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Thermo and Fluid Dynamics

On the Design of Energy Efficient Aero Engines
Some Recent Innovations

By

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Göteborg, Sweden, 2011

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[Artist's impression of a future energy efficient aircraft driven by counter-rotating propeller engines. Source: Volvo Aero Corporation]

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Abstract

In the light of the energy crisis of the 1970s, the old aerospace paradigm of flying higher and faster shifted towards the development of more energy efficient air transport solutions. Today, the aeronautical research and development community is more prone to search for innovative solutions, in particular since the improvement rate of change is decelerating somewhat in terms of energy efficiency, which still is far from any physical limits of aero engine and aircraft design. The Advisory Council for Aeronautics Research in Europe has defined a vision for the year of 2020 for aeronautical research in Europe which states a 50% reduction in CO_2, 80% reduction in NO_x and a 50% reduction in noise.

Within this thesis work, methods for conceptual design of aero engines and aircraft performance have been developed and applied to evaluate some innovative aero engine concepts that have the potential to fulfil or even surpass society's expectations on the aerospace industry in the future. In particular, the impact of a varying engine size and weight on the aircraft performance has been modelled in order to quantify the fuel consumption of different aero engine concepts. Furthermore, methods for designing and analyzing propeller performance have been developed. The methods have been incorporated into a multidisciplinary optimization environment which gives the benefit of interdisciplinary quantification of design changes and the impact of those on energy efficiency.

The potential of the variable cycle engine for medium range jets were studied and the results showed a quite large reduction in fuel consumption compared to the conventional turbofan engine. Furthermore, the inter-turbine reheated aero engine concept was evaluated and the results indicated a large NO_x reduction potential at almost the same energy efficiency as the conventional engine. The idea of applying catalytic combustion in aero engines was also studied showing potential of significant reductions of NO_x. Finally, an innovative propeller design based on Prandtl's work in the 1920s is suggested and discussed.

This work has contributed with new methods for conceptual aero engine design that are in use within the industry and academia. The results from the studies concerning innovative aero engine concepts show that major improvements in terms of energy efficiency and emissions still are possible for the aerospace industry to achieve.

Keywords: aero engine, energy efficiency, turbofan, propeller, inter-turbine reheat, emissions, NO_x, variable cycle, MDO

List of Publications

This thesis is based on the work contained in the following papers and reports:

I. Avellán, R. and Grönstedt, T., *Preliminary Design of Subsonic Transport Aircraft and Engines*, the 18th ISABE Meeting, September 2-7, 2007, Beijing, China, ISABE-2007-1195.

II. Lundbladh, A. and Avellán, R., *Potential of Variable Cycle Engines for Subsonic Air Transport*, the 18th ISABE Meeting, September 2-7, 2007, Beijing, China, ISABE-2007-1156.

III. Ekstrand, H., Avellán R. and Grönstedt, T., *Minimizing Direct Opeating Costs (DOC) for a small European Airline*, the 18th ISABE Meeting, September 2-7, 2009, Beijing, China, ISABE-2007-1105.

IV. Avellán, R. and Grönstedt, T., *An Assessment of a Turbofan Using Catalytic Interturbine Combustion*, Proceedings of ASME Turbo Expo 2009, June 8-12, Orlando, Florida, USA, GT2009-59950.

V. Ekstrand, H., Avellán R. and Grönstedt, T., *Derated Climb Trajectories for Subsonic Transport Aircraft and their Impact on Aero Engine Maintenance Costs*, the 19th ISABE Meeting, September 7-11, 2009 Montreal, Canada, ISABE-2009-1340.

VI. Olausson, M., Avellán, R., Sörman, N., Rudebeck, F. and Eriksson, L.-E., *Aeroacoustics and Performance Modeling of a Counter-Rotating Propfan*, Proceedings of ASME Turbo Expo 2010, June 14-18, Glasgow, UK, GT2010-22543.

VII. Avellán, R. and Grönstedt, T., *Potential Benefits of Using Inter-Turbine Reheat in Turbofan Engines*, 2011. To be submitted to the Journal of Engineering for Gas Turbines and Power.

VIII. Avellán, R. and Grönstedt, T., *A Gas Turbine Engine*, International Patent WO 2009/082275A1, July 2, 2009.

IX. Avellán, R. and Lundbladh, A., *Air Propeller Arrangement and Aircraft*, International Patent Application, WO2011/081577A1, July 7 2011.

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I would like to thank my supervisor Tomas Grönstedt for all effort, great discussions and ambitious way of supervising. I do believe it might not always be an easy task to supervise a student that delivers… something else that perhaps might more interesting…

I am very grateful for all discussions with Anders Lundbladh; I got answers to everything I asked and often answers to questions I did not know I had…yet. I did surely not know that complex problems could be so easily explained, so quickly.

All colleagues at Chalmers; you have made the time at the department to a very pleasant memory. Also, I would to express my gratitude to Gunnar Johansson, Ulf Håll and Lars-Erik Eriksson for very insightful conversations of various topics. You made me realize I have still plenty to learn...

My colleagues at Volvo Aero have of course been an important support through this journey, those who inspire me whether they now it or not.

At last but not least, my family; I could not have done this without you.

There is a theory which states that if ever anybody discovers exactly what the Universe is for and why it is here, it will instantly disappear and be replaced by something even more bizarre and inexplicable. There is another theory which states that this has already happened.

Douglas Adams

To Maria and Tom.

Nomenclature

Acronyms

ACARE	Advisory Council for Aeronautics Research in Europe
BPR	Bypass-Ratio
CFD	Computational Fluid Dynamics
CAA	the Civil Aviation Authority, Computational Aero-Acoustics
C^3	Clean Catalytic Combustor
EIS	Entry Into Service
E^3	Energy Efficient Engine
GDP	Gross Domestic Product
GE	General Electric
GTF	Geared Turbofan Engine
HPC	High-Pressure Compressor
HPT	High-Pressure Turbine
IPC	Intermediate Pressure Compressor
IPT	Intermediate-Pressure Turbine
IRA	Intercooled Recuperated Aero engine
LDI	Lean-Direct Injection
LHV	Lower Heating Value
LPP	Lean-, Pre-vaporized, Pre-mixed
LPT	Low-Pressure Turbine
LRC	Long Range Cruise
LTO	Landing and Take-Off
MDO	Multi-Disciplinary Optimization
MRG	Medium Range Generic
MTU	Motoren- und Turbinen- Union
OPR	Overall Pressure Ratio
P&W	Pratt & Whitney
RQL	Rich-burn, Quick-quench, Lean-burn
R&D	Research & Development
TIT	Turbine Inlet Temperature
TRL	Technology-Readiness Level
WIPO	World Intellectual Property Organization
SFC	Specific Fuel Consumption

Greek

η	efficiency
ϕ	equivalence ratio
Λ	geometric sweep angle

Latin

b	span
D	Drag
h	Specific Enthalpy
L	Lift
m	mass
$\dot{m}$	mass flow
n	Load factor
v	velocity

W	Weight
T	Thrust
t/c	thickness-to-chord

Subscripts

ε	downwash
f	fuel
p	propulsive
th	thermal
i	induced
o	overall
tr	transfer
w	wet
N	Net
t/o	take-off
¼	quarter chord
$\perp$	orthogonal to
∞	free-stream

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Foreword

Ever since the first successful powered flight by the Wright brothers in 1903, there has been a tremendous development in the field of aeronautical research. The Wright brother's achievement was quite astonishing at the time, and still is. Their first flight over the sand dunes of Kill Devil Hill in North Carolina, U.S., was reported to last for 12 seconds covering a ground distance of some 37 meters giving a an approximate ground speed of 11 km/h.
Today, long distance aircraft such as the Boeing 777 or the Airbus 380 cruise at almost 900 km/h covering ground distances of almost half of earth's circumference in less than 20 hours which is also quite an achievement, in our time.

The old aerospace paradigm of flying higher and faster pushed the development during the major part of the twentieth century with the prime era of the NASA space flight program, the supersonic transport (SST), the Concorde and indeed all the military aircraft developed during this period. During the early 1970s, in the shadow of, and the light of, the energy crisis, the aerospace industry experienced a slight change in this mind set and the quest for more energy efficient air transport solutions was raised. One important consequence of this was the broad search for innovative aircraft and engine designs that was initiated by the energy crisis. This was probably the first time in history that a government called for innovative energy efficient solutions in order to meet the demands from the society concerning greener air transports.

Today, the aeronautical research and development community is more prone to search for innovative solutions, in particular since the improvement rate of change is decelerating somewhat in terms of energy efficiency, which still is far from any physical limits of aero engine and aircraft design. At the same time the society intensively calls for greener air transport, especially as a consequence of the climate reports produced by the Intergovernmental Panel on Climate Change (IPCC) and the impact of aviation on the global atmosphere. Despite this, the aero engine and aircraft development continues at a rather descent pace, and the modern turbofan aero engine is quite an impressive piece of art. However, one can be sure that the best aero engine designs are not yet known and are waiting to be developed...

My hope is that if anyone who eventually would read this thesis would be inspired, and find at least one new question to be answered as a consequence. If so would be the case, then my mission would be completed.

Richard Avellán
Göteborg, August 2011

1 Innovation and Technology

Over the past there has been tremendous development of science and technology in the world. Except for the traditional explanatory variables regarding long-term growth, e.g. demographic evolution, arable land, presence of fossil fuels and raw material science, technology and innovation are of crucial importance for long-term growth.

The argument that the prosperity of the western world's society relies heavily on successful science, technology development and innovation can hardly be questioned. Some researchers even claim that science, technology and innovation are the only comparative advantages Europe can bring to bear in order to secure its share of the world's future growth (Berg, 2010).

The innovation process can briefly be described as the process of transforming inventions into advantageous outcomes for the society. An indicator, although not complete, of a country's innovation capacity is the number of patents granted in relation to its gross domestic product. Figure 1 shows the top 20 countries in terms of international patents granted (WIPO, 2010) and their corresponding GDP (IMF, 2011). It is worth noting that most of the countries appearing on the WIPO top 30 list are also in the GDP top 30 list. With some exceptions, the invention market share of each country follows the gross domestic product share. Worth noting is the fact that Japan and South Korea have relatively high numbers of patents granted compared to their GDP.

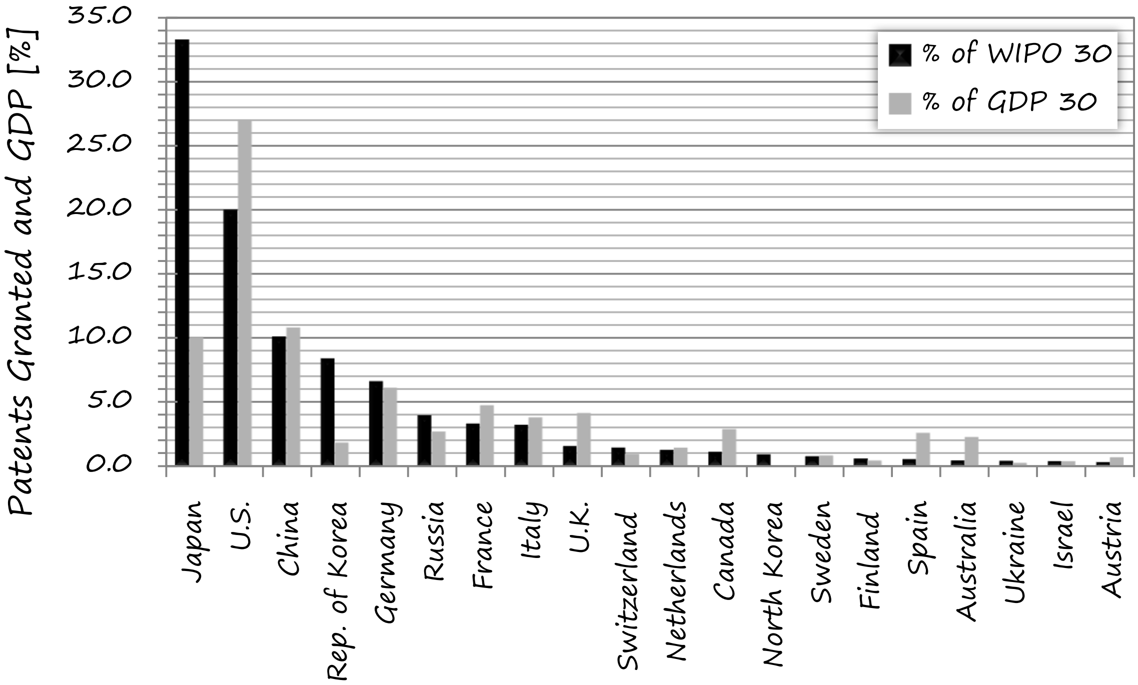


Figure 1 Number of patents granted in relation to total WIPO 30 patents and GDP in relation to total GDP 30.

Whether an invention will successfully be transformed into an innovation is not explicitly determined by the patent itself. Other aspects, not easily quantifiable and not of technical nature, also play parts in the process. There are many cases of successful, and not so successful, technologies introduced in the past that could be studied in order to create better understanding for the future, and so is done in many fields.

A man who is usually credited for establishing the research field of technology history and the coupling of technology, culture and society is Melvin Kranzberg (1917-1995) co-founder of the Society for the History of Technology (Hansen, 2003) and professor in history of technology at Georgia Tech and editor of the journal of Technology and Culture (Gelder, 1995, Garfield, 1992). One of Kranzberg's arguments was that technology development could not be understood without understanding how it was linked to the society. He is also known for the six laws of technology (Kranzberg, 1986) that are briefly introduced in this text and suitable for the introduction of this work.

Kranzberg's first law of technology states, "Technology is neither good nor bad; nor is it neutral", which implies that the application of new technology is always associated with trade-offs. A large-scale example of such a trade-off is the introduction of DDT to eliminate disease-carrying pests and, thus, to raise the agricultural productivity. In India in the 1950s and 1960s the use of DDT cut malaria from 100 million cases per year to only 15,000. This was a tremendous technological achievement, but later it was discovered that DDT threatened the ecological system by entering the food chain of birds, fish and eventually of man. In the west, DDT was banned and replaced by more expensive alternatives, but in India its use was continued since it was considered to be a net good (Lawton, 2009). This directly relates to the fourth law of Kranzberg; "Although technology might be a prime element in many public issues, nontechnical factors take precedence in technology-policy decisions". This means that no matter how good or revolutionary the new technology might be the success of its introduction or acceptance depends on a number of things, many of them of nontechnical nature.

The second law of Kranzberg, probably the most relevant for the purpose of this thesis is stated as; "invention is the mother of necessity", the most successful inventions creates a market and a way forward. In essence, a human brain, an engineering department or research society will respond to the demands placed upon it.

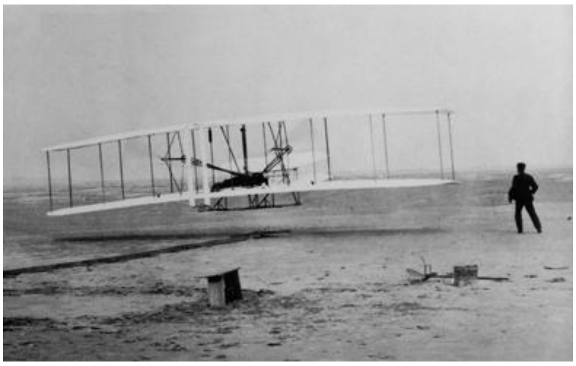
In general and according to the author's opinion, the scientific and engineering community should strive to generate a large number of ideas, and eventually the ultimate solution will become clear. For instance, Thomas Edison, the inventor of the light bulb, worked with idea quotas; one small invention every 10 days and one major invention every six months. This is well in line with Kranzberg's third law "Technology comes in packages, big and small".

Before attempts are made to create the way forward it is useful and of great interest to realize where we came from and where we are heading. For this reason a brief history of aviation innovations is presented with a special emphasis on how breakthrough technologies and innovation have emerged in the past. History may also serve as a source of inspiration while pursuing future solutions for providing energy efficient aero engines. This approach is supported by the fifth and sixth laws of Kranzberg, "All history is relevant, but the history of technology is the most relevant" and "Technology is a very human activity - and so is the history of technology".

1.1 Historical Notes on Aviation Innovations

1.1.1 The first powered heavier-than-air flight in history

The first flight using a heavier-than-air machine took place at the sand dunes of Kill Devil Hills, 6 kilometres south of Kitty Hawk, North Carolina, on December 17, 1903. The brothers, Orville and Wilbur Wright, certainly made a historical breakthrough when they started from level ground and flew their powered biplane Wright Flyer I approximately 3 meters above the ground, lasting 12 seconds covering a distance of approximately 37 meters (Anderson, 2000) (Hansen, 2003). A highly recommended excerpt from the diary of Orville Wright is attached in the appendix of this thesis and describes this remarkable event as told by the brothers themselves.



The Wright brothers submitted their patent application in 1903 and 1906 they finally received the patent on their airplane they tested in North Carolina. The brothers started a legal suit against Glenn Curtiss who built an airplane with many of Wright's innovations included. The case was never settled to Wright's satisfaction.

Figure 2. From the left: The first heavier-than-air flight in history on December 17, 1903 and the Wright Brother's patent on their Flying Machine. Source: NASA.

The Wright brothers did not have any formal education; they had a more practical engineering background as they operated a bicycle repair shop and factory in Dayton, Ohio. The brothers early developed a genuine interest for aviation. They spent a lot of time and effort in experimenting with kites and gliders. One of the great obstacles they had to overcome in order to perform the first powered flight was related to the power plant. They had problems in finding an engine with a good enough power to weight ratio since most engines at that time were extremely heavy. Eventually the brothers designed and built their own piston engine during the winter of 1903. The engine developed approximately 12 hp at a weight of 90 kg. Another obstacle, perhaps even more difficult to overcome, was to find an efficient propeller that they could use for their airplane. The brothers had to develop their own propeller design methodology due to the lack of progress within the research field. One of the brother's great contributions to the field of aviation research was the development of propeller blade design by theory coupled to verifying experiments. The propeller for the Wright Flyer I developed an efficiency of 66% compared to the propeller designed by Samuel Langley which had an efficiency of 52% (Garber, 2011). The Wright brothers continued their propeller development and developed a more efficient design; "the bent-end" propeller that was used between 1905 and 1915. The "bent-end" propeller from 1911 was reproduced and tested by the Wright Experience research team in 1999 showing a peak efficiency of 81.5% (Kochersberger et al., 2000) which is remarkable considering that "modern" wooden propellers reach an efficiency in the range of 84 to 85%.

With a lacking theoretical background and also very little theoretical work done on air propellers, the brothers decided to build their own wind tunnel in order to develop and test a propeller based on experiments.

Curiosity, ambitions and persistence, a systematic way of working and a dedicated interest has in the history of aviation innovation shown to be at least as important to finding a way forward as formal education. A particular example of is discussed below, when Dr A. A. Griffith rules out the inventions of Frank Whittle as impractical.

1.1.2 The inventor of the modern airplane

Even though the Wright brothers are very much remembered as the originators of modern aviation, they did not actually invent the airplane. Credits for being the inventor of the airplane is usually given to Sir George Cayley. In 1799 he described the design of an airplane, as we know it today, using a fixed wing design for generating lift, a separate mechanism for propulsion (he envisioned paddles) and a vertical tail for stability. The next 50 years, after the work of Sir George Cayley, although intense activity pursued in attempts to conquer the air, little progress made within aeronautical research until the late 19th century when Otto Lilienthal, also known as the glider man, published a book entitled *Der Vogelflug als Grundlage der Fliegekunst* which is one of the early classics in aeronautical engineering (Anderson, 2000). Lilienthal carefully analyzed the flight of birds and also applied it to the design of mechanical flight. This book contained one of the most extensive aerodynamic data sets available at this point in time. Lilienthal made more than 2000 successful flights before he eventually suffered a fatal accident during one of his gliding experiments. On his gravestone in the Lichterfelde cemetery the epitaph "*Opfer müssen gebracht werden*" ("*sacrifices must be made"*) is carved. The Wright brothers very much relied on the early work of Lilienthal in the beginning of their own experiments.

1.1.3 The National Advisory Committee for Aeronautics is born

Despite the historical flight of the Wright Brothers in 1903, the United States fell behind in aeronautical research and the nation felt that it needed a centre for aeronautical research in order to catch up with Europe technologically, and NACA was born on March 3 1915. The first meeting was arranged in the office of the secretary of war on April 23, 1915. The seriousness of this matter and the importance of catching up with Europe can be symbolized by the leading personalities from both academia and the military that were attending the meeting shown in Figure 3.

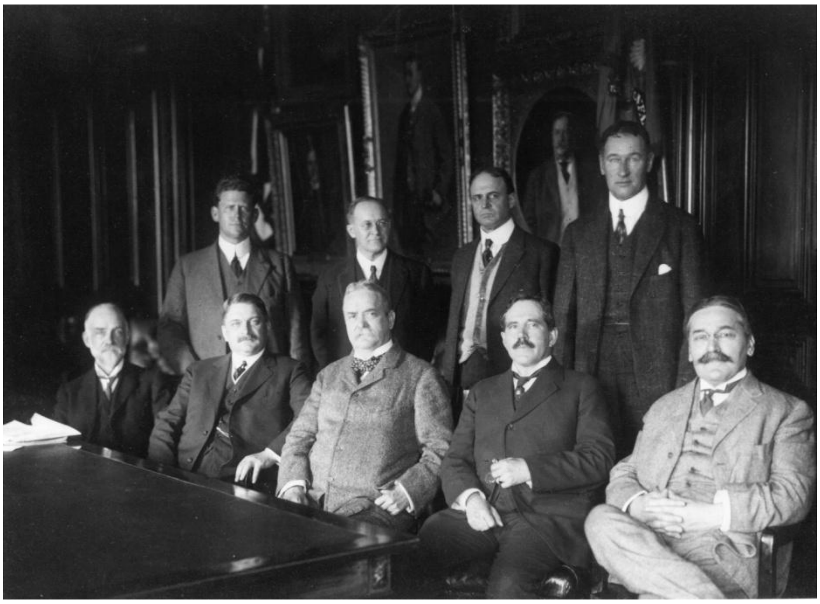


Figure 3. The first NACA meeting in history on April 23, 1915. Source: NASA.

