven compressors for the supply of the environmental control system could reduce sfc by 1.5%-2.5% depending on the application and the operating conditions of the aircraft and the engine. Turbine entry temperature could reduce around 15-25 degrees. Reductions in sfc and TET increased with time and altitude. Implementation of a third port within the core of the engine to supply air at pressures and temperatures required for the operation of the environmental control system could reduce sfc but savings would be higher if bleed air is eliminated. Based on these results, the overall fuel reduction was 2.5% for the A340-500 class aircraft, 2.4% for the A330-200 class aircraft and 3.4% for the A380-100 class aircraft.

5. Mass changes would depend on the configuration of the electric system and the extent of adopting all electric technologies. Mass savings can only be realized if the all electric concept is adopted and the conventional hydraulic system is replaced. Adoption of more electric technologies would introduce a small mass penalty. Percentage mass penalties were higher for the A330-200 class aircraft. Adoption of all electric technologies and elimination of the hydraulic system was estimated to reduce the OEW by 0.4% to 0.1% for the A330-200 class aircraft, 0.37% to 0.25% for the A340-500 class aircraft and 0.3% to 0.1% for the A380-100 class aircraft. The total fuel reduction with implementation of all electric technologies and elimination of hydraulics and bleed air could be as high as 2.6% for the A330-200 class aircraft, 2.75% for the A340-500 class aircraft and 3.5% for the A380-100 class aircraft.

6. Several power system architectures are possible. Generation and distribution of Variable Voltage Variable Frequency increased mass savings and reduced mass penalties. The main advantage of the system is that the power output varies with engine/generator rotational speed and the generators can be sized for the minimum power requirements. In addition, supply of VVVF power to induction motors improves their performance characteristics eliminating the need to oversize the motors or use controllers.

7. Adoption of all or more electric technologies could allow for the design of more efficient aircraft. Increasing the aspect ratio of the A80-100 class aircraft to 8.33 could improve aerodynamic efficiency leading to an overall fuel reduction of 4.4% and a MTOM reduction of 2.0%. In the case of twin engine aircraft reductions in MTOM and reduced off-take penalties could allow an increase in engine bypass ratio leading to increased overall performance. For the A330-200 class aircraft elimination of bleed air and increase in bypass ratio resulted in a fuel reduction of 4.5%.

8. The impact of the all or more electric technologies depends on the configuration of the aircraft in question. The design specifications of the aircraft along with the number and type of engines would affect a number of parameters including, power requirements, mass changes, sfc and overall fuel usage savings. In general fuel savings increase with number of passengers, aircraft range, engine bypass ratio and reduced number of engines. Increased number of passengers implies increased secondary power requirements and bleed air in particular. For a given sfc reduction, savings increase with aircraft range. Increased engine bypass ratio leads to increased off-take penalties and as a result elimination of bleed air would have a more pronounced effect on sfc reduction. Finally, as the number of engines reduces the power requirements placed on each engine increase leading to increased off-take penalties. The adverse effect of off-takes on engine performance becomes more critical in the case of twin engine aircraft due to the design point of the engines. The engines of a twin engine aircraft would be sized for the take-off conditions with one engine inoperative. For this reason the engines are oversized. At cruise conditions, the engines operate at significantly lower power setting and as a result the power extracted in the form of shaft power and bleed air constitutes a larger percentage of the power recirculated with the engine. Although in the case of a four engine aircraft the engine can be sized accordingly to minimize the effect of off-takes, such an action becomes more difficult in the case of a twin engine aircraft.

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