Seated from left to right: Dr. William Durand, Stanford University, California. Dr. S.W. Stratton, Director, Bureau of Standards. Brig. Gen. George P. Scriven, Chief Signal Officer, War Dept. Dr. C.F. Marvin, Chief, United States Weather Bureau Dr. Michael I Pupin, Columbia University, New York. Standing: Holden C. Richardson, Naval Instructor. Dr. John F. Hayford, Northwestern University, Illinois. Capt. Mark L. Bristol, Director of Naval Aeronautics. Lt. Col. Samuel Reber, Signal Corps. Charge, Aviation Section Also present at the First Meeting: Dr. Joseph S. Ames, Johns Hopkins University, Baltimore, MD. Hon. B. R. Newton, Asst. Secretary of Treasury.

After the United States entered the First World War in 1917 things started to happen; in the third annual NACA meeting it was decided that a research facility was to be built on the Signal Corps Experimental Station, Langley Field, Hampton, Virginia. This was the starting point for establishing a number of research centres and research facilities in the USA. On the 4 October, 1957, the world was stunned when Russia launched the Sputnik I satellite and closely after this, on 29 of July 1958 the National Aeronautics and Space Administration (NASA) was born and the race for space started.

Most of the research carried out at the NACA and NASA facilities is publicly available, which has been, and still is, of great value for the aeronautical society. This openness should be brought forwards as another ingredient for successful aviation innovation.

1.1.4 The Jet Engine

In 1903 the Norwegian Aegidius Elling (1861-1949) demonstrated a gas turbine that developed positive net power (Andersson and Karling, 2003). The concept was patented in 1884. The first patent related to jet-propulsion is from 1908 by the French inventor René Lorin, in which he suggests using a piston engine with several nozzles to translate the kinetic energy in the jet to propulsion power. In 1913, Lorin also patented a quite detailed design of a jet engine based on ram compression in supersonic flight (Mattingly, 2006, Prisell, 2003). The first patent that can be related to the modern design of the turbojet engine is from 1921 by the French inventor Guillaume. In this patent Guillaume describes an axial flow machine (both axial compressor and axial turbine). In the light of Guillaume's work it is questionable whether a majority of the turbojet related approved after 1921 provide sufficient novelty to be acceptable in its full claims.

However, in practice the jet engine era did not really take off until the true jet engine pioneers entered the scene in the 1930s, i.e. Dr Hans von Ohain (1911-1998), Sir Frank Whittle (1907-1996) and Secondo Campini (1904-1980).

Dr Hans von Ohain defended his thesis at the University of Göttingen in 1935, the same year he applied for his jet engine patent, which also was approved the same year. The patent described a jet engine using a radial compressor and a radial turbine. After demonstrating the very first prototype HeS1 (Heinkel Strahltriebwerk) rated at 1,1 kN at 10000 rpm, von Ohain was employed by Heinkel where he became the manager of the jet engine department. At this time the development of the test aircraft He178 started, and this also meant that the first prototype engine, HeS1, had to be rescaled in order to meet the performance requirements of the aircraft (approximately 5 kN). The very first jet engine propelled flight was conducted in August 27, 1939. The engine was the HeS3B (Heinkel Strahltriebwerk), the aircraft was a test aircraft, He178, and the test pilot was Erich Warsitz. The first flight lasted for some minutes and proceeded well.

At the same time in England Sir Frank Whittle was working with his idea of the jet engine. His father had his own machining tool shop where Whittle worked after school hours which gave him useful practical skills for his future career. After a couple of unsuccessful applications for the Royal Air Force (RAF) pilot training, Whittle was accepted in 1923 and graduated from the RAF at Cranwell in 1928 with the senior thesis "Future Developments in Aircraft Design", in which he described his idea concerning jet propulsion. Whittle went on and patented his jet engine idea in 1930. After some time a meeting was arranged with the British Aeronautical Ministry where he presented his idea, the ministry however had a scientific advisor, Dr. A. A. Griffith, who more or less levelled Whittle's idea to the ground. Whittle received a response from the ministry; in a letter they wrote that jet engines were very unpractical devices; they were far too heavy. Furthermore, high cycle temperatures and the lack of heat resistant materials were some of the unsolved problems they claimed. The time was not mature for his invention, so Whittle let his idea rest for a while, until two former colleagues contacted Whittle and explained their interest for his jet engine idea.

Great Britain now lost the chance to take the lead in the jet engine development due to the incorrect assessment provided by A. A. Griffith. This shows the great importance that people in leading positions, must have the technical competence to correctly assess innovations in order to promote the development of breakthrough concepts.

2 Scope, Purpose and Objective of the thesis

The purpose of this thesis is to contribute in creating new ways forward for the aerospace industry to answer the society's need for greener air transport. In accordance with the ACARE 2020 research goals (Argüelles et al., 2001), the work attempts to find out and evaluate new, as well as old, ideas in reaching those stringent research goals. As an input to the thesis work, some pre-studies of innovative aero engine technologies were conducted in 2005 at Volvo Aero and Chalmers University of Technology, pointing out certain important engine technologies that should be considered (Lundbladh and Grönstedt, 2005). The objective of this thesis was also to develop a number of methods and models necessary for assessing aero engine technologies that could contribute to radical improvements in CO_2 and NO_x emissions.

The status of the research group's[1] capability to model and assess future aero engines at the time for the start of this work in the winter 2005 was confined to the following;

- Steady- and transient performance modeling and assessment of gas turbines and aero engines without the detailed connections to the aircraft application
- Engine weight and dimensions modeling
- Component design and analysis using computational fluid dynamics (CFD) and computational aero acoustics (CAA).

The system modeling capability is continuously evolving as a result of many on-going projects, master thesis project and industrial cooperation. It should be pointed out that this includes the underlying parameter assumptions as well, such as engine component efficiencies, turbine entry temperatures, metal temperatures to mention a few.

The focus of this work has been to develop the methods necessary to perform full MDO assessments of future aero engines with a particular focus on the coupling of the engines and the aircraft. Additionally, these methods where to be applied to produce and assess new research questions that would have the potential to take the aerospace industry closer to and beyond the ACARE 2020 vision.

The work performed has been limited to the conceptual design of aero engine design, meaning thermodynamic cycle optimizations including aircraft performance, engine weight and engine dimensions. This has allowed evaluating a number of solutions for minimizing emissions of CO_2, NO_x and to some extent noise.

[1] Aero-acoustics & Turbomachinery at Chalmers University of Technology, Division of Fluid Dynamics, Department of Applied Mechanics.

3 Literature Review

3.1 Research and Development Goals for the Aerospace Industry

Looking back at the introduction of the jet engine into the commercial market in the 1950s, one can conclude that the market accepted the new technology quite well despite the quite high fuel consumption. The reason for this is certainly related to the tremendous increase in flight speed offered by jet driven aircraft and the resulting decrease in travel times. Currently the internal R&D at the large engine manufacturers pushed the jet engine technology into the market which at the same time also was pulled by the need for shorter travel times. After a while, this revolutionary jet engine technology became established in the market and a new pull originated from the airline operators; the need for jet engines that generated lower operating costs. Eventually, this market pull forced the OEMs to develop more fuel efficient jet engines along an evolutionary technology path, which has led the aerospace industry to provide the market with the highly efficient turbofan engines of today.

Why put so much effort into the introduction of revolutionary technologies that will give step changes in aero engine efficiency? Is the pace of development provided by the evolutionary path of technology development not sufficient? A part of the answer lies in the intensified climate debate occurring over the last decade, another part of answer lies in profitability. The technology pull from the society has been vastly intensified in recent years after the IPCC reports (IPCC, 1990, IPCC, 1995, IPCC, 2001, IPCC, 2007) on the climate change and more specifically the impact of aviation on the global atmosphere (J.E.Penner et al., 1999). These reports states that significant reductions in greenhouse gas emissions are technically possible and can be economically feasible. This can be achieved by applying an extensive array of technologies and policy measures that accelerate technology development. An response to these reports came from European aerospace industry in the year of 2000 (Argüelles et al., 2001) setting the framework for how aerospace industry in Europe should strive to respond to society's needs. The research goals are targeting a reduction in CO_2 emissions by 50%, NO_x by 80% and noise by 50% at the year of 2020 as compared by modern technology in service in the year of 2000. These R&D goals are further evolved and projected into the year of 2050 (Darecki et al., 2011) and are quantified as a 65% reduction in CO_2, 90% reduction in NO_x and a 65% reduction in noise compared to the same baseline as the 2020 aerospace R&D vision.

3.2 European Aero Engine Research Programs

One could observe the intensified European aerospace R&D during the first decade of the 21st century by the introduction of large aero engine R&D projects such as the EEFAE ANTLE and CLEAN projects (Wells et al., 2001) within the fifth EU framework program and within the sixth framework program VITAL (Korsia and Spiegeleer, 2006) and the NEWAC (Wilfert et al., 2007) projects. Within the on-going seventh EU framework program the projects DREAM (EU, 2011b) and Clean Sky (EU, 2011a) are studying advanced aero engine technologies such as the contra-rotating and the geared open rotor engines. The DREAM consortium includes key European engine manufacturers, research institutes and SMEs. The DREAM project aims at developing technologies that will go beyond the ACARE goals in SFC, up to TRL level of 4 to 5. Then, these technologies will be candidates to be transferred into the Clean Sky project to finally reach TRL 6. Within Clean Sky, the relevant technologies needed to reach, and go beyond the environmental targets set by ACARE will finally be demonstrated on flying tests beds.

3.3 U.S. Aero Engine Research Programs

Like many countries in the western world, the U.S. is dependent on foreign energy suppliers. This relationship became evident during the OPEC oil embargo in the winter of 1973 and 1974. As a consequence of this, and on a direct request from the U.S. congress in 1975, NASA initiated the aircraft energy efficiency (ACEE) research program (Aiken and Petersen, 1982) with the aim of reducing fuel consumption of commercial subsonic air transport (DeGeorge, 1988). Except for aircraft related fields of research, three different engine programs were initiated, these were the engine component improvement program, ECI, the energy efficient engine, E^3, program (Ciepluch et al., 1987) and the advanced turboprop program, ATP (Whitlow and Sievers, 1984).

3.3.1 Engine Component Improvement Project (1976 – 1982)

According to the project statement of the ECI program, from December 1976, the main objectives were to (Bowles, 2010);

(1) *"Develop performance improvement and retention concepts which will be incorporated into new production of the existing engines by the 1980-1982 time period and which would have a fuel savings goal of 5 percent over the life of these engines, and*
(2) *To provide technology which can be used to minimize the performance degradation of current and future engines."*

At the time being, in the mid and late 1970s, there were four major engines powering all commercial jet-driven aviation in the U.S.; GE CF6, P&W JT8D, JT9D and JT3D. The project came to focus on developing fuel-saving techniques for three of these four engines, since the JT3D was considered aged at the time. It is said that during the project, there were some problems in the relationship between GE and P&W mainly because of the fact that P&W had dominated the market for commercial aero engines since the end of world war II but in the late 1970s P&W started to lose market shares to GE (CF6 vs. JT9D). The project simply supported technology improvements to two competitors that had to collaborate.

Technologically the project was divided into two different, but interrelated subprojects, the performance improvement- and engine diagnostics programs. The performance improvement program came down, after a final review by NASA, to 16 technology improvement concepts that were to be further developed. Examples of the most important technology areas of these concepts were active clearance control in the turbines, aerodynamics of compressors and turbines, thermal barrier coatings in the turbines (McAulay, 1980).

The ECI program is considered one of the most successful programs within the ACEE program since it achieved the fuel savings of 5% claimed in the project statement, in a very short time frame, and the technology improvements were brought into service rapidly.

3.3.2 Energy Efficient Engine Program (197X-198X)

The E^3 project goals took into account fuel savings, economic and environmental improvements, and were defined as (Ciepluch et al., 1987);

- Reduce SFC by 12%
- Reduce SFC performance deterioration by 50%
- Reduce direct operating costs by 5%

- Meet FAA noise regulations
- Meet EPA proposed emissions standards

These goals were defined using the turbofans in service for widebody aircraft in the late 1970s as baseline, i.e. GE CF6 and P&W JT9D. The project also included Boeing, Douglas and Lockheed as well as the airlines Pan American and Eastern airlines in order to discuss engine design options and to receive operational expertise. The overall project goal was to have a new turbofan engine ready for commercial use in the late 1980s or early 1990s. Both GE and P&W were given the mission to design and build a new turbofan engine, the E^3 engine, but not designed for commercial-ready-to-use, but for proof-of-concept testing, a technology demonstrator.

GE completed the program successfully in 1983 and they reported a 13% reduction in SFC compared to the CF6 engine. GE incorporated several technologies developed within the E^3 ACEE program in the GE90 engine which was the first engine to use fan composite blades allowing for a 800 lb weight reduction, produced 60% less emissions of nitrogen oxide and was quieter.

P&W also had success with the E^3 program but not until 2007 when the geared turbofan engine was presented for the Mitsubishi regional jet, with much of the results from the P&W participation within the ACEE programs incorporated.

3.3.3 Advanced Turboprop Project (19XX-19XX)

While starting as a small-scale propeller project in a collaboration between NASA Lewis and Hamilton Standard, the last large propeller manufacturer in the U.S., the project continued as a huge research project involving both NASA, the engine manufacturers P&W, GE and Allison as well as the aircraft manufacturers Boeing, Lockheed and McDonnell-Douglas. At some time the project involved all four NASA research centers; Lewis, Langley, Dryden and Ames, 40 industrial contracts and 15 university grants.

Technically, after initial studies by Boeing, Lockheed and McDonnell-Douglas, four ares of concern was pointed out; propeller efficiency at cruise, internal and external noise levels, aircraft installation aerodynamics and maintenance costs. The project had four technical stages; conceptual development from 1976 to 1978, enabling technology from 1978 to 1980, large scale integration from 1981 to 1987 and flight research during 1987.

During the project two different concepts were studied in detail, the single-rotating propeller and the counter-rotating propeller. NASA worked on the single-rotating turboprop together with P&W, Allison and Hamilton Standard, while GE worked on their own with the counter-rotating turboprop, or Unducted Fan (UDF) as they called it. The single-rotating puller configuration involved a relatively complex gearbox and in contrast to this design GE developed a gearless pusher design. The UDF was flight tested on a B727 in 1986 and the single-rotating turboprop was tested on a Gulfstream II in 1987.

The project showed by studies, scale model tests and flight tests that turboprop engines with propellers utilizing thin, swept, highly loaded blades could contribute to a +20% reduction in fuel consumption compared to equally advanced turbofan engines. The reason for the technology to enter service is claimed to be the fact that the fuel price came down to normal levels at the end of the 1980s.

In particular the ATP program investigated the potential benefits of utilizing propeller technologies for high speed transport in terms of the LAP project (DeGeorge, 1988) and the Unducted Fan (UDF) project (GE, 1987b, GE, 1987c, GE, 1987a). At the end of the 1980s, after successful flight demonstrations of the LAP and the UDF engines, the interest to introduce new innovative engines on the market declined as the fuel prices came down to historically normal levels.

3.3.4 Ultra-Efficient Engine Technology (2000-2005)

Starting in 2000, the project focused on technology development in six technology areas; low emissions combustion, highly loaded turbomachinery, high temperature materials and structures, intelligent propulsion controls, propulsion-airframe integration, and integrated component technology demonstrations at TRL 3 to 5. The program objectives of the UEET project was to (Shaw, 2000);

(1) *Demonstrate propulsion technologies that enable fuel burn reductions of up to 15%, and*
(2) *Combustor technologies (configuration and materials) that enable LTO NO_x reductions of 70% relative ICAO 1996 standards.*

The project ended in 2005, claimed to have met the project objectives at TRL 4.

3.3.5 NASA N+3 NRA (2007-

N+3 NRA is the short version for "Advanced Concept Studies for Subsonic and Supersonic Commercial Transports Entering Service in the 2030-2035 period". The overall project objectives are to;

(1) *Development of prediction and analysis tools for reduced uncertainty in design process.*
(2) *Development of concepts/technologies for enabling dramatic improvements in noise, emissions and performance characteristics of subsonic/transonic aircraft.*

The specific technology goals are presented in Table 1. Technology goals for the NASA Subsonic Fixed Wing Aircraft..

Corners of the trade space	N+1 (EIS 2015)[2], Conventional Configurations, Relative 1998 Single-Aisle Aircraft, i.e. B737/CFM56	N+2 (IOC 2020)[2], Unconventional Configurations, Relative 1997 Twin-Aisle Aircraft, i.e. B777/GE90	N+3 (EIS 2030-2035)[2], Advanced Aircraft Concepts, Relative 2005 Technology Baseline
Noise	-32 dB (cum. below stage 4)	-42 dB (cum. below stage 4)	-71 dB (cum. below stage 4)
LTO NO_x Emissions (below CAEP 6)	-60%	-75%	better than -75%
Performance Aircraft Fuel Burn	-33%[3]	-40%[3]	better than -70%[3]
Performance Field Length	-33%	-50%	exploit metro-plex[4] concepts

Table 1. Technology goals for the NASA Subsonic Fixed Wing Aircraft.

[2] TRL range: 4-6
[3] Additional 10% improvement be may added due to operational capability improvements
[4] Concepts that enable the optimal use of runways at multiple airports within the metropolitan areas

Phase 1 of the project (2008-2010) included 6 teams studying advanced concepts realizing the N+3 aircraft. The team leaders were Northrop Grumman, Boeing (two projects), Massachusetts Institute of Technology, Lockheed Martin and GE Aviation (NASA, 2008a) .

Some interesting results and concepts have been developed and reported in open literature. The team led by MIT and also including Aerodyne Research, Aurora Flight Sciences, and Pratt & Whitney presented a radical aircraft concept called the “double bubble” which is predicted to meet the N+3 technology goals (Greitzer et al., 2010). This concept is one of the concepts that has been chosen to be further evaluated in Phase II of the project (Croft, 2011). In short the concept utilizes all composite materials for the airframe structure, Natural Laminar Flow, BPR 20 engines, boundary-layer ingestion, a maximum allowed turbine metal temperature of 1500 K, advanced LDI combustor technology to mention a few.

Except for the MIT concept, three other teams have been granted further studies in phase II. Boeing will continue to work on its truss-braced wing and hybrid electric powered subsonic ultra green research design (SUGAR) (Bradley et al., 2010). In addition to studying light weight materials and engine concepts, Boeing will design and test wind tunnel- and computer models of the airplane.

Cessna Aircraft will continue to develop and test a new protective skin for the airframe that would help protect the aircraft from lightning electromagnetic interference, extreme temperatures and object impacts (D’Angelo et al., 2010).

Northrop Grumman will continue developing wing leading edge high-lift devices (Bruner et al., 2010).

3.3.6 Military Research Relevant for Commercial Applications

Of great interest for the commercial aircraft industry are the IHPTET (1988-2005) and VAATE (1999-) research programs. The IPHTET (Integrated High Performance Turbine Engine Technology (IHPTET) program was a joint effort of DoD, NASA and the industry to provide revolutionary performance and operational improvements for current and future military engines. The broad research objective was to double the propulsion capacity of turbomachinery at the year of 2000 without compromising the safety, reliability or maintainability of the current propulsion systems (AIAA, 1991). At the end of the project it was demonstrated a 70% increase in thrust-to-weight, +60ºF combustor inlet temperature capacity, a 32% production cost reduction and a 31% maintenance cost reduction at TRL 6 (EICKMANN et al., 2006). Some of the reasons for IHPTET to be considered successful is said to be because of it addresses defense critical technologies, its dual use, its well defined goals, objectives and milestones and its integration of a variety of disciplines. Some of the advanced technologies that are in use or close to enter service is the super-cruise capability of the F-22 and the vertical lift capability of the STOVL-version of F-35.

The VAATE (Versatile Affordable Advanced Turbine Engines) project started in 1999 and is planned to end in 2017 (AIAA, 2006). The overall project objectives of VAATE are defined as;

- 200% increase in engine thrust-to-weight ratio (a key jet engine design parameter)
- 25% reduction in engine fuel consumption (and thus fuel cost)
- 60% reduction in engine development, procurement, and life cycle maintenance cost

Like IHPTET, this project is also a joint effort between DoD, NASA and Industry and three main areas of interest will be studied during the project; versatile core, intelligent engine and durability.

3.3.7 Miscellaneous work

Between 2003 and 2006 a collaboration project between the U.S. and Europe took place, in particular the Cambridge University in the U.K. and Massachusetts Institute of Technology in the U.S. led a project called the Silent Aircraft Initiative which also included several partners from academia and the industry (Dowling and Hynes, 2006, SAI, 2006). The silent aircraft initiative aimed for an aircraft optimized for minimum noise in the 2030 timeframe. A conceptual design, SAX-40, has been presented that is predicted to generate noise levels 25 dB lower than current aircraft.

3.4 Recent Engine Technology Advancements

The aero engine technologies that will have a potentially important impact on the aerospace society's ability to achieve or even go beyond the ACARE targets are numerous. The most important scientific contributions relating to a number of key technologies relevant to this thesis are summarized in this chapter. The important areas of improvement are divided into high bypass ratio engines (or increasing propulsive efficiency), novel cycles, evolutionary improvements, miscellaneous improvements and combustor technologies for ultra-low emissions.

3.4.1 On-going and Recent Work on High Bypass-ratio Engines

Within the Clean Sky project the research and development of the open rotor engines has had its revival. Counter-rotating open-rotor demonstration engines are being developed at the moment with the aim of conducting flight test manifesting the technology at TRL 6 in the year of 2015. The overall objective is to show a -20% fuel burn benefit compared to modern engines in service at the year of 2000 (ACARE goal).

The GTF cycle has been studied and presented in several publications (Riegler and Bichlmaier, Kurzke, 2009) and although it is clear that the introduction of the fan gearbox system will decouple the low-pressure components allowing for more independent fan and LPT design optimizations, it is not easily quantified to what extent, if any, the GTF will be more fuel efficient than its equally advanced conventional turbofan counterpart.

The GTF engine for the regional jet market is getting closer to entry into service as the Pratt & Whitney developed PW1524G has entered the flight testing phase (Pratt&Whitney, 2011). It is claimed that the fuel burn benefit will be 16% compared to today's engines in service.
The CFM consortium, i.e. GE and Snecma, will develop a high-BPR engine called Leap-X (CFM, 2011) without a mechanical fan gearbox but using comparable technology. CFM discusses 16% benefit in fuel consumption over today's engines in service as well.

3.4.2 Recent Studies of Novel Cycles

Intercooled engines with- and without recuperators has been discussed since the early days of jet propulsion. A quite recent study by Lundbladh and Sjunnesson compares intercooled and recuperated engines with conventional technology (Lundbladh and Sjunnesson, 2003). The study shows that recuperation alone will not give any benefits in terms of fuel burn or operating costs, while the intercooled engine could give a 6% benefit over the conventional cycle in terms of fuel burn. It was also concluded that the intercooled-recuperated (IRA) engine could provide fuel efficiency reductions, however in terms of direct operating costs the cycle did not provide any clear benefits. Also worth noting is that the study assumed a fixed LPT.

A quite comprehensive study of the IRA engine was presented by MTU in 2004 (Boggia and Rüd, 2004) which included cycle optimizations and preliminary design studies of the various sub-systems such as the heat-exchanger and the recuperator. The final cycle was an OPR 30 three-shaft geared turbofan with a variable LPT. The study showed an 18.7% reduction in SFC compared to a conventional BPR 5 turbofan engine of 1995 standard, and a 60% NO_x margin to the ICAO-96 standard. Furthermore, the complexity of the cycle, possible life and reliability issues are commented. In the study by Kyprianidis and Grönstedt (Kyprianidis et al., 2011) potential benefits of the same order are reported.

For a wider discussion of heat-exchanger technologies an extensive study performed by McDonald analyzes the application and potential benefits of recuperation in aero engines in general (McDonald et al., 2008a, McDonald et al., 2008b, McDonald et al., 2008c).

Reheated aero engines, as is the case with a majority of many current suggestions on radical changes to the turbofan engine, has been studied in the past. However, recent studies concerning inter-turbine reheat, especially for commercial subsonic transport applications, have received very little attention. In 1976, NASA presented a contractor report that concentrated on investigating unconventional aircraft engines for ultra low energy consumption (Gray, 1976). In this report inter-turbine reheat applied to a two-spool turbofan was investigated among several other technologies. Except for the conclusion of a higher power output for the reheated turbofan the author states that “adding reheat to the Brayton cycle increases the average temperature during heat addition but increases the average temperature of heat rejection even more…”. The increased requirement for turbine cooling air resulted in an SFC penalty of some 8% compared to their conventional engine cycle (two-spool turbofan). They did not proceed with any more detailed studies of the reheated engine as it was determined that even a 100% engine weight saving could not offset the large SFC penalty in terms of the fuel savings potential. In this work it is argued, that the two spool ITB configuration studied within the work by Gray does not allow the introduction of the ITB sufficiently early in the expansion in order to achieve a high efficiency cycle.

In 2001, Liu and Sirignano presented a detailed performance study of inter-turbine reheated turbojets and turbofans (Liu and Sirgnano, 2001). They investigated both discrete inter-turbine engines (one and two inter-stage burners) and continuous inter-turbine engines (CTB). Their studies involved analytical design equations using constant gas properties and the analysis did not include the effect of engine weight and nacelle drag and their relation to the complete mission optimization. Furthermore, the modeling of the LP-turbine cooling which, as indicated in the NASA study by Gray, could be a potential show stopper is missing in their study. However, as the authors state, they were presenting a proof-of-concept of the ITB and CTB engine configurations. They showed, among other things, the existence of a maximum thermal efficiency as a function of power split between HP- and LP turbines. They also showed that ITB and CTB engines benefit more from higher bypass-ratios than their conventional counterparts. Furthermore, they demonstrate that at the very low turbine inlet temperatures where the conventional engines fail to work properly the inter-turbine reheated engines worked very well. For the turbofan engine configuration under study it was shown that for the entire subsonic flight range the one-stage ITB turbofan had up to 50% higher specific thrust, incurring an SFC penalty in the range of 10-15% depending on the cycle definition.

3.4.3 Combustion Technologies for Ultra-Low Emissions

Between 1972 and 1976, the Experimental Clean Combustor Program (ECCP) (Roberts et al., 1977, Gleason and Bahr, 1979), the first major NASA led effort to develop low-emission combustor technology was executed. The project included the major engine manufacturers P&W and GE. The program primary objectives were to;

(1) The generation of combustor system technology required to develop advanced commercial aircraft engines with lower exhaust pollutant emissions than those of current technology engines, and
(2) The demonstration of the pollutant emission reductions and acceptable performance in a full-scale engine in 1976.

More specifically, the technical goals for P&W were to develop technology that would provide a 54% reduction of NO_x emissions, a 59% reduction in CO and a 83% reduction in UHC emissions compared to the exiting baseline JT9D-7 combustor.

For GE, the technical goals were defined as; a reduction of NO_x emissions by 61%, a reduction of CO emissions by 71% and a reduction of UHC emissions by 90% compared to their baseline CF6-50C.

P&W developed the Vorbix (two-stage vortex burning and mixing) combustor that were reported very successful in terms of pollutant emissions reductions, NO_x emissions were reported 10% below the project goal, CO emissions were 26% below the project goal and UHC was reported 75% below the project goal. Compared to the baseline, JT9D-7 combustor, the NO_x emissions were reduced by 58%, CO emissions were reduced by 69% and UHC were reduced by 96%. Furthermore, there was a smoke number objective that was not fulfilled (Roberts et al., 1977).

GE developed the double-annular combustor (DAC) that did not quite meet the stringent project emission goals, especially for emissions of NO_x. It is proposed in the final report that the NO_x emissions target could be met by applying a revised NO_x standard allowing higher NO_x levels for engines with pressure ratios above 25. The P&W baseline JT9D-7 had a pressure ratio around 23 while the CF6-50C had a pressure ratio of about 30.

4 The Energy Efficiency of Aviation

The whole idea of transportation is to bring items or people, i.e. payload, to its destination. It is desirable to do this in an optimal way. The term optimal should in this context be understood as the best possible means of transport in terms of cost, time, comfort, safety, environmental impact or a combination thereof. In a simplified manner, the optimum could be defined with only one of those measures, e.g. the lowest cost, or the quickest way of moving people or items between two locations. In practice, the preferred transport solution is more complicated than that, it constitutes a well balanced solution that to some extent offers all of these properties.

In recent years the focus has shifted from the old aerospace design paradigm "higher and faster" to greener airplane designs, i.e. the focus has shifted more or less from travel time to environmental impact. The environmental impact can be quantified in terms of CO_2, NO_x and noise emissions generated by the aircraft and engine(s). For instance in the year of 2000, the Advisory Council of Aeronautics research in Europe (ACARE) defined a vision for the European aerospace industry to work towards a 50 % CO_2 reduction, a 80 % NO_x reduction, and a 50 % noise reduction to be achieved by the year 2020 (Argüelles et al., 2001).

To be able to assess any improvements in aviation efficiency one must be clear of the meaning of efficiency related to airplanes. As mentioned above, the very purpose of air transport, or any means of transport, is to deliver people and/or payload from one destination to another. The aircraft produces useful output in terms of moving a given mass (payload) a certain distance. The energy required to produce that output is taken from the chemically stored energy in the aircraft fuel, translated into mechanical work and ultimately thrust, by the use of a suitable heat engine. One realizes that a direct measure of the air transport output can be described as the air transport output produced per unit fuel energy consumed according to equation (1),

$$\text{specific air transport output} = \frac{payload \times block\ distance}{m_{fuel} \times LHV} \qquad (1)$$

The reciprocal of equation (1) is called energy intensity, E_I as defined by Lee (Lee et al., 2001), and is exemplified in Figure 4 for a number of modern, and historical aircraft (Bridgeman, 1948, Bridgeman, 1953, Jackson, 2005, Boeing, 2011). Note that the payload term in equation (1) can consist of cargo, luggage, passengers or combinations thereof. For passenger transports however, the transport output is frequently given as revenue passenger kilometers, *RPK* (number of passengers multiplied by block distance), or available seat kilometers, *ASK* (number of seats multiplied by block distance). Furthermore the relation between *RPK* and *ASK* is called the load factor and is a measure of the utilization of the aircraft capacity. Noticeable in Figure 4 is the fact that the most efficient piston driven aircraft, here illustrated by the Lockheed L-1049 Super Constellation show approximately the same energy intensity, close to 1 MJ/ASK, as the modern jet aircraft investigated here. Is it then true that the technology development achieved nothing in terms of energy efficiency during 75 years of technology development?

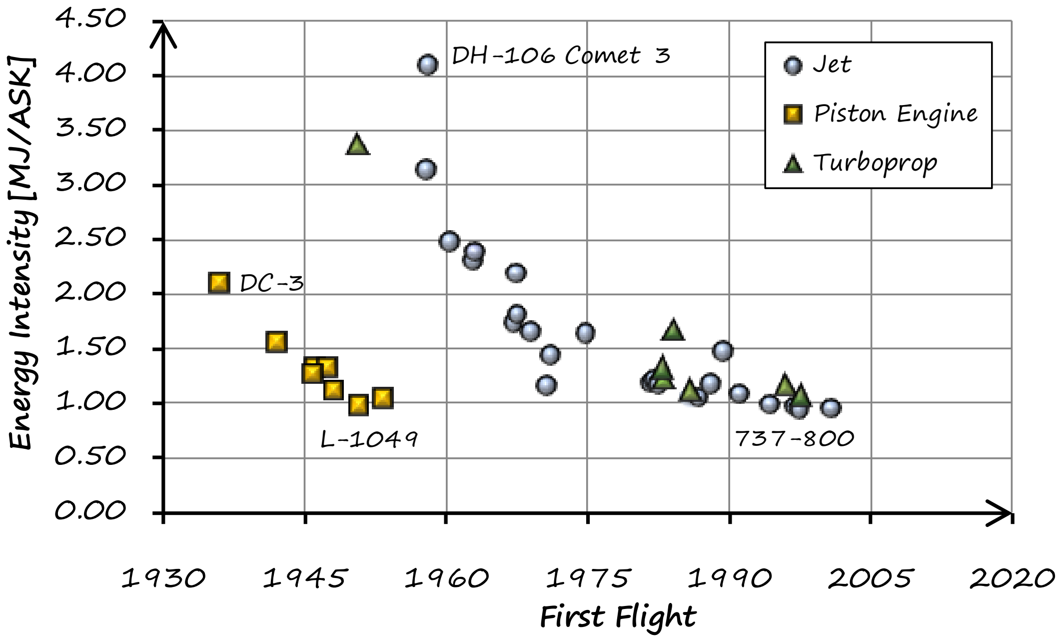


Figure 4. Historical energy efficiency trend for commercial transport aircraft.

The air transport output measure as described by equation (1) is indicative but not complete if one also takes the value of people's time into account. The numerator of equation (1) is in some contexts misleadingly described as transport productivity (Martinez-Val et al., 2005), but according to the author's opinion productivity should also involve a measure of time and reveal how fast as well as efficient a certain transport process is completed. It is suggested that an adequate measure of air transport productivity therefore is,

$$\text{air transport productivity} = payload \times block\ distance \times speed \qquad (2)$$

with units of tonne-kilometers/hour or passenger-kilometers/hours. Consequently equation (1) is now re-written as,

$$\text{specific air transport productivity} = \frac{ASK \times V_{flight}}{m_{fuel} \times LHV} \qquad (3)$$

This equation also reveals some of the progress made in the last 75 years, and is illustrated in Figure 5.

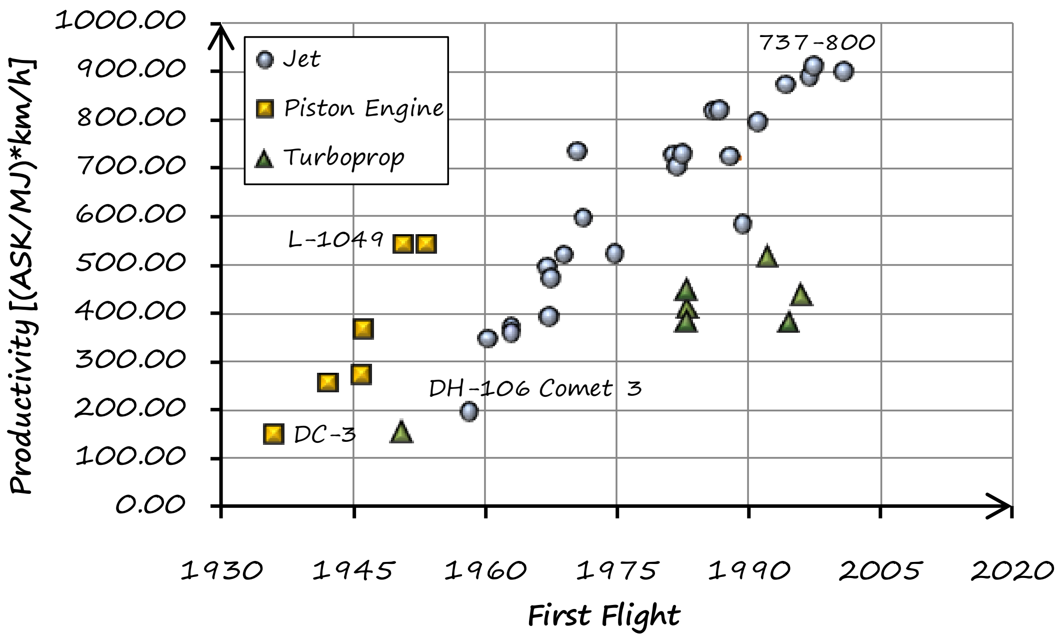


Figure 5. Historical productivity trend for commercial transport aircraft.

If one instead of the absolute fuel consumed in the denominator of equation (3) considers the rate at which the chemical energy in the fuel is converted into heat, i.e. $\dot{m}_{fuel} \times LHV$ and rewrites this expression with use of the definition of SFC so that $\dot{m}_{fuel} \times LHV = SFC \times T \times LHV$ a useful expression for air transport energy efficiency valid for unaccelerated, steady level-flight can be obtained according to,

$$\text{air transport efficiency} = \frac{payload \times V_{flight} \times L/D}{SFC \times W \times LHV} \qquad (4)$$

This expression summarizes the different factors affecting overall air transport efficiency for any aircraft. It reveals that to maximize efficiency we want to maximize aerodynamic efficiency, L/D, of the aircraft, the amount of payload that can be contained and the speed by which the flight takes place. At the same time it is highly desirable to minimize the engine SFC and the overall weight of the aircraft and engines. These factors are interrelated in a complex way and the overall design goal must be to find the most well-balanced design solution that maximizes air transport efficiency. A further break-down of air transport efficiency and how to improve it is undertaken in the following sections.

4.1 Specific Fuel Consumption

Specific fuel consumption is defined as fuel flow divided by net thrust according to,

$$SFC = \frac{\dot{m}_{fuel}}{T_N} \qquad (5)$$

The definition of SFC is commonly used and constitutes a simple way of comparing different aero engines in various thrust classes. The SFC variable expresses how much fuel that is required per unit of net thrust delivered by the engine. The SFC measure is directly related to overall engine efficiency and the relation is easily established from the definition of overall engine efficiency. Consider, as above, the net energy input to the aircraft, i.e. fuel energy, or more specifically the rate of which fuel energy is converted to heat. The output from the aero engine is the propulsive power delivered. The overall engine efficiency is then,

$$\eta_o = \frac{T_N \times V_{flight}}{\dot{m}_{fuel} \times LHV} \qquad (6)$$

and combined with equation (5) one gets,

$$\eta_o = \frac{V_{flight}}{SFC \times LHV} \qquad (7)$$

The SFC is thus directly related to overall engine efficiency which raises the need for establishing the foundations of the energy conversion process, i.e. the process from chemical energy contained in the fuel all the way to the thrust delivered to the aircraft.

For air-breathing engines, the fuel is oxidized to release heat with the help of air flowing through the engine, in particular the oxygen contained in the air. In the case of an ideal combustor the combustion process is said to be complete, meaning that the heat released in the combustor directly relates the heating value of the fuel. In reality, the combustion will incorporate losses due to various reasons, so that the efficiency of the energy conversion (accurate enough for combustion temperatures below approximately 1650 K) can be expressed as,

$$\eta_{comb} = \frac{\dot{m}\Delta h}{\dot{m}_{fuel} \times LHV} \qquad (8)$$

Note that it is general practice for aero engines to use the lower heating value, *LHV*, in contrast to the higher heating value, *HHV*, since the difference, $\Delta H_{vap,H2O}$,is the heat of vaporization for the water content and will not be recovered as long as the engine process does not incorporate a condenser. Therefore, if different fuel types are to be compared it will make most sense to subtract the heat of vaporization for water.

The thermal efficiency is defined as the ratio of the kinetic energy increase of the air going through the engine, to the thermal energy released by burning the fuel according to,

$$\eta_{th} = \frac{\dot{m}\left(V_{jet}^2 - V_{flight}^2\right)}{2 \times \dot{m}_{fuel} \times LHV} \qquad (9)$$

Finally, the efficiency at which the kinetic energy of the gases leaving the engine is converted to propulsive power is expressed by the ratio of propulsive power and kinetic energy. Assuming that the flow is fully expanded to a single velocity, V_{jet}, we get:

$$\eta_p = \frac{\dot{m}\left(V_{jet} - V_{flight}\right)V_{flight}}{\frac{1}{2}\dot{m}\left({V_{jet}}^2 - {V_{flight}}^2\right)} \qquad (10)$$

which can be re-written as,

$$\eta_p = \frac{2}{1 + \frac{V_{jet}}{V_{flight}}} \qquad (11)$$

Equation (11) is the well-known Froude equation for propulsive efficiency. This expression it is very important for the foundations of designing energy efficient aero engines because it shows the importance of achieving low exhaust velocities. This is because the residual exhaust velocity, jet velocity in excess of flight speed, is a direct measure of propulsive inefficiencies. The theoretical limit of propulsive efficiency is achieved when the jet velocity approaches the flight velocity. Unfortunately, the net thrust approaches zero as the jet velocity approaches the flight velocity since $T_N = \dot{m}\left(V_{jet} - V_{flight}\right)$, and the engine size approaches infinity due to the same reason.

Since modern jet engines for civil air transport utilizes two jet streams, it is logical to introduce the transmission efficiency, η_{tr}, which is a measure of the efficiency of the energy transfer from the core exit to the kinetic energy in the exhaust streams. The thermal efficiency is now divided into transfer efficiency and core efficiency according to,

$$\eta_{th} = \eta_{core} \times \eta_{tr} \qquad (12)$$

The transfer efficiency is equal to,

$$\eta_{tr} = \eta_{Fan} \times \eta_{LPT} \times \eta_{nozzle} \times \eta_{mech} \qquad (13)$$

This efficiency reflects all the losses incorporated when the energy is transferred from the core exit to the nozzle exit plane; low pressure turbine losses, fan losses, duct losses, losses in the nozzles and if present mechanical gear box losses.

Furthermore, the core efficiency is defined as,

$$\eta_{core} = \frac{\dot{m} \times h_{core,exit}}{\dot{m}_f \times LHV} \qquad (14)$$

where $h_{core,exit}$ is the energy available at the core exit, i.e. the point during the expansion when all power requirements for core stream compression are fulfilled. A detailed explanation of the definition of core exit in a turbofan see for instance Kurzke (Kurzke, 2007).

The overall efficiency can now be expressed as,

$$\eta_o = \eta_{core} \times \eta_{tr} \times \eta_p \qquad (15)$$

and finally, the SFC as,

$$SFC = \frac{V_{flight}}{\eta_{core} \times \eta_{tr} \times \eta_p \times LHV} \qquad (16)$$

Equation (16) relates the relevant efficiencies incorporated in conceptual and preliminary design to the specific fuel consumption.

Figure 6 shows the jet engine development quantified in terms of SFC as a function the entry into service year, starting at the introduction of the jet engines during the 1950s continuing with the first turbofans during the 1960s and 1970s followed by the current state of the art high bypass ratio turbofan technology (Gunston, 2005, NASA, 2008b). The impact of technology revolutions are observed as step change improvements. Estimating the development trends for the current high bypass ratio engine technology leads to the conclusion that a new technology breakthrough is necessary to reach the ACARE 2020 aero engine vision.

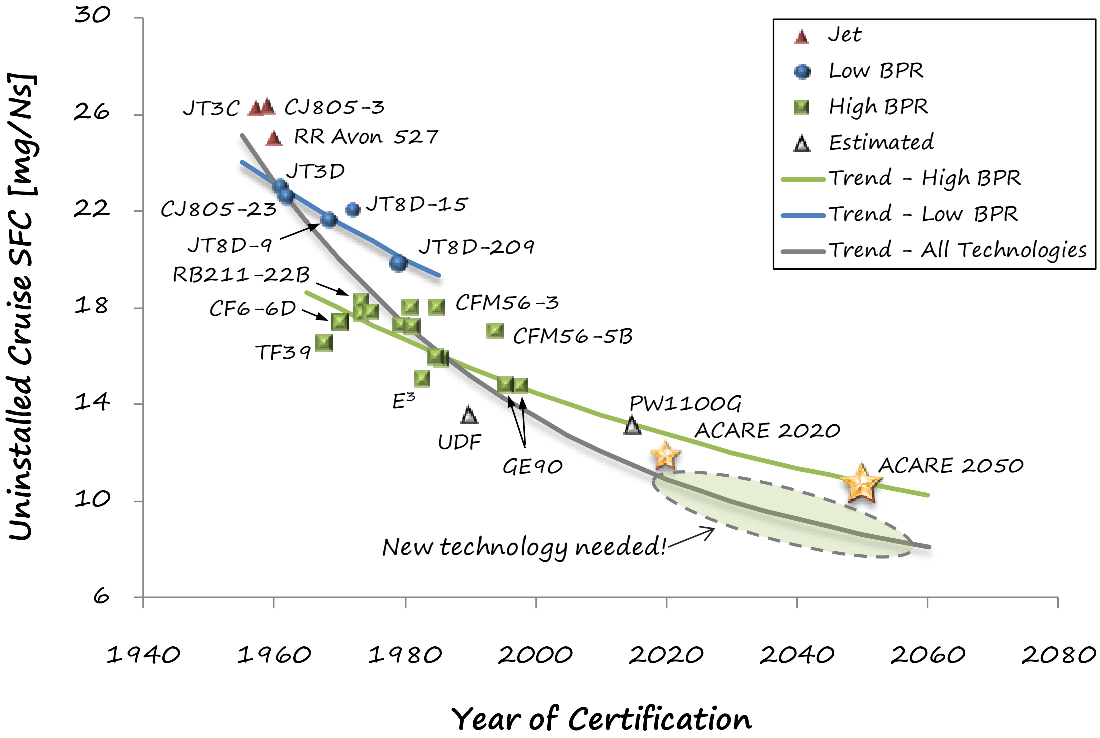


Figure 6. Trends of jet engine technology development quantified in terms of uninstalled cruise SFC as a function of year of certification.

5 Aircraft Performance

The whole problem is confined within these limits: to make a surface support a given weight by the application of power to the resistance of air.

Sir George Cayley (1809)

An aircraft in translational motion can be described according to,

$$T \times cos\alpha_T - D - W \times sin\theta = m\frac{dV}{dt} \qquad (17)$$

$$L + T \times sin\alpha_T - W \times cos\theta = m\frac{V^2}{R} \qquad (18)$$

Consider the case of unaccelerated level-flight. Level-flight means that the aircraft is moving parallel to the horizon and therefore $\theta = 0$. Furthermore, unaccelerated means that the right hand sides of equation (17) and (18) are equal to zero. Also for most cases, the angle α_T is small enough to allow approximating $cos\alpha_T \approx 1$ and $sin\alpha_T \approx 0$. Now, equations (17) and (18) becomes,

$$T = D \qquad (19)$$

$$L = W \qquad (20)$$

Equations (19) and (20) are the equations for unaccelerated level-flight and will be referred to at several instances of this text. In words, the drag is balanced by the engine thrust, and the weight of the aircraft is balanced by the lift. Elementary, but very important.

6 Simple Methods for Modeling Aero Engines and Quantifying Trends in Technology Development

6.1 Propulsive Efficiency

The limiting theoretical performance for the propulsor technology can be derived from simple momentum theory. Consider a stream tube of air according to Figure 7. The stagnation pressure remains constant and equal to the far upstream conditions, denoted as ∞, all the way up to the propeller disc location denoted as 1. It then changes discontinuously at the propeller disc, from station 1 to 2, as work is imparted to the stream by the propeller, and then remains constant again far downstream of the propeller disc. At some distance downstream of the propeller disc, the static pressure has returned to ambient pressure but the final wake velocity, V_e, is higher than the free stream velocity. This residual exhaust velocity creates the thrust and it is also directly related to a reduction in propulsive efficiency.

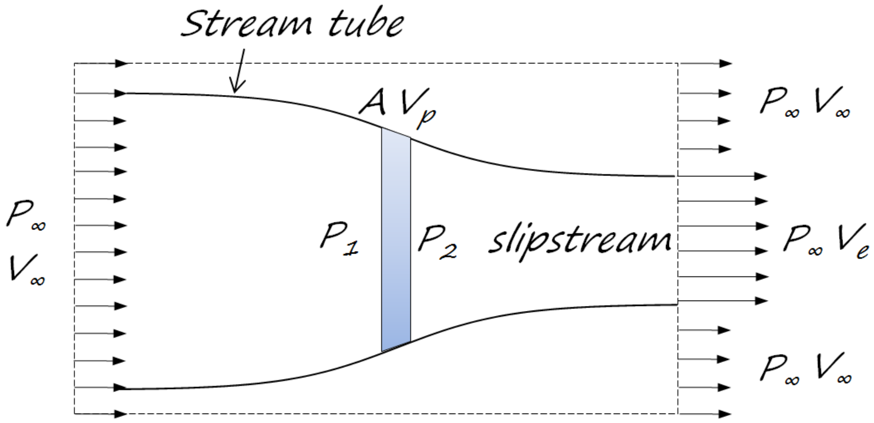
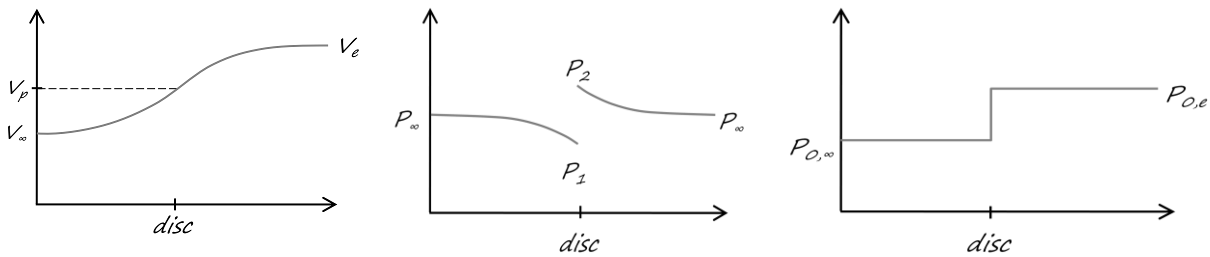


Figure 7. Schematic view of the actuator disc model of propeller performance.



Furthermore, the actuator disc propeller model assumes an infinite number of blades and the flow is assumed to be irrotational and incompressible. Based on these assumptions it is possible to compute the ideal propeller efficiency.

The thrust developed by the propeller is the pressure loading of the propeller disc over the disc area A, i.e.

$$T = (P_2 - P_1)A \tag{21}$$

Furthermore, the thrust developed can be determined by computing the momentum change over the control surface according to,

$$T = \dot{m}(V_e - V_\infty) = \rho A V_p (V_e - V_\infty) \tag{22}$$

Applying Bernoulli's equation up- and downstream of the propeller disc gives,

$$P_1 = P_\infty + \frac{\rho}{2}\left(V_\infty^2 - V_p^2\right) \tag{23}$$

$$P_2 = P_\infty + \frac{\rho}{2}\left(V_e^2 - V_p^2\right) \tag{24}$$

The pressure difference is then,

$$P_2 - P_1 = \frac{\rho}{2}(V_e^2 - V_\infty^2) \tag{25}$$

From above, equation (25) (22) and (21) yields,

$$V_p = \frac{V_e + V_\infty}{2} \tag{26}$$

The jet power added to the flow by the propeller is,

$$P_{jet} = \frac{\dot{m}}{2}(V_e^2 - V_\infty^2) = TV_p \tag{27}$$

The ideal power available to the aircraft is also called the propulsive power, $P = TV_\infty$, so the efficiency at which the propulsive power is created is computed according to,

$$\eta_p = \frac{P}{P_{jet}} = 1 - \frac{P_{loss}}{P_{jet}} = \frac{TV_\infty}{TV_p} = \frac{V_\infty}{V_p} = \frac{2V_\infty}{V_\infty + V_e} = \frac{2}{1 + \frac{V_e}{V_\infty}} \tag{28}$$

which is the Froude efficiency derived earlier in equation (11) and is a direct measure of the kinetic energy losses (induced velocity) associated with thrust production as described above.

For a fan pressure ratio approaching 1.0 the induced velocity is approaching zero and the ideal efficiency is close to 100%. The ideal propulsion efficiency should be considered a theoretical upper limit for different engine technologies. In real cases, the flow is viscous, compressible, rotational and the propeller or fan will be designed with a finite number of blades. The real flow effects will lower the efficiency through a multitude of loss mechanisms. However, for comparison of different engine technologies the ideal efficiency indicates what is possible to gain when lowering the fan pressure ratio, and as a direct consequence thereof increasing the fan diameter.

As can be seen in Figure 8, today's jet engines, i.e. turbofans, are designed for a cruise Mach number of 0.78-0.85 depending on the particular airplane under study. Typical fan pressure ratios are in the range of 1.5 to 1.75. The corresponding ideal propulsion efficiency is some 82% for these engines. Furthermore, the geared turbofans under development are expected to be designed for fan pressure ratios in the range of 1.3 – 1.45 (Riegler and Bichlmaier) giving some 5% higher propulsive efficiency than current turbofans in service.

The open rotor engines however, will probably have fan pressure ratios close to 1.05 – 1.10 which gives an ideal propulsion efficiency of 97% obtained at a somewhat lower cruise Mach number of 0.75. Since the airplane fuelburn is directly related to SFC, which is directly related to efficiency, this can be translated to fuelburn savings in the range of 20% stemming from the higher propulsive efficiency as a consequence of the chosen fan configuration.

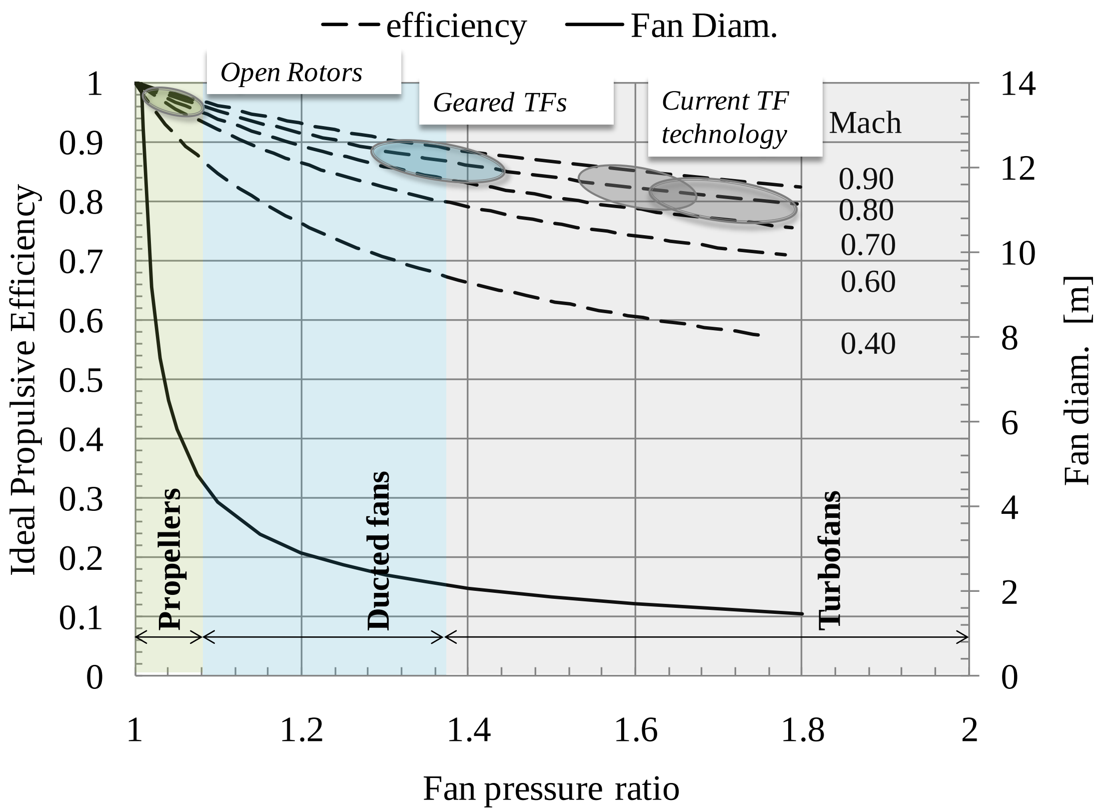


Figure 8. Ideal propulsion efficiency at infinite BPR as a function of fan pressure ratio and Mach number at a flight altitude of 35000 ft. Also shown in the figure is the actuator disc diameter needed to produce 30000 lbf of static thrust at ISA conditions. Note that this diagram is valid for the actuator disc model described in this section.

6.2 Thermal Efficiency

Consider the TS-diagram in Figure 9. The ideal Brayton cycle is represented by the thermodynamic states a-b-c-d-a. Isentropic compression takes place between a and b, followed by isobaric heat addition between b and c, isentropic expansion between states c and d. The cycle is then closed by isobaric heat rejection to the surroundings between d and a.

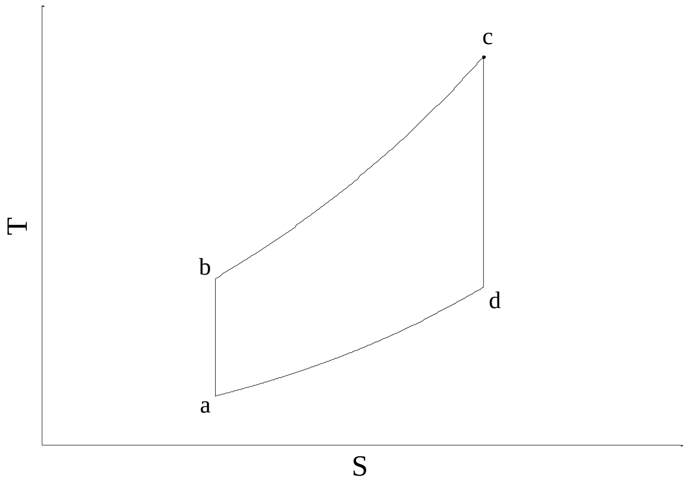


Figure 9. Ideal Brayton cycle.

The thermal efficiency of the Brayton cycle is the ratio of the net work produced by the cycle and the heat added to the cycle according to,

$$\eta_{th,Brayton} = \frac{W_{net}}{q_{in}} = \frac{C_p(T_c - T_d) - C_p(T_b - T_a)}{C_p(T_c - T_b)} = 1 - \frac{T_d - T_a}{T_c - T_b} = 1 - \frac{q_{out}}{q_{in}} \quad (29)$$

In the limiting case of an overall pressure ratio chosen such that T_b approaches T_c, the thermal efficiency will approach the Carnot efficiency of a heat engine operating between the hot temperature T_c, and the cold temperature T_a,

$$\eta_{th,Carnot} = 1 - \frac{T_a}{T_c} \quad (30)$$

and the net work of the cycle will approach zero. At the "other end" of the cycle, when the pressure ratio approaches one, the net work will once again approach zero and so will the efficiency. This indicates that for obtaining maximum work from a Brayton cycle, the pressure ratio must be carefully chosen. In terms of efficiency, the pressure ratio should be as high as possible approaching the pressure ratio needed to establish a compressor exhaust temperature close to T_c Even from this simple reasoning it can be concluded that an optimized Brayton cycle requires well balanced cycle design parameters in order to achieve high specific work and efficiency at the same time. This is further clarified in Figure 10, where three different Brayton cycles are illustrated together with the resulting trend in efficiency and specific work.

Also shown in Figure 10 is the important result that real Brayton cycles with component losses also have an optimum pressure ratio with respect to efficiency.

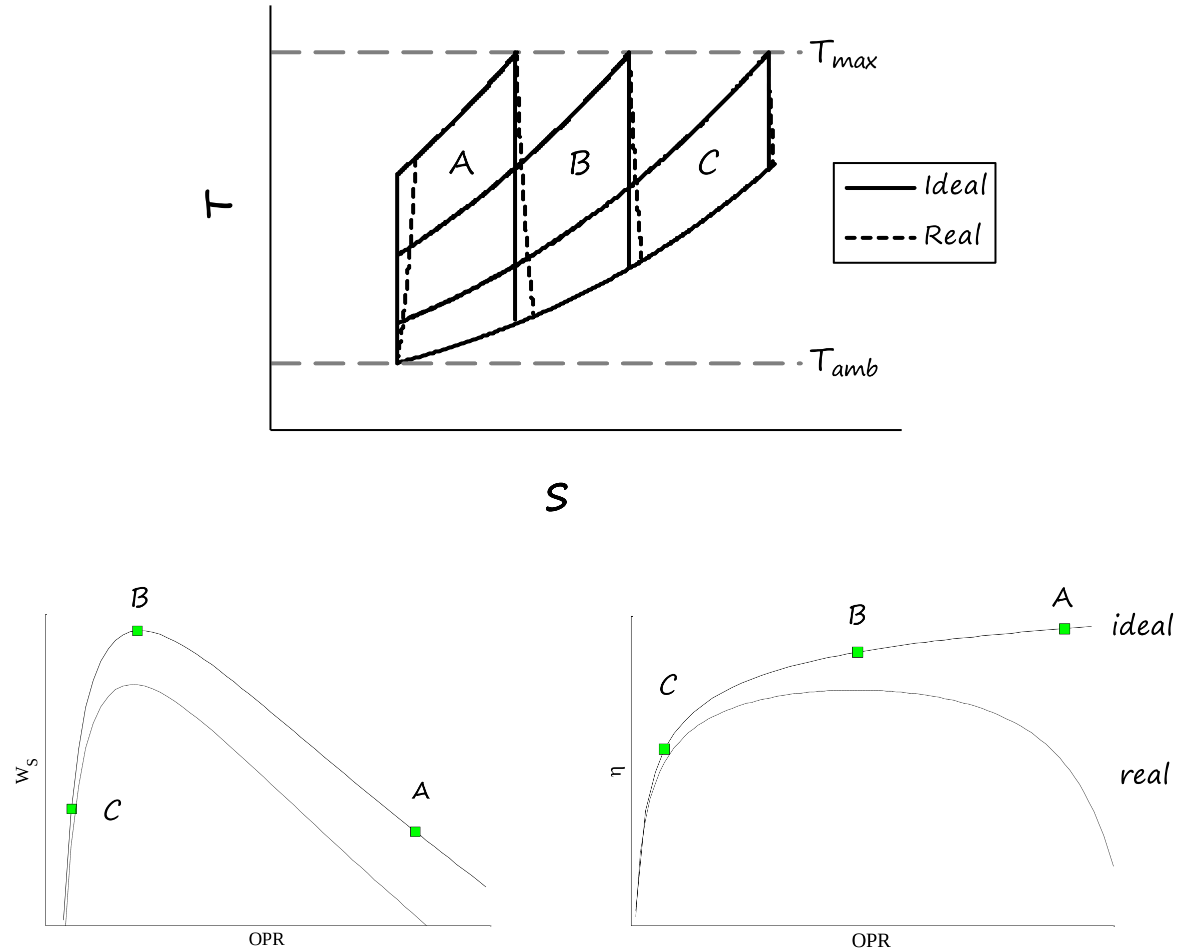


Figure 10. Specific work and thermal efficiency of the Ideal and real Brayton cycle as a function of overall pressure ratio.

6.3 Emissions of Nitrogen Oxide

Ideal combustion of a hydrocarbon fuel using air as the oxidizer results in carbon dioxide and water in the exhaust. In real aero engine combustion chambers however, there are a number of additional emissions formed during the combustion process. Even though the nitrogen is normally considered inert, small amounts of oxides of nitrogen are formed at the high temperatures in the primary combustion zone. The formation of nitrogen oxides, i.e. NO and NO_2, is due to several complex mechanisms, but the fundamental requirement is that excess oxygen is available and that nitrogen molecules start to dissociate and oxidize. High temperatures and excess oxygen is the foundation for the most important mechanism, thermal $NO_{x,}$ formation as described by for instance Zeldovich (Lefebvre, 1999),

$$\begin{aligned} &O_2 \leftrightharpoons 2O \\ &N_2 + O \leftrightharpoons NO + N \\ &N + O_2 \leftrightharpoons NO + O \\ &N + OH \rightleftharpoons NO + H \end{aligned} \qquad (31)$$

Oxides of nitrogen are also formed by the nitrous oxide mechanism[14] and are initiated by the formation of nitrous oxide,

$$N_2 + O \leftrightharpoons N_2O \tag{32}$$

The nitrous oxide is then oxidized to NO by the reaction,

$$N_2O + O \leftrightharpoons NO + NO \tag{33}$$

and also by the reactions;

$$N_2O + H \leftrightharpoons NO + NH \tag{34}$$
$$N_2O + CO \leftrightharpoons NO + NCO \tag{35}$$

Another source of NO_x formation is the oxidation of nitrogen in the fuel. Light hydrocarbon fuel fractions can contain small amounts of nitrogen (< 0.06 per cent by weight), and heavier fuel fractions can contain up to 1.8 per cent by weight. In addition to thermal NO_x, prompt NO_x may be formed. The process of formation is not entirely understood (Sjöblom), but it is observed that under certain conditions, NO_x is formed early in the flame region. Nicol et al (Nicol, 1992) suggests the initiating reaction,

$$N_2 + CH \leftrightharpoons HCN + N \tag{36}$$

Under lean-premixed conditions, the HCN oxidizes to NO mainly by a sequence of reactions involving HCN → CN → NCO → NO, and the nitrogen atom oxidizes to NO mainly by the second Zeldovich reaction given above. Furthermore, Nicol et al, showed that for equivalence ratios of 0.8 burning methane fuel in a lean-premixed combustor, the thermal NO_x contribution was about 60 percent of the total.

In Figure 11, the normalized NO and NO_2 content by mass is shown to peak at equivalence ratios close to 0.8 which corresponds to oxygen excess levels in the range of 4-5 % by mass. The results are computed using chemical equilibrium models (Gordon and Mcbride, 1994, McBride and Gordon, 1996) and represents an infinite residence time. In a practical application, the influence of the chemical kinetics has lead to a class of combustor designs that cool the primary zone as quickly as possible. However, from the diagram in Figure 11 it is interesting to note that in the region of maximum combustion temperatures, at the slightly fuel rich side of the stoichiometric conditions, the NO_x levels decrease substantially due to very low levels of excess oxygen.

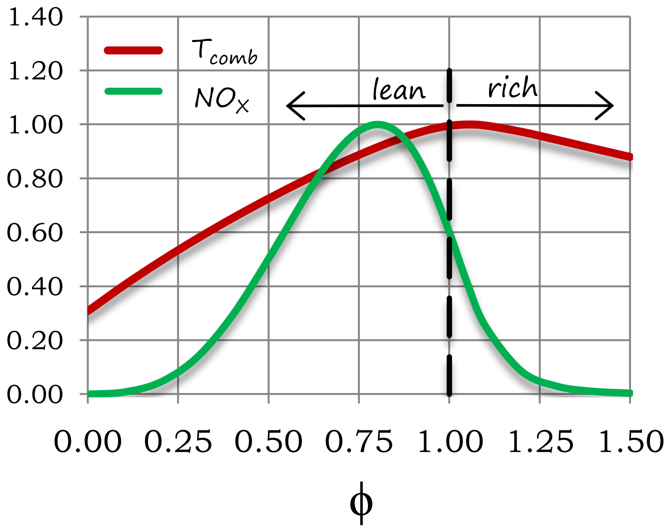


Figure 11. Normalized emissions of nitrogen oxide by mass for a combustion air temperature of 800 K and a Jet-A temperature of 288.15 K as a function of equivalence ratio.

In order to capture the effect of the engine design parameters on NO_x a semi-analytical expression as given by Lefebvre (Rizk and Mongia, 1993) may be used,

$$EINO_x\left(\frac{g}{kg}\right) = 0.459E - 8{P_3}^{0.25}F\tau e^{0.01T_{st}} \qquad (37)$$

where P_3 is the compressor discharge pressure [kPa], F is the fraction of air to the primary zone, τ is the residence time [ms] and T_{st} is the stoichiometric flame temperature [K].

6.4 Aerodynamic Efficiency

As part of this thesis, conceptual methods have been developed to find the impact of flight Mach number, altitude as well as the aircraft wing design parameters on overall aircraft performance of modern and next generation aircraft. Special attention has been devoted to modeling the compressibility drag, due to its strong impact on the optimum cruise Mach number for jet driven transport aircraft.

The lift-to-drag ratio, L/D or in non-dimensional terms C_L/C_D, is an efficiency measure that reveals how efficiently the lift force needed to balance the aircraft weight, W, is generated Therefore it is also referred to as an aerodynamic efficiency. For a given aircraft weight in steady level flight the lift surface must create lift so that $L \approx W$, and the non-dimensional drag can be computed according to,

$$C_D = C_{D0} + C_{Di} + C_{Dc} \qquad (38)$$

where C_{D0} is the lift independent drag coefficient, C_{Di} the induced drag coefficient and C_{Dc} is the compressibility drag coefficient. This is easily converted back to dimensional terms as $D = C_D \times q \times S$, and in the same manner lift is computed according to $L = C_L \times q \times S$. For equation (38) to be applied it is necessary to understand the different terms incorporated in the equation.

6.4.1 Lift-independent Drag

The lift independent drag is assumed to be independent of the creation of lift. It consists of friction- and pressure separation drag. The friction drag is computed from the friction coefficient of a flat plate, c_f, and the wet surface, S_w, and is then corrected for the real-shape using a form factor so that,

$$\sum CD_0 = c_f \times S_w \times ff \tag{39}$$

where ff is the form factor.

The art of determining the aircraft lift-independent drag is thus reduced to computing the wet surface of the different airplane components exposed to the air flow and determining the proper form factors for those components. If the aircraft technology under study is not subject to any laminar flow technology implementation this term has a weak dependence on aerodynamic technology maturity.

6.4.2 Lift-Induced Drag

Lift-induced drag stems from the vortices resulting from the spanwise flow reaching the tips of the finite wing.

A more intuitive but also approximate explanation to the creation of induced drag is given by simple momentum theory but does not explain why the end vortices are formed. Consider a streamtube of air with the free stream velocity V_∞ and diameter equal to the wing span b. From the continuity equation the mass flow of the stream tube can be expressed as,

$$\dot{m} = \rho V_\infty A = \rho V_\infty \frac{b^2}{4}\pi \tag{40}$$

To create lift there must be a deflection of the stream tube by a small angle ε opposite to the direction of lift, so that a downward component of the free stream velocity is created according to,

$$w = V_\infty sin\varepsilon \approx V_\infty \varepsilon \tag{41}$$

So that lift can be written as,

$$L = \dot{m}w = \rho V_\infty^2 \frac{b^2}{4}\pi\varepsilon \tag{42}$$

The total force acting on the wing is a sum of the lift force normal to the incoming flow, and a induced drag component, D_i, acting in the direction of the flow. Under the assumption of small angles, i.e. ε close to zero, the induced drag component can be explained as the difference of the free stream velocity and its horizontal component, ΔV_i according to,

$$\Delta V_i = V_\infty - V_\infty cos(\varepsilon)$$

Which by Taylor expansion can be rewritten as,

$$\Delta V_i = V_\infty \left(1 - \left(1 - \frac{\varepsilon^2}{2}\right)\right) = V_\infty \frac{\varepsilon^2}{2} \tag{43}$$

Applying equation (41) with (43) gives,

$$\Delta V_i = \frac{w\varepsilon}{2} \tag{44}$$

Thus, the induced drag can be written as,

$$D_i = \frac{\dot{m}w\varepsilon}{\mathbf{2}} = \frac{L\varepsilon}{2} \Leftrightarrow C_{Di} = \frac{C_L\varepsilon}{2} \tag{45}$$

Applying equation (42) into (45) results in,

$$C_{Di} = \frac{C_L^2 q_\infty S}{2\rho\pi \frac{b^2}{4} V_\infty^2} = \frac{C_L^2}{\pi AR} \tag{46}$$

in which *AR* is the aspect ratio of the wing defined as b^2/S and is a measure of the slenderness of the wing, i.e. a high aspect ratio indicates a long narrow wing whilst a low aspect ratio indicates a short stubby wing.

The preceding derivation of lift induced drag using simple momentum theory turns out to be correct for the ideal case representing an elliptical wing with constant downwash velocity across the span *b*. For real wings, the Oswald span efficiency, *e*, is invoked into equation (46) which will be less than one for the real planar wing case, revealing how much the wing deviates from ideal. A simple expression for lift induced drag will therefore be,

$$C_{Di} = \frac{C_L^2}{\pi ARe} \tag{47}$$

6.4.3 Compressibility Drag

Compressibility drag is important to include early on in the conceptual design process since the cruise Mach number chosen for long range cruise will define the energy efficiency design point (EDP) for the engines as well as the aircraft. Consider the accelerating flow over the upper surface of an aircraft wing. At some free stream Mach number the flow will accelerate over the wing to an extent that it eventually reaches the sonic velocity at some point in the flow field. This flight Mach number is defined as the critical Mach number, M_{cr}. Any further increase in flight Mach number will create local supersonic flow velocities with a terminating normal shock wave and rapid drag rise as a consequence. In contrast to the exact and physical definition of the flight critical Mach number, the drag-divergent Mach number, M_{DD}, is not as easily defined, some examples are given here,

1. The Boeing definition states; *M_{DD} is the free stream Mach number for which the drag due to compressibility first reaches 20 counts above the incompressible level, i.e. $\Delta C_D = 0.0020$*
2. *The Douglas definition states; M_{DD} is the free-stream Mach number for which the slope of the drag rise first reaches the value 0.10, i.e.*

$$\frac{\partial C_D}{\partial M} = 0.10 \tag{48}$$

A third, and more practical definition is based on Boeings [cost index explained] definition of LRC;

3. The LRC cruise speed is the speed above the maximum range speed that results in a 1% decrease in fuel mileage.

Due to the high cruise speeds utilized by jet driven transport aircraft, transonic airfoil- and wing theory must be applied. The basic equation used here was defined by Korn in the 1970s and provides a simple means of estimating airfoil performance,

$$M_{DD} + \frac{C_l}{10} + \frac{t}{c} = \kappa \qquad (49)$$

where κ is a technology factor representing the airfoil design technology level, for example κ = 0.95 represents NASA supercritical airfoils, and κ = 0.87 conventional airfoil technology such as the NACA 6-series airfoils (Mason, 1990). From the important work on wing planar forms by Jones and others during the second world war, we know that the compressibility effect is dependent on the Mach number normal to the leading edge of the wing, so that equation (49) can be re-written for swept-wings according to,

$$M_{DD} = \frac{\kappa}{cos\Lambda} - \frac{t/c}{cos^2\Lambda} - \frac{C_l}{10cos^3\Lambda} \qquad (50)$$

By assuming an empirically derived shape of the drag rise, for instance;

$$\Delta C_{Dc} = 20(M - M_{cr})^4 \qquad (51)$$

In combination with the Douglas criteria for drag-divergence,

$$\frac{\partial C_D}{\partial M} = 80(M - M_{cr})^3 \qquad (52)$$

and according to the definition above M equals M_{DD} when this condition is fulfilled. Solving for M_{cr} yields,

$$M_{cr} = M_{DD} - \left(\frac{0.1}{80}\right)^{1/3} \qquad (53)$$

For Mach numbers above the critical value, the drag rise can now be modeled as,

$$\Delta C_{Dc} = 20(M - M_{cr})^4 \qquad (54)$$

A method for conceptual studies of the effects of wing sweep, thickness and airfoil technology with respect to compressibility drag rise is now available. Of course, if airfoil/wing data is available to the user a more accurate semi-empirical expression can be developed, however for the purpose of predicting LRC for existing aircraft the algorithm defined by equation (49) through (54) seem to be sufficiently accurate. As an example, Mason et al used a value for κ of 0.89 when analyzing the compressibility drag characteristics of a B747-200, and a value of 0.955 when analyzing the newer B777 aircraft.

In Figure 12 some results for the medium range generic (MRG) aircraft are shown in order to illustrate the results of conceptual methods for computing aircraft drag. The baseline aircraft is in this example assumed to have an average maximum thickness-to-chord ratio of 10.5%, shown by the solid lines in the figure. The results for a two percent thicker wing, i.e. 12.5%, are also shown in the figure. Note that the compressibility drag rise dependence on lift. The lines connecting the drag curves, $M(L/D)_{max}$ and *LRC*, illustrates the points of minimum fuel consumption and the typical long range cruise points.

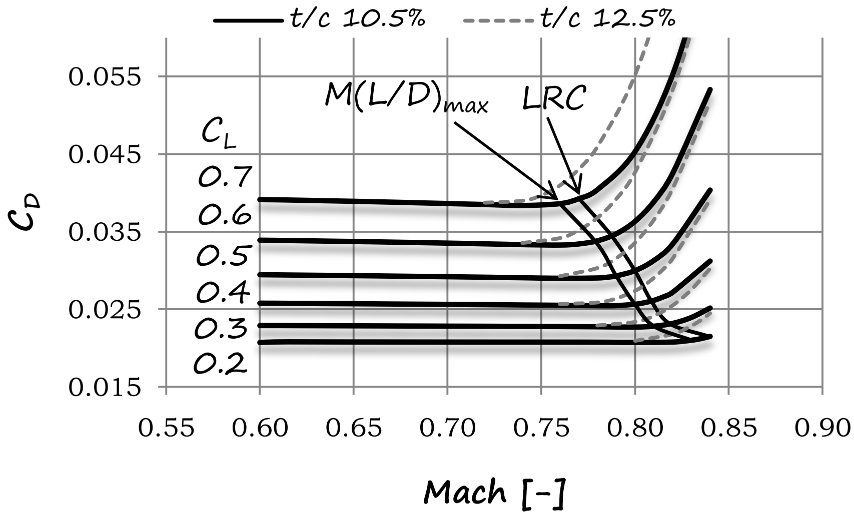


Figure 12. MRG compressibility drag as a function of Mach number for lift coefficients between 0.2 and 0.7.

6.5 Aircraft Weight

Models for estimating aircraft and engine weight can be divided into three main groups; statistical, quasi-analytical, and analytical weight estimation methods. The proper method to use for a particular design study depends on which phase in the design process the weight estimation is to occur, and what data that is available for verification and validation. As an example, consider a design study for a new aircraft. Early in the conceptual design process, when the need exists to get a first order estimation and a proper scaling of the airplane weight, one might consider statistical expressions.

At the highest system level, aircraft gross weight is broken down into overall empty weight, W_{oe}, the weight of the payload, W_{pay}, and fuel weight, W_{fuel}, so that the gross weight or take-off weight is written as,

$$W_{T/O} = W_{OE} + W_{pay} + W_{fuel} \quad (55)$$

Continuing the weight breakdown according to equation (55), the overall empty weight consists of the airplane structure and all the items necessary for operating it. The structure weight, which consists of the wing, fuselage, tail, surface controls, nacelle group and the landing gear according to,

$$W_S = W_{wing} + W_{fus} + W_{tail} + W_{surf} + W_{nac} + W_{ldg} \quad (56)$$

6.5.1 Validation of Aircraft Weight Methods

For the validation of the weight prediction methods, the data given by Torenbeek has been used (Torenbeek, 1982). The semi-analytical expressions show good agreement with the data presented by Torenbeek, however the airplane types are quite old so that additional calibrations should be performed. The airplane modeled for the purposes of this method verification was a medium range generic aircraft, MRG, i.e. a narrow body airliner in the size of the widely used Boeing 737-800 airplane. The results for the structure sub-groups are shown in Figure 13 through Figure 18. The MRG airplane total take-off weight was calculated to be 79395 kg, which is within 1% (0.5%) of the manufacturer’s data reported (Boeing, 2011). Although this particular point was close to the manufacturer’s data, this high accuracy is not to be expected for all predictions. The calibration data used here is taken from rather old aircraft and should be re-calibrated on modern designs if such data is available.

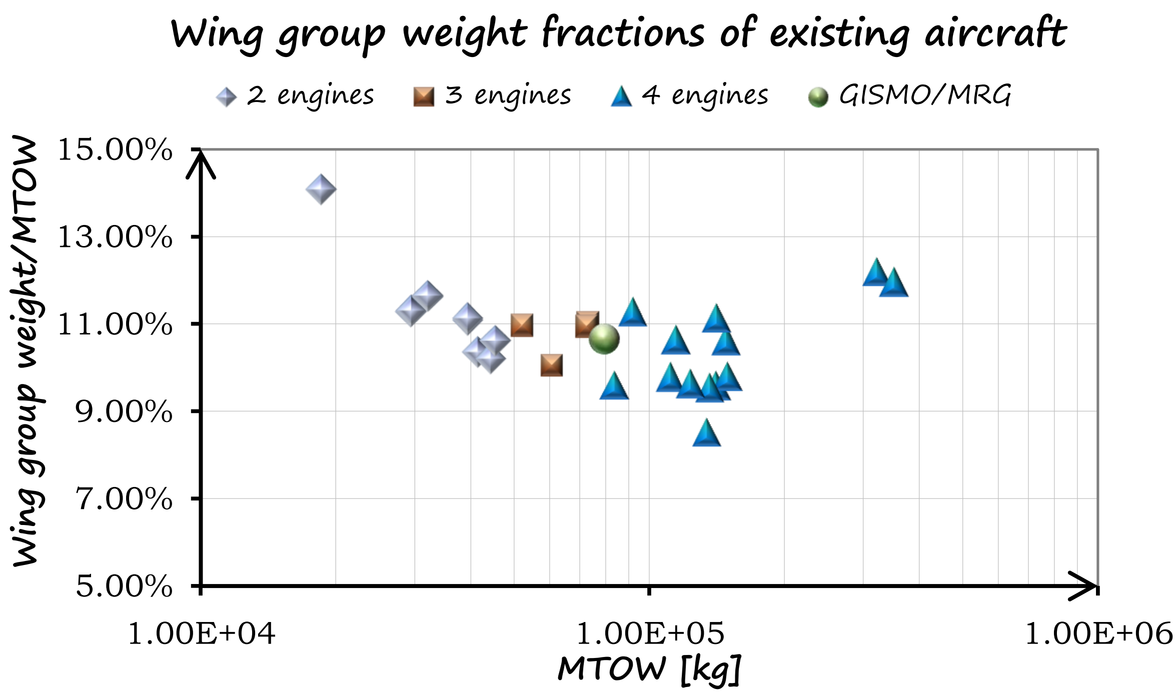


Figure 13. Comparison of the predicted MRG wing weight to existing 2-, 3- and 4-engined jet aircraft.

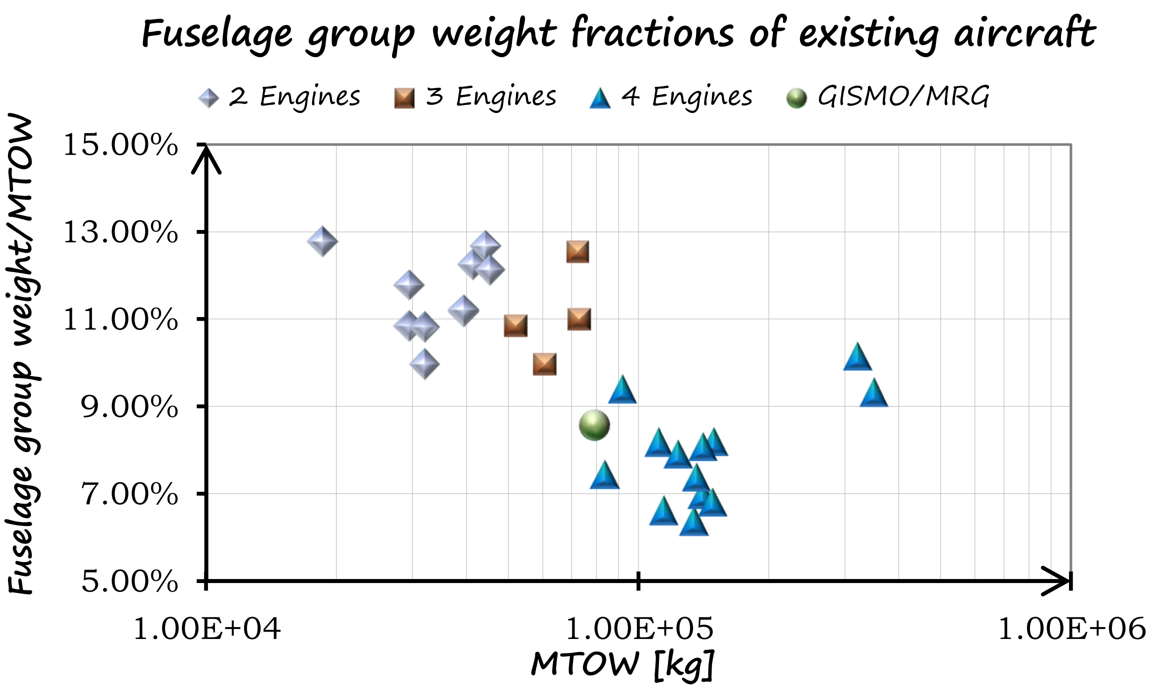


Figure 14. Comparison of the predicted MRG fuselage weight to existing 2-, 3- and 4-engined jet aircraft.

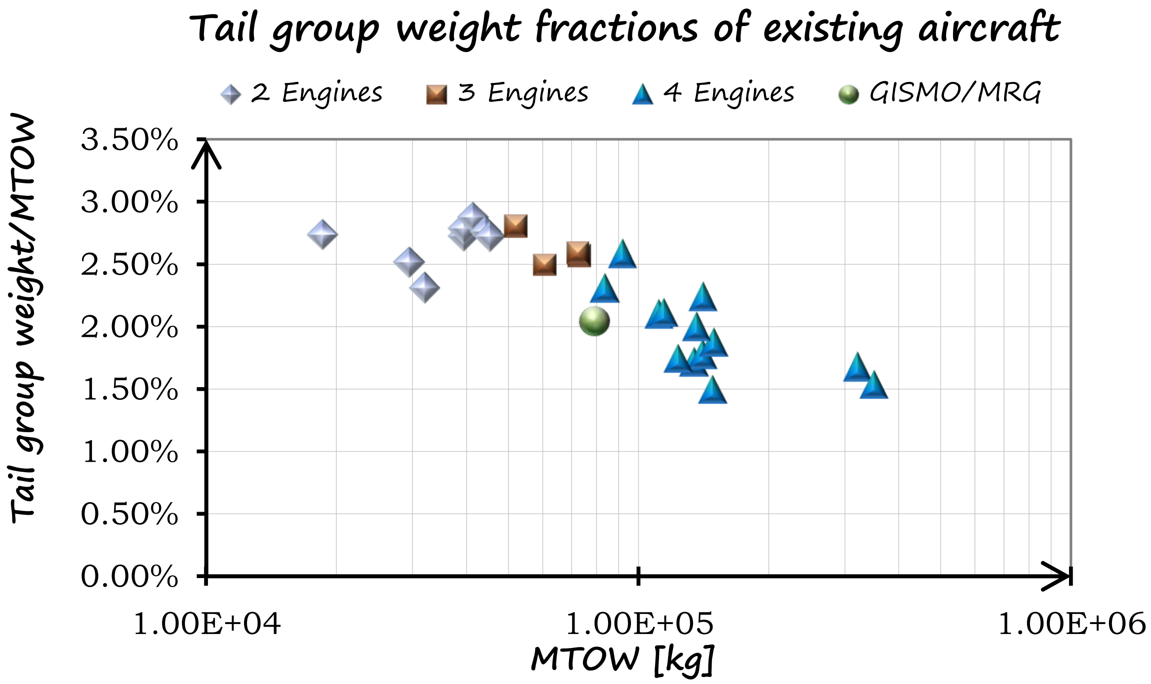


Figure 15. Comparison of the predicted MRG tail group weight to existing 2-, 3- and 4-engined jet aircraft.

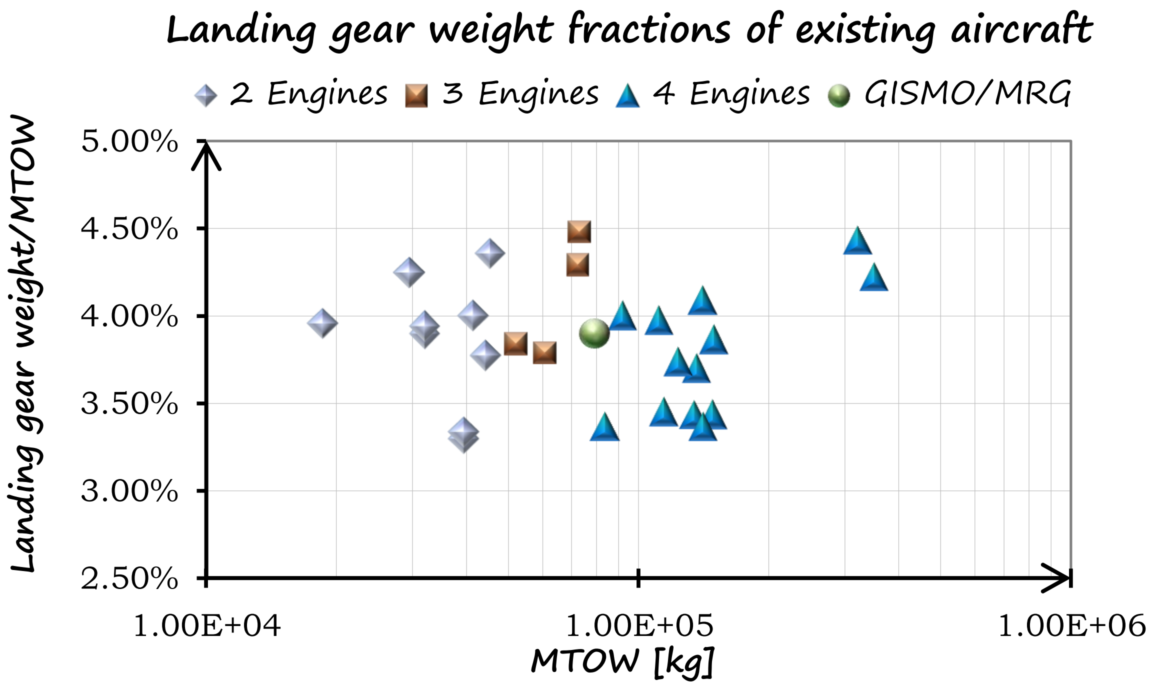


Figure 16. Comparison of the predicted MRG landing weight to existing 2-, 3- and 4-engined jet aircraft.

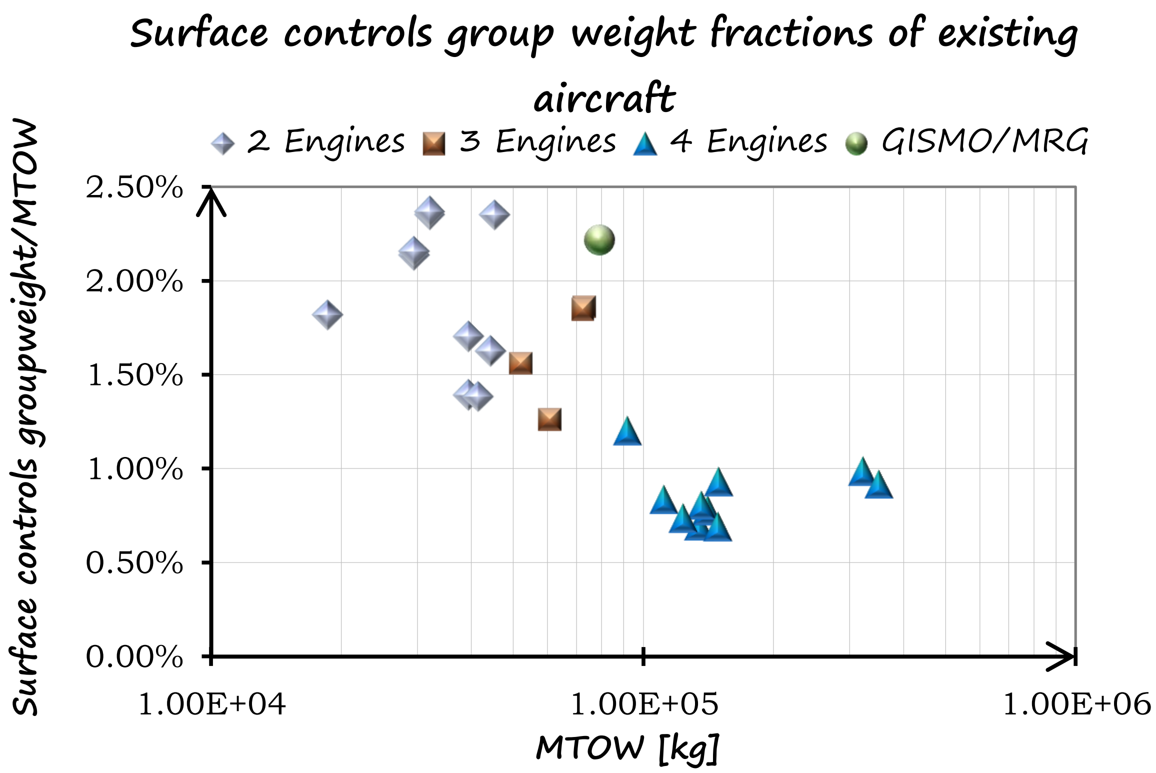


Figure 17. Comparison of the predicted MRG surface controls weight to existing 2-, 3- and 4-engined jet aircraft.

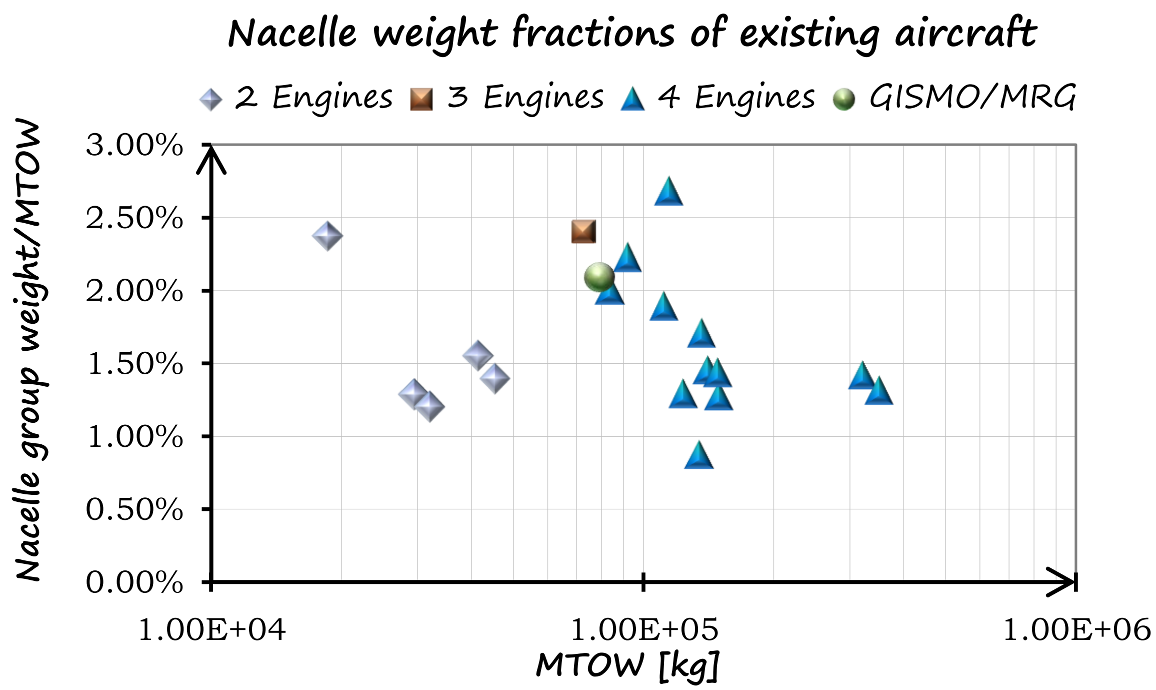


Figure 18. Comparison of the predicted MRG nacelle weight to existing 2-, 3- and 4-engined jet aircraft.

To get a more up-to-date validation of the GISMO weight code, a comparison was made with the

FLOPS code developed Northrop Grumman [Referens till NASA N+3]

Weight Item	FLOPS	GISMO	Relative diff.
Wing group [kg]	8795.6	8455.0	-3.9%
Tail group [kg]	1313.1	1617.5	23.2%
Fuselage group [kg]	8225.9	6792.6	-17.4%
Landing gear [kg]	3340.3	3093.0	-7.4%
Surface controls [kg]	N/A	1758.9	N/A
Nacelle group [kg]	1795.3	1657.0	-7.7%
Structural weight [kg]	23470.2	23374.0	-0.4%
Propulsion group [kg]	6014.2	5986.9	-0.5%
Systems & Equip. [kg]	10927.0	10939.9	0.12%
OEW [kg]	41667.0	42868.5	2.9%
Fuel [kg]	20827.6	20488.3	-1.6%
Payload [kg]	16726.2	16038.0	-4.1%
TOGW [kg]	**79220.7**	**79394.8**	**0.22%**

Table 2. Validation of GISMO weight for a B737-800 aircraft. Comparison with the Northrop Grumman FLOPS code.

It is noted that that both methods produce results that are within 0.5% from the MTOW reported by Boeing for the aircraft type studied here.

6.6 Performance - Component Maps

For the purposes of this thesis, no new methods regarding performance analysis, off-design behavior, has been necessary to develop. However, due to its importance for aero engine performance the existing methodology that has been used is presented briefly herein.

The group's performance methods, in particular the methods for generating component maps, are based on the work by Robbins and Dugan (Robbins and James F. Dugan, 1965). They presented a method for obtaining the performance map of a new multistage compressor based on empirical data from previous compressor designs. A brief explanation follows.

The line connecting all the points of maximum efficiency, for each speed, is called the backbone, *bb,* of the compressor map, or simply the compressor backbone. Any value along this line is called a backbone value. Furthermore, the point of maximum polytropic efficiency for the backbone line and therefore the entire compressor map, is called the reference point, *rp,* and is not necessarily the design point.
The method consists of three major steps that are; calculation of the compressor backbone, calculation of the stall-limit, or surge line, and finally calculation of the off-backbone behavior along the constant speed lines.

- Calculation of the compressor backbone and surge line:

Given the pressure ratio at the reference point, π_{rp}, and the rotational speed, n, the backbone mass flow, $\dot{m}_{bb}$, isentropic efficiency, η_{bb}, and pressure ratio π_{bb} for the compressor backbone are acquired. The tables f_1, f_2, f_3 and f_4 represent data derived from a number of historical compressor designs and are given as:

$$\frac{\pi_{bb}}{\pi_{rp}} = f_1(\nu, \pi_{rp}) \quad (57)$$

$$\frac{\dot{m}_{bb}}{\dot{m}_{rp}} = f_2(\nu, \pi_{rp}) \quad (58)$$

$$\frac{\eta_{bb}}{\eta_{rp}} = f_3(\nu, \pi_{rp}) \quad (59)$$

$$\frac{\pi_{surge}}{\pi_{rp}} = f_4(\dot{m}, \pi_{rp}) \quad (60)$$

where $= \frac{n}{n_{rp}}$.

- Constant speed characteristics (off-backbone behavior):
  Once the compressor backbone is determined, the variation of pressure ratio, mass flow and efficiency along constant speed lines can be determined as a function the flow parameter, ϕ, which is scaled with the backbone value, ϕ_{bb}, to obtain the relative flow parameter ϕ_{rel}, according to:

$$\frac{(T_2 - T_1)}{(T_2 - T_1)_{bb}} = f_5(\phi_{rel}) \quad (61)$$

$$\frac{\eta}{\eta_{bb}} = f_6(\phi_{rel}) \quad (62)$$

where $= \frac{\dot{m}\sqrt{\theta}}{\delta}\sqrt{\frac{\tau}{\pi}}$, $\phi_{rel} = \frac{\phi}{\phi_{bb}}$, $\delta = \frac{P}{101325.0}$, $\theta = \frac{T}{288.15}$, $\tau = \frac{T_2}{T_1}$ and $\pi = \frac{P_2}{P_1}$

An interesting empirical observation of the method described here, is that values from the entire compressor map tend to collapse into a single curve according to equation (61) and (62) respectively and in such way represents the off-design behavior of the compressor. And together with the tables containing the backbone data, the entire compressor map of the new compressor design is determined. This method was implemented by Grönstedt (Grönstedt, 2000) and during that process some shortcoming of the method were observed. The original correlation only contained compressor data up to 110% of the reference rotational speed. The method used to determine off-backbone behavior agreed poorly with empirical data. Also noted was that the original compressor data were based on compressor designs from the period 1950-1965. Furthermore compressor choking effects were not included in the original model. McKenzie (McKenzie, 1997) also noted that the strategy of using a compressor backbone line for prediction of compressor performance probably is a useful strategy but that the original data did not represent new designs successfully. So the original method was modified with internal Volvo Aero compressor data to get better agreement with modern designs.

7 Means of Improving Overall Engine Efficiency

7.1 Aero engine technology trends

Forecasting methodologies is an entire field of research itself and commonly applied by economists, business marketing divisions and demographic researchers, to mention a few. These methodologies also have a very useful application for any kind technology development is sometimes necessary apply in order to predict future state-of-the-art technology and market outlooks. In particular, the forecasting technique quantifies the rate at which technology improvement has to occur at in order to reach future research and technology goals.

Consider the task of forecasting, i.e. estimating the performance, y, of a particular technology over time, t. Assume that the change, hopefully an improvement, in performance is dependent and proportional to the present time so that,

$$\frac{dy}{dt} = ky \tag{63}$$

where k is a constant. Equation (63) has the well known solution $y(t)=Ce^{kt}$, that would imply exponential performance improvement of the technology over time which is not a very realistic assumption. In real world cases most technologies evolve and improve at some rate until physical limits approaches, the market is becoming saturated or perhaps the costs associated with any further improvements are just too high to motivate further development. The technology improvement eventually declines, showing asymptotic behavior approaching the upper limit of the particular technology’s performance. Based on this reasoning, equation (63) can be written as,

$$\frac{dy}{dt} = by(L - y) \tag{64}$$

In which the asymptote L is introduced. The performance of the technology over time is now assumed to be proportional to the current performance level, y, and the remaining gap to the asymptote, L-y. Equation (64) can be shown to have a solution according to,

$$y(t) = \frac{L}{1 + ae^{-bt}} \tag{65}$$

where a is a constant of integration. This equation is usually referred to as the logistics equation, or Pearl-Verhulst equation after its founders, and is widely used by for instance demographers, biologists (Kingsland, 1982), economists and also for business forecasting in general. For the purpose of estimating technology trends within this section, this equation is applied.

7.2 Component efficiencies

The progress of more energy efficient aero engines relies much on the component efficiency development. The evolvement of 3D flow computational capability within the industry has contributed to efficiency improvements in the last decades, will yet continue to contribute to the future improvement. Presented in figures 18 throughFigure 23 are polytropic efficiency trends based on data from (Grieb, 2004) and (Grönstedt, 2011).

The component efficiency trends shown here are estimated by assuming restricted technology growth according to Pearl amongst others. Equation (65) described in section 7.1 is assumed to be applicable. The coefficients *a* and *b* are given by regression analysis using the efficiency data together with an assumed asymptotic component efficiency of 95% for compressors and 97% for uncooled turbine efficiency. It can be argued to be conservative but is based on the author's opinion and a recent study on turbomachinery efficiency limits (Hall, 2010).

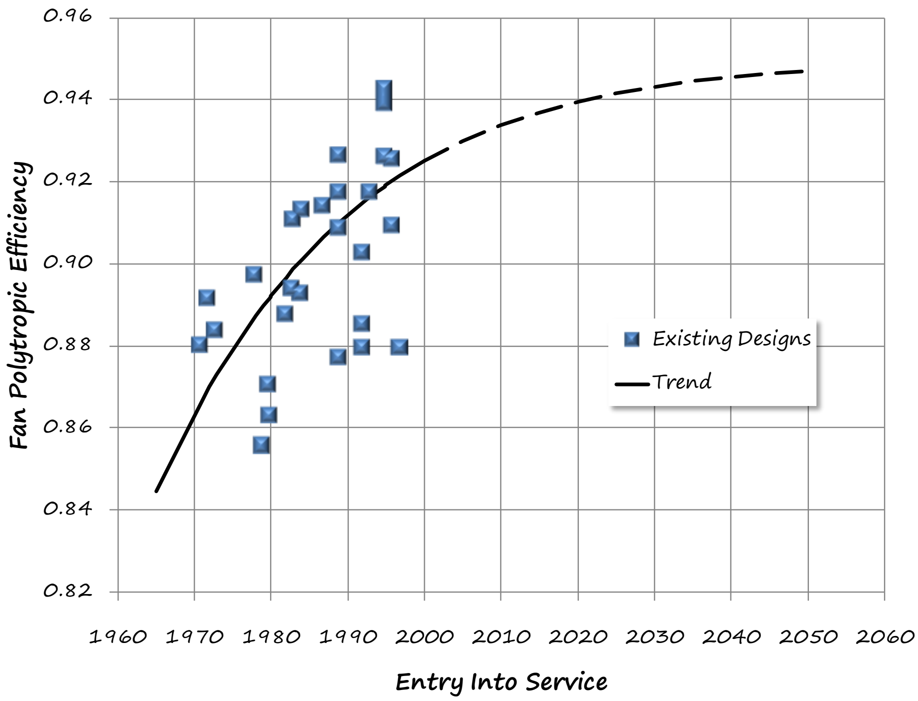


Figure 19. Data and trend of Fan polytropic efficiency.

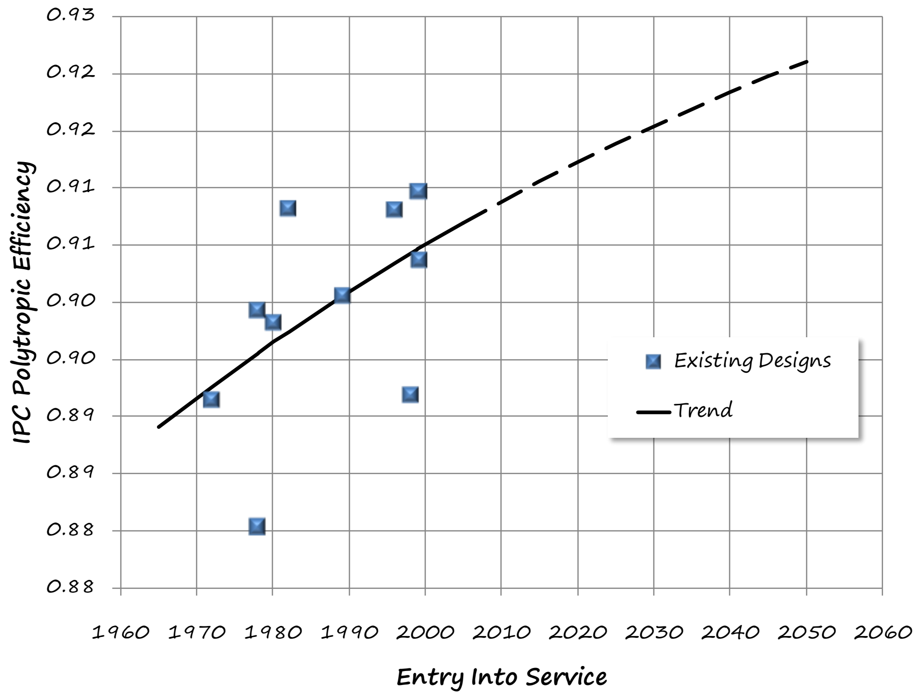


Figure 20. Data and trend of IPC polytropic efficiency.

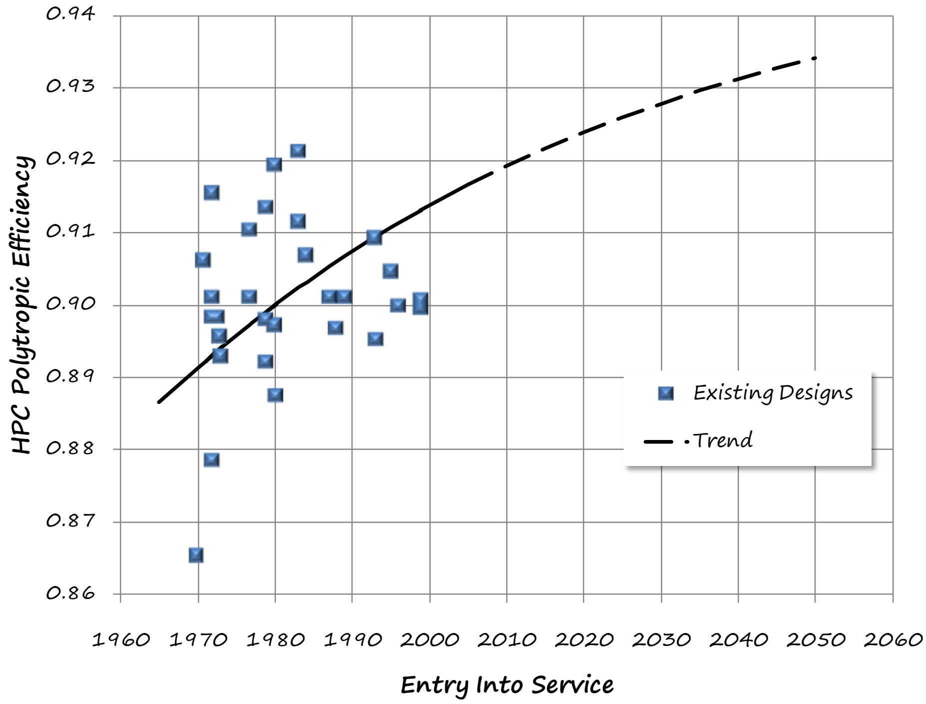


Figure 21. Data and trend of HPC polytropic efficiency.

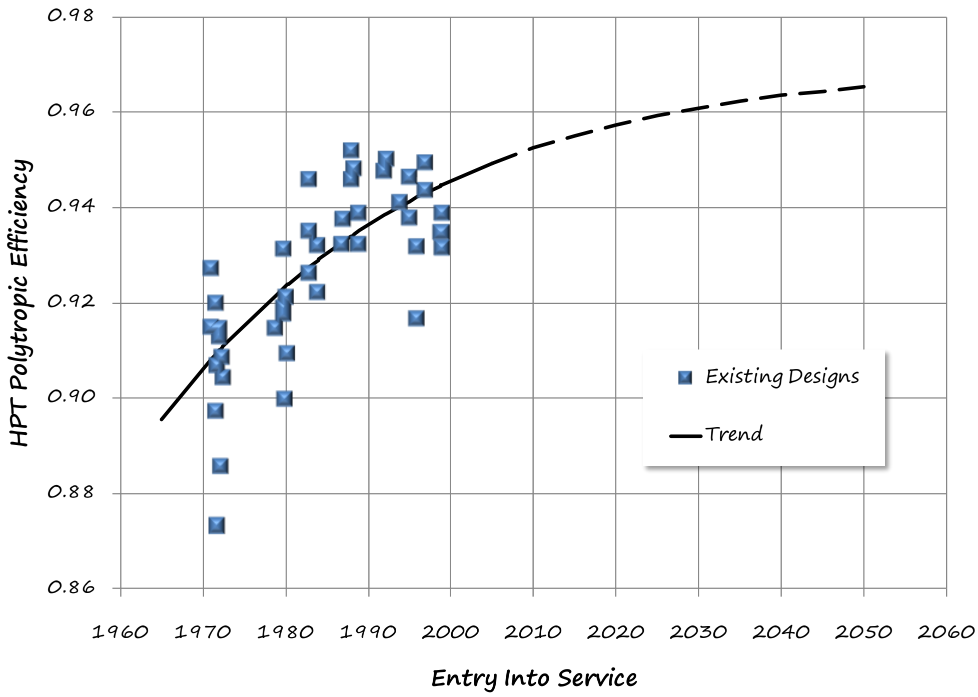


Figure 22. Data and trend of HPT polytropic efficiency.

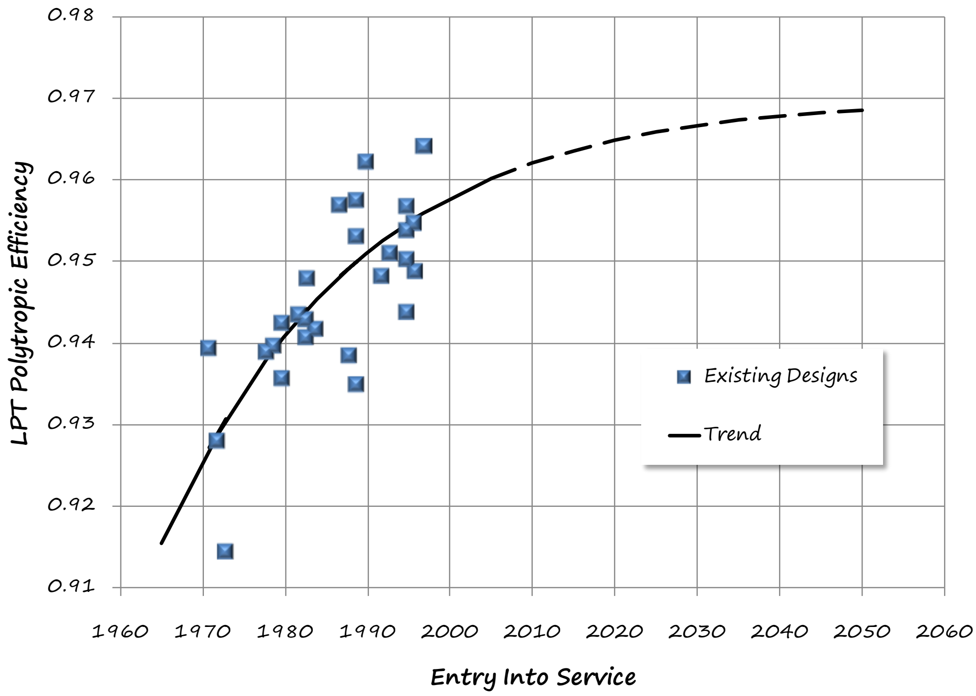


Figure 23. Data and trend of LPT polytropic efficiency.

7.3 Propulsor Technology

One key to increased aero engine efficiency is to increase the propulsive efficiency, as explained earlier in section 6.1. One way of increasing the propulsive efficiency of turbofans is to lower the fan jet velocity which is given by the fan pressure ratio, and the use of a lower fan pressure ratios require larger fans for a given thrust requirement.

Ignoring any possible maximum fan size diameter constraints imposed by the aircraft installation, the increased fan diameter leads to designs utilizing lower rotational speeds in order to lower the fan blade tip speed to avoid high levels of compressibility losses and noise. To match the larger and slower fans, in terms of torque, the LPT must be designed with increasing number of stages and airfoils, and most likely an increase of the diameter with increasing cost and weight as a possible consequence. On the other hand, if a fan gear box is introduced to de-couple the fan from the low-pressure spool, then the design speed of the fan, LPT and LPC could be carefully chosen without the trade-off in fan and LPT performance described here. The resulting engine configuration is called a geared-turbofan engine (GTF) and was first introduced in a large scale in the early 1970s by Garrett AiResearch when they developed the two-spool TFE731 geared turbofan for the business jet segment (Steele and Roberts, 1972). The engine is still manufactured today by Honeywell Aerospace with over 11000 units built (Honeywell, 2011).

Modern variants of the GTF engine intended for the next generation medium-range jets are getting closer to entry into service as the Pratt & Whitney PW1000G engines are developed and flight tested at the moment (Pratt&Whitney, 2011). In terms of performance, it is not explicitly clear whether the GTF would outperform its conventional counter-part assuming equally advanced technologies, e.g. the new Leap-X by CFM International (CFM, 2011). One common assumption is that the high-speed LPT of the GTF will possibly achieve higher efficiency due to the aerodynamically lightly loaded stages which can be understood by looking at historical Smith charts of turbine efficiency versus flow and aerodynamic stage loading parameters (Smith, 1965). However the combination of increased blade speed, stage pressure ratios in excess of 2.2 and Mach numbers possibly exceeding 1.2 (Malzacher et al., 2006), could on the other hand cancel this potential benefit, so yet there are other trade-offs introduced for the LPT design; for instance the trade between the high flow Mach number and the aerodynamic stage loading and also the mechanical loading versus the blade speed. In terms of mechanical loading, an important constraint is the AN^2 loading parameter (A is the annulus area and N is the rotational speed) which is an indicator of the blade root stress. For slow conventional LPTs of direct driven turbofans, the mechanical loading of the blade roots has not been a limiting factor as for the highly stressed HPTs. Recent results from design and testing of a LPT for GTF application indicate an increase in AN^2 by more than a factor of two (Malzacher et al., 2006) compared to conventional designs.

The next geared turbofan is expected to have design fan-pressure ratios between 1.3 and 1.45 and bypass-ratios in the range of 10 to 12 (Riegler and Bichlmaier)[5]. Even if these designs could be configured without a fan gearbox, at least for the pressure ratios close to 1.4, further improvements of the propulsive efficiency by going to ultra-high bypass-ratios in excess of 13 will most likely incorporate fan gearboxes and also the introduction of at least one variability of the fan, e.g. a variable fan nozzle and/or variable pitch fan blades. The reason for this is the fan off-design performance at lower fan-pressure ratios. At top-of-climb and cruise

[5] The article by Riegler and Bichlmaier, accessed in 2011, can be downloaded from the MTU website; www.mtu.de.

conditions, the fan nozzle is choked. At take-off however, the nozzle will unchoke causing a reduction in corrected fan-flow and the fan operating point will move towards the surge line. This behavior will be more pronounced with decreasing fan pressure ratios calling for a fan variable nozzle in order to increase the fan flow during take-off.

7.4 Engine Core Technology

From fundamental thermodynamics it is known that one route for improving the thermal efficiency of the engine, is the use of higher overall engine pressure ratios and turbine inlet temperatures. Another way of achieving this is by achieving higher component efficiencies.
Modern large aero engines have pressure ratios over 40 and turbine inlet temperatures exceeding 1800 K at take-off conditions. Trends of these thermodynamic design parameters are shown in Figure 24 and Figure 25 (Benzakein, 2010).

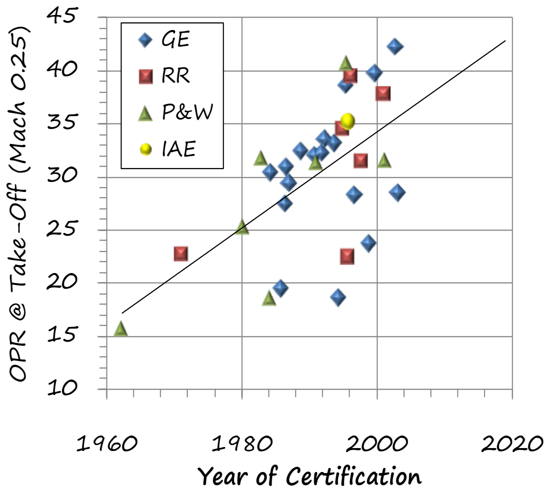


Figure 24. OPR as a function of engine certification year.

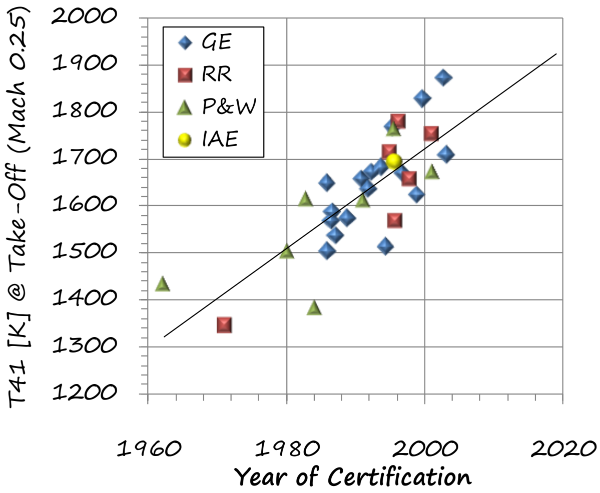


Figure 25. Turbine inlet temperature as a function of certification year.

The increase in turbine inlet temperature will increase the core specific power and result in smaller cores and thereby increasing the engine bypass ratio. On the other hand, the increase in turbine inlet temperature (TIT) will further increase the demand for turbine cooling for a given turbine material technology. Also worth noting is that the temperature of cooling air from the compressor discharge will also increase due to the higher OPRs. Turbine materials and cooling technology are therefore of crucial importance for the core engine technology development. Interesting to note is the fact that within NEWAC the highly innovative concept of applying cooled cooling air for the HPT was investigated. This concept would have the potential of allowing higher OPRs and TITs in future designs. Added to the turbine cooling requirement is the certification regulation for NO_x that are becoming more stringent and also impose constraints for the OPR and TIT design parameters.

Also studied within the NEWAC project are different technologies for controlling the core flow, in order to further increase the compressor stability, stall margin and efficiency (NEWAC, 2008).

7.5 Cycle Modifications

When the evolutionary way of improving aero engine energy efficiency rate of improvement run into practical, or physical boundaries suppressing any further development at decent pace, e.g. in terms of component efficiency improvement, there exist possibilities for applying modifications to the conventional engine cycle in order to further increase the aero engine efficiency.

From fundamental thermodynamics it is known that the net work output from a gas turbine is the difference of the work generated by the turbine(s) and the work consumed by the compressor(s). Furthermore it is known that the work is proportional to the specific volume of the media being compressed or expanded, which means that the required compression work will decrease if cooling is applied in the compression process and the expansion work will increase if the media is reheated during the expansion process. These cycle modifications are called intercooling and inter-turbine reheat respectively.

A third possible modification is recuperation, i.e. using a heat-exchanger to heat the compressor discharge air by the exhaust waste heat. This cycle modification can also be combined with intercooling and reheat.

The intercooling concept has the potential to further increase the OPR of the engine and to reduce the NO_x emission levels by decreasing the compressor discharge temperature. Within the NEWAC project the intercooling concept has been studied alone and together with a recuperator (the IRA concept,) (Boggia and Rüd, 2004). The potential benefits of the latter are restricted to lower OPRs due to the fact that the heat exchange is driven by the difference in the turbine exit temperature and the compressor discharge temperature.

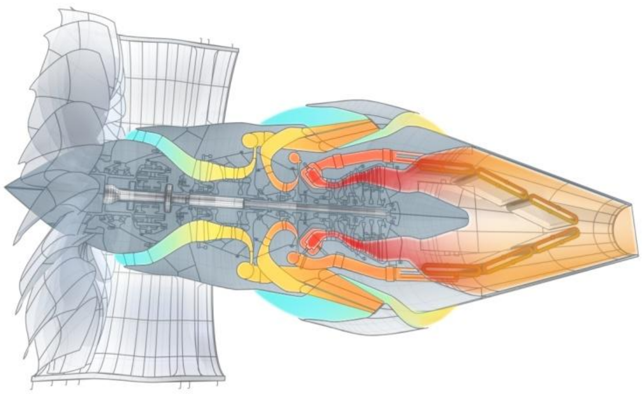


Figure 26. The IRA concept. Source: NEWAC.

7.6 Alternative cycles

Alternative cycles in this context refer to thermodynamic cycles different from the Brayton (Joule) cycle in terms of the thermodynamic process. The temperature-entropy diagram in Figure 27 shows the Brayton cycle, enclosed by the thermodynamic states a-b-c-d-a, and the Humphrey cycle described by the states a-b-c'-d'-a. The Brayton cycle undergoes isentropic compression between states a and b followed by isobaric heat addition between states b and c , isentropic expansion between c and d and finally isobaric heat-rejection to the surroundings between state d and a. The Humphrey cycle is essentially the same thermodynamic process except for one very important difference, the heat addition between states b and c' is isochoric, i.e. constant volume combustion, and this yields a quite large theoretical thermodynamic efficiency increase compared to the Brayton cycle. This is the reason for studying constant-volume combustion for continuous cycles.

The pulse-detonation engine (PDE) cycle has even greater theoretical potential for achieving higher thermal efficiency. It relies on an unsteady combustion, with repeatedly generated supersonic detonation waves inside a combustor generating high pressures and temperatures allowing for high theoretical thermal efficiencies. Although the unsteady nature of the of the PDE cycle it is possible derive close-form solutions for the PDE performance and to compare the ideal efficiency of the cycle to the steady Brayton and Humphrey cycles (Heiser and Pratt, 2002). An indicative example is given in Figure 27.

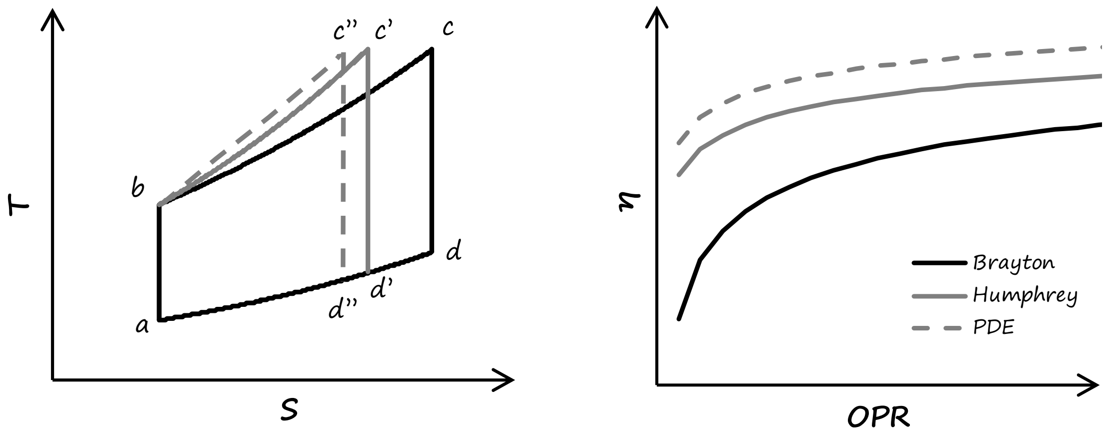


Figure 27. Schematic TS-diagrams and ideal thermal efficiency for the ideal Brayton, Humphrey and PDE cycles.

The possible application of pulse-detonation combustion into aerospace propulsion has been, and is studied widely. For the commercial aero engine industry, the idea of replacing the constant-pressure combustion by a pulse-detonation combustion device in a turbofan is studied at the moment, see for instance the NEWAC-project (NEWAC). To fully take advantage of the potential performance benefit of the PDE concept some intense research and innovative solutions must be produced. One of the practical problem is the expansion process in the intermittent cycle that will cause design issues regarding the turbines. It is claimed that practical turbine engines utilizing pulse-detonation combustion have the potential to reduce SFC by 5 to 10% over existing technology (Kaemming et al., 2006).

8 Means of Lowering NO_x Emissions

Shown in Figure 28 is NO_x certification data of existing engines (CAA, 2011) and ICAO NO_x regulatory levels for engine certification (ICAO, 1993). At present time, the CAEP/6 level is valid for any new engine entering production until the January 1, 2014 when the more stringent CAEP/8 regulatory levels will be effective. Also shown are the ACARE 2020 and 2050 NO_x goals assumed to be defined as an 80 and 90% reduction of the CAEP/2 levels respectively. Despite the higher pressure ratios of modern engines, the NO_x emissions are being still being reduced as a consequence of refined combustor designs. However, it is reasonable to assume that for the industry to meet the ACARE 2020 and 2050 NO_x goals, innovative combustor designs must be introduced. Intense aero engine combustion research is undertaken in Europe. As an example it can be mentioned that within the European research project NEWAC the research goals for the combustor designs were defined as a 60 to 70% reduction relative the ICAO CAEP/2 regulation which is close to the ACARE 2020 goal. Some results for the lean combustion technologies studied within NEWAC project are presented in Figure 28 (Rolt and Kyprianidis, 2010).

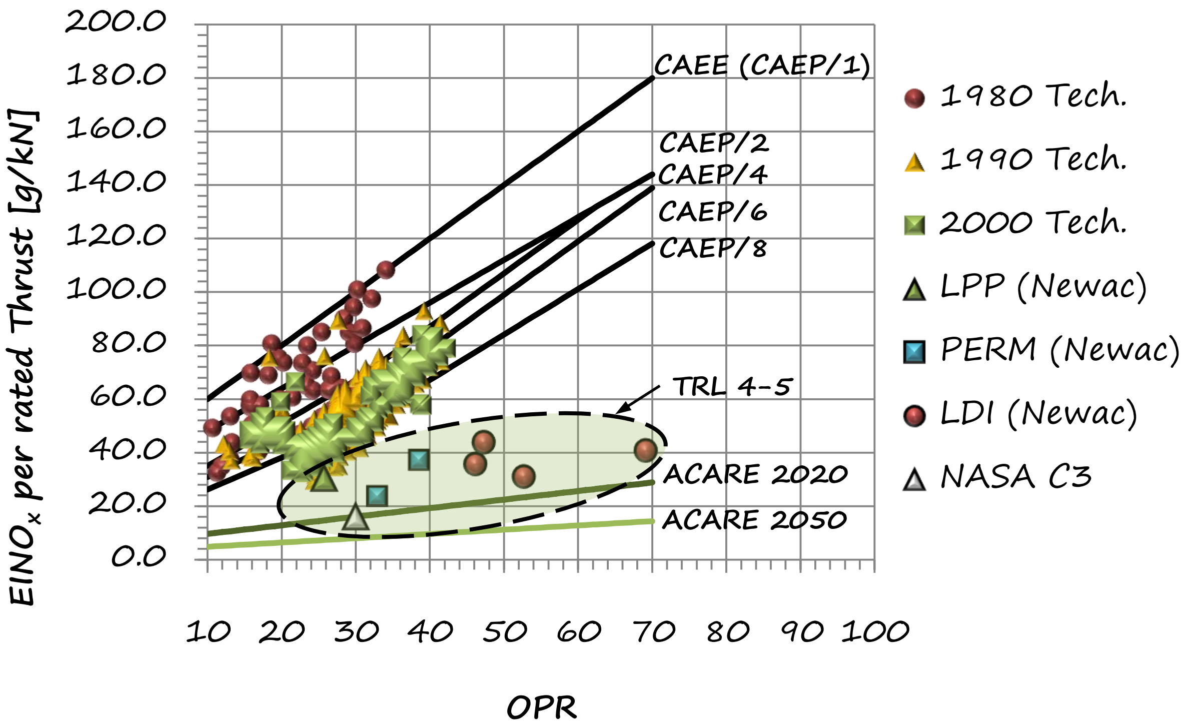


Figure 28. Emissions of NO_x per rated thrust as a function of overall pressure ratio for ICAO certified engines and results from lean combustion research projects as presented by NASA and NEWAC.

The absolute level of emissions of nitrogen oxides is basically dependent on two variables; the overall efficiency of the engine, SFC, and the NO_x emission index. Expressed in terms of an equation it can be described as;

$$\dot{NO_x} = SFC \times T \times EINO_x(P_3, T_3, \phi) \qquad (66)$$

Which states that for a given thrust, the NO_x level is a function of the SFC and the combustor NO_x index which in turn is dependent on compressor discharge temperature and pressure, equivalence ratio, quality of the mixing process, residence time etc. The relation described by equation (66) shows a typical aero engine design trade-off; the quest for lower SFC indicates engine designs utilizing higher pressures and temperatures which imply higher values of NO_x.

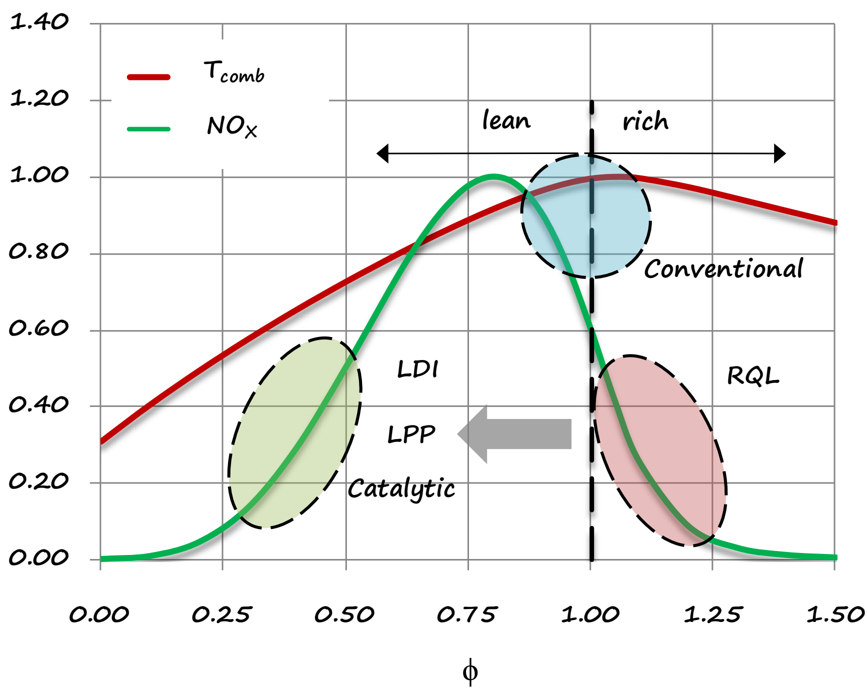


Figure 29. Conceptual illustration of different combustion concepts for achieving low NO_x emissions. The ordinate shows normalized temperature and NO_x. The equivalence ratio, ϕ, of the abscissa refers to local values for the combustor primary zone.

Conventional combustors operate close to stoichiometric conditions in the primary zone at temperatures well above the temperature at which thermal NO_x start to form, see Figure 29. Downstream the primary zone, the remaining combustor air is mixed with the combustion gases to reduce the temperature down to the turbine inlet temperature requirement.

Common for lean burn concepts, is that the combustion peak temperature is reduced by stabilizing the combustion at lower equivalence ratios, i.e. using excess air in the primary zone and therefore lowering the NO_x emissions. Some of the different concepts are briefly described here.

The Rich-burn, Quick-quench, Lean-burn (RQL) concept is designed for preventing stoichiometric conditions by a three-stage process. First, the flame is stabilized by a fuel-rich primary zone with the remainder of the combustion air rapidly mixed in the quench zone. In the lean-burn zone the burning is completed at relatively low temperatures and NO_x emissions as a consequence. One key to success for this concept is a properly designed quenching zone avoiding stoichiometric conditions and subsequently high NO_x emissions by mixing faster than the reactions complete.

The Lean-, Pre-mixed-, Pre-vaporized (LPP) concept relies on properly defined inlet conditions at the combustor entrance achieved by eliminating droplets by pre-vaporizing the fuel and mixing the fuel and air to a uniform lean mixture before entering the combustor. Combustion then occurs at lean conditions, possible close to the lean extinction limit and subject to auto-ignition and flashback. In an attempt to overcome these deficiencies the lean direct injection (LDI) has been proposed in which, and in contrast to the LPP concept, all of the fuel is injected at the combustor entrance where it is simultaneously mixed, vaporized and burned, minimizing the problems with auto-ignition and flashback.

Of the concepts mentioned so far, the LPP concept show potential for the lowest NO_x emission levels (Tacina, 1990), but NASA and other research institutions, have focused their combustion research on the LDI concept (Lee et al., 2007) possibly due to the fact it is more viable in the near future and suitable for high OPR engines.

All of the concepts described so far refer to homogeneous lean combustion concepts. Homogeneous in this case refers to the use of a gaseous state within the system; i.e. the reaction of a gaseous fuel with a gaseous oxidizer. The different homogeneous lean combustion concepts are in practice different ways of mixing the fuel with the air prior to the combustion zone. There is also another category of lean combustion concepts called heterogeneous lean combustion concepts which also involves a solid state in the reaction process. Ideally, this solid state is a material that lowers the activation energy significantly without participating in the reaction itself, and therefore providing stable combustion at very low equivalence ratios and resulting in ultra-low NOx emissions. One of these possible concepts, catalytic combustion, is one of the most promising known concepts for achieving low NO_x emissions. Catalytic combustion for aero engines has been studied by NASA within the clean catalytic combustor program (C^3) (Ekstedt et al., 1983). Combustor designs incorporating catalytic reactors were designed, built and tested at TRL 4 – 5 showing very low emissions of NO_x and excellent combustor performance. However, the concept suffers from similar problems as the LPP concept regarding the fuel/air preparation inlet system and in addition to this, the steady-state temperature capability of the catalytic reactor materials is a field that need improvement before it would be possible for this concept to be applied in aero engines.

9 Practical Considerations and Limits of Aero Engine Design

9.1 Design point(s)

9.1.1 Thermal Design Point (TDP)

The thermal design point is the most demanding part of the flight mission in terms of engine temperatures. This point is usually assumed to be the hot-day, take-off point.

9.1.2 Aero Design Point (ADP)

The aero design point is usually taken to be the top-of-climb point. This point is the point of the highest corrected mass flow, and typically sets the gas path dimensions.

9.1.3 Energy Design Point

A point during the flight mission where the average cruise flight takes place. During the cruise flight it is of great importance to minimize fuel burn, i.e. energy efficiency, since most of the flight is taking place here. This is especially important for long distance aircraft.

9.2 Constraints

9.2.1 HPC Exit Temperature

Lower SFC requires higher overall pressure ratios and turbine temperatures. Current aero engines in service utilize overall pressure ratios in the range of 30 – 40 at ISA, SLS. Already at these pressure ratios, the compressor exit temperature reaches 850 K which is close to the upper limit for materials that are widely used in compressors. Going to even higher pressure ratios will require high-temperature materials for the last compressor stages and this will of course have a negative impact on weight and cost of the engine.

9.2.2 Turbine Cooling

As for the HPC exit temperature constraint, the need for lower SFC implies higher turbine temperatures and this will increase the demand for advanced turbine cooling. The cooling technology level can be expressed in terms of cooling effectiveness, ε, according to,

$$\varepsilon = \frac{T_{gas} - T_{met}}{T_{gas} - T_{cool}}$$

For a given bulk gas temperature the reduction in metal temperatures achieved by cooling can be further increased by either increasing the efficiency of the cooling process itself, or increasing the cooling mass flow. As an alternative the maximum allowable metal temperature may be increased by developing new materials with improved high temperature properties. Since the introduction of the jet engine, the turbine materials have become very advanced super-alloys designed for operating metal temperatures above 1200 K.

9.2.3 Low-Cycle Fatigue

The number of cycles to failure due to low-cycle fatigue (LCF) can be quantified by using the empirically derived universal slopes method, which is a modified version of the Coffin-Manson & Basquin equations (Manson, 1965).

$$\Delta\varepsilon_{total} = \Delta\varepsilon_{elastic} + \Delta\varepsilon_{plastic} = 3.5\frac{\sigma_{ult}}{E}N_f^{-0.12} + \varepsilon_f^{0.6}N_f^{-0.6} \qquad (67)$$

where N_f is the number of cycles to failure, σ_{ult} is the ultimate tensile strength, E is Young's modulus, ε_f is ductility and $\Delta\varepsilon$ is the strain range.

9.2.4 High-Cycle Fatigue

For estimating high cycle fatigue (HCF) for aero engine conceptual design it is appropriate to use S-N diagrams, also called Wöhler diagrams, for the particular material under study. The S-N curve is derived from material tests where the specimen has been subject to a cyclic stress S and the number of cycles until failure of the specimen is determined. For some materials, e.g. steel, the S-N curve eventually flattens out so that no matter how many stress cycles that are applied the specimen will not fail. The material is said to have a certain stress endurance limit, and the designer could possibly eliminate the HCF-problem by not allowing the stresses exceed this limit. While some materials have this endurance limit, other materials do not, e.g. aluminum. For these cases the designer must take into account the number cycles to failure at a certain cyclic stress level.

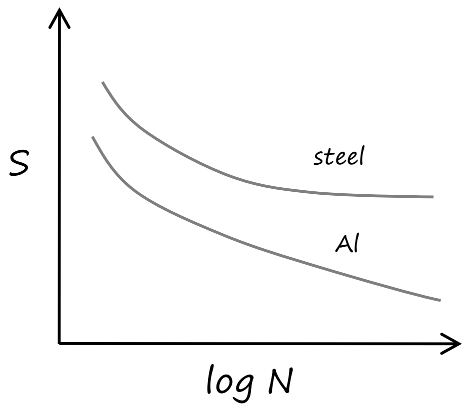


Figure 30. Conceptual sketch of the S-N diagram.

9.2.5 Creep

Creep can somewhat simplified be described as the progressive deformation of a material that occurs under mechanical stress at high temperatures. Creep is assumed to become a potential problem when operating temperatures exceed 50% of the material melting point (Naeem, 2009). There are several methods available for estimating creep and one of the most cited is the one by Larson and Miller (Larson and Miller, 1952). According to this method, the time to creep rupture, t_f, of a material at a given stress level will vary with the temperature in such a way that the Larson-Miller parameter, P_{LM}, remains constant, i.e.,

$$P_{LM}(\sigma) = \frac{T\left(log t_f + C\right)}{1000} \qquad (68)$$

In which C is a material constant.

10 Discussion & Concluding Remarks

Since the Wright brothers first powered flight in 1903, the old aerospace paradigm of flying higher and faster has pushed the development during the major part of the twentieth century with the prime era of the NASA space flight program, the Boeing 2707 supersonic transport, the Concorde and indeed all the military all the military aircraft during this period. During the early 1970s, in the shadow of, and in the light of, the energy crisis, the aerospace industry experienced a slight change in this mind set and the quest for more energy efficient air transport solutions was raised. One important consequence of this was the broad search for innovative aircraft and engine designs that was initiated by the U.S. Congress in response to the energy crisis congress. This was probably the first time in history that a government called for innovative energy efficient solutions in order to meet the demands from the society concerning greener air transports.

Today, the aeronautical research and development community is more prone than ever to search for innovative solutions, in particular since the improvement rate of change is decelerating somewhat in terms of energy efficiency, which still is far from any physical limits of aero engine and aircraft design. At the same time the society intensively calls for greener air transport, most likely as a consequence of the climate reports produced by the Intergovernmental Panel on Climate Change (IPCC) and the impact of aviation on the global atmosphere.

The work presented in this thesis was performed in two parts; development of new methods necessary for performing multidisciplinary optimization of the aircraft and engines and application of those methods to produce and assess innovative engine concepts that could have the potential to bring the aerospace industry closer to, and eventually beyond the ACARE 2020 vision. In particular, the methods developed within the scope of this thesis focused on modelling the impact on aircraft performance, and in particular mission fuel consumption, of varying engine size and weight during optimizations. These methods have been incorporated into a computer program called GISMO. A great benefit with multidisciplinary tools is the interdisciplinary quantification of component design changes and the impact of those on the overall goal function, in contrast to intradisciplinary quantification of design changes. Also developed within the scope of this thesis were a method for designing and analyzing propeller performance, in particular the performance of counter-rotating propellers.

The studies has been limited to the conceptual design of aero engines, meaning thermodynamic cycle optimizations including aircraft performance, engine weight and engine dimensions. This has allowed evaluating a number of solutions for minimizing emissions of CO_2 and NO_x.

Initial studies attempted to quantify the difference in optimizing a new aero engine design using the detailed connection between the engine(s) and aircraft compared to a multipoint optimization using the engine performance tool disconnected from the aircraft model. The study showed the importance of optimizing the engines coupled to the aircraft.

For a given aircraft and engine, a simple, but powerful method of minimizing the direct operating cost for an airline is to carefully choose the flight speed depending on the current fuel prices. By applying the methods developed in this thesis work, studies were carried out in order to quantify the impact of fuel price variations for airlines, and the potential fuel savings that could be obtained by carefully selecting the flight speed in order to compensate for those variations. Another study that resulted from the methods developed quantified the potential savings in aero engine maintenance cost, i.e. component design life, by trading climb thrust and time to first cruise altitude. More specifically by applying derated thrust during the climb phase of the flight missions the results indicate a 7% engine life increase for a 0.7% increase in climb fuel, or 0.1% increase in total flight mission fuel consumption. It is pointed out that the results should be on the conservative side due to the fact that the effects of lowered blade metal temperature on oxidation and hot corrosion of the blade surface layers are not taken into account.

Concepts for increasing energy efficiency and potentially reducing NO_x were studied within this thesis. From the variable cycle engine study it was concluded that by varying the flow by means of variable geometry during the flight mission, a 5% reduction in fuel consumption could be obtained when comparing to the conventional turbofan on a typical medium range aircraft. It is also noted that the trend of aero engine design is ever increasing BPR which mean smaller cores, which in turn mean higher core pressures and temperatures at off-design. It is then questionable whether this relatively complex cycle will be needed in the future?

The inter-turbine reheat concept was also assessed. It is widely appreciated that the inter-turbine reheated cycle will increase the specific power, or thrust, by a substantial amount compared to the conventional cycle and also that the efficiency will be lower unless any form of waste heat recovery system is applied. However, in this study it was shown that there exist inter-turbine reheated cycles that for a given maximum turbine temperature will be more efficient that the conventional cycle and that this will likely occur at a higher overall pressure ratio than that of the conventional cycle. It is also suggested that the point of maximum efficiency is located quite early in the expansion process. Mission optimizations of the reheated engine and the conventional engine resulted in potential NO_x emissions reduction in the range 21 to 35%.

Regarding energy efficiency, the results show that the reheated engines will have small improvements, close to 1%, compared to the conventional re-optimized turbofan engine for combustor exit temperatures up to 1800 K. Further increase in temperature will be more beneficial for the conventional engine, as a result of the increased cooling requirement of the intermediate-pressure turbine for the reheated engine.

In terms of NO_x reductions, a variable cycle design that incorporates a catalytic combustor was studied and also patented during this thesis work. In the patent and the related paper, an idea is presented of how catalytic combustors could be introduced by the application of a variable cycle engine design that would use the catalytic reactor only for the colder cruise phase of the flight mission. The results indicated major NO_x reductions as expected, in the order of 22 to 46%, but at the expense of a fuel burn increase. Although the engine cycle proposed here is rather complex requiring a number of bleed valves for switching between cruise and maximum thrust settings, it is still an interesting and possible way of implementing catalytical reactors in aero-engines for lowering NO_x emissions substantially. The performance analysis of this particular engine cycle is not completed at this time, further analysis would be necessary to determine the particular benefit of this engine cycle.

An idea of idea of how to design and build more efficient air propellers in particular counter-rotating propeller is introduced. The idea has been filed as an international patent application.

The conventional propeller limits the flight speed of the aircraft, since the combination of flight speed and propeller rotational speed causes performance losses and noise due to shocks forming at the propeller tip. However, by introducing swept propeller blades with advanced airfoils the propeller can be operated at higher speeds with acceptable performance loss.

This invention seeks to improve the propeller design even more, by introducing non-planar propeller blades, so called box-blades. Initial studies indicate that a box-blade could reduce the propeller blade induced drag by some 20 to 50% of the propeller and also the blade root bending moment by some 20 to 80% depending on the final configuration. The idea originally stems from Prandtl's work from the 1920s (Prandtl, 1924) in which he showed that the wing system for minimum induced drag was a box-wing system, i.e. a box-like wing with equal lift and lift distribution over the two horizontal wings.

This propeller design is interesting from several points-of-view; firstly it applies the Prandtl theory for a minimum induced drag system, secondly it is very interesting from a mechanical design perspective, since the configuration itself will be stronger than an equally loaded single blade and could therefore possibly allow thinner and/or forward-sweep of the propeller blades which in turn have the potential to further increase aerodynamic benefits of the propeller. This invention will be further evaluated by experiments.

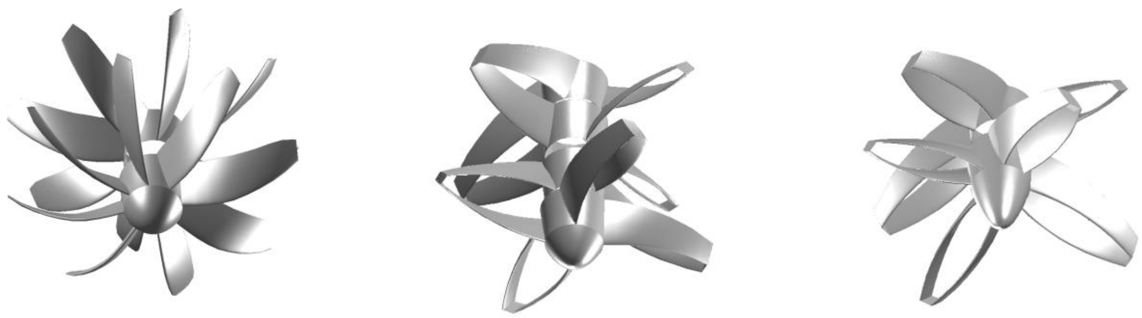


Figure 31. A conventional counter-rotating propeller to the left, forward-aft swept box-bladed design in the middle and a forward-swept box-bladed design to the right.

11 Technology Assessments Performed and new Ideas Produced

The technologies assessed and the ideas produced within the scope of this thesis are presented in greater detail in the papers attached to this thesis.

11.1 Preliminary Design of Subsonic Transport Aircraft and Engines

Paper I. This paper was presented at the 18th ISABE meeting in Beijing, China in 2007. The paper was the first publication during the authors work. The overall purpose of the work was to present a new method, GISMO, for evaluating the coupling of the engine(s) and the aircraft in terms of weight and drag caused by varying engine size and technology. In particular, the research questions to be answered were defined as;

1. *What's the meaning of the term optimum aircraft engine?*
2. *What are the effects of leaving the aircraft out of the engine design process, if any?*

The first question might sound a bit vague, however the meaning of the question, and especially the answer, in this context was to illuminate the importance of being able to quantify the requirements, trade-offs and possibly conflicting goal functions concerning aero engine conceptual design. This was done to motivate the aero engine performance engineer's need for basic knowledge about aircraft aerodynamics and weight modeling. So, what is the answer to this question? The answer is that the optimum aircraft engine is the engine that best fulfills or even surpasses the customer's specification. The challenge is of course to be able to select the optimum engine. To be able to do this one must be able to quantify many trade-offs and couplings between the engine and aircraft.

The second question is not easily answered. This particular study only gives an indicative measure of what the differences in optimum engine design might be when leaving the coupling of varying engine size and aircraft out of the equation. If you want to create the way forward and generate real value for the customer it might be of great value to understand the impact of design changes along the product development process, in particular when starting a new R&D program or when entering a new engine development program.

Some concluding remarks concerning this particular publication would be that the study presents new methods for quantifying the aircraft and engine coupling, and an attempt to quantify the effect of leaving it out of the equation. An important result was the implementation of the modified Korn equation to properly model the transonic drag rise. It should also be pointed out that there are many constraints not imposed within this study, but those who are handled answers the research question asked.

11.2 Potential of Variable Cycle Engines for Subsonic Air Transport

Paper II. This paper was presented at the 18th ISABE meeting in Beijing, China in 2007. The purpose of the study and the research question to be answered was confined to;

1. *Quantifying the potential performance benefits of incorporating variable geometry in the conventional turbofan engine.*

The author's contribution to this study was to provide the performance- and flight mission simulations for assessing the potential of the variable cycle turbofan engine.

The potential for incorporating variable geometry in turbofan engines is quite substantial. A fundamental "design problem" with aero engines is that they are throttled down to lower power settings when entering the cruise phase at altitude, and the pressures and temperatures are lowered resulting in a reduced thermal- and overall efficiency.

Within this study a variable cycle engine is defined as one where a part power thrust can be achieved with significantly different mass flows and pressure ratios than for a constant geometry machine with the same cycle at max thrust. This would be achieved by varying the geometry at selected engine sections in particular the turbine and nozzle areas so that the fan – or the core flow is varied accordingly.

The study gives an indication of the potential benefit of incorporating variable cycle engines for commercial subsonic transports. More design constraints need to be added to the process, e.g. weight, mechanical design and cost before attempts are made to start development of this type of engine.

It is concluded that applying variable cycle engine, by varying the flow during the flight mission, could yield a 5% benefit in fuel consumption when compared to the conventional turbofan on a typical medium range aircraft. It should be noted that the analysis assumed that the variable component geometry would not reduce component efficiencies, moreover the trend of aero engine design is ever increasing BPR engines which mean smaller cores which in turn mean higher core pressures and temperatures at off-design. It is then questionable whether this relatively complex cycle will be needed in the future?

11.3 Minimizing Direct Operating Costs (DOC) for a small European Airline

Paper III. This paper was presented at the 18th ISABE meeting in Beijing, China in 2007. The purpose of the study was to explain the commonly used cost index parameter, CI, used by airlines to fly at maximum profitability by compensating for variations in fuel price by adjusting the aircraft speed according to the cost index.

The author’s contribution to the study was the performance- and flight mission analyzes performed to quantify cost index variations of typical medium range aircraft compensating for fuel price variations. The results indicate a 3-4% reduction in fuel consumption by choosing a lower flight speed, i.e. minimum fuel speed, for a typical mission.

The scientific value of the study is modest, however the engineering value should be greater, in particular for the end customer, i.e. the airlines.

11.4 An Assessment of a Turbofan Engine Using Catalytic Interturbine Combustion

Paper IV & VIII. This paper was presented at ASME Turbo Expo 2009, Orlando, Florida, USA. The patent was filed on July 2, 2009. The purpose of the paper and patent was;

1. *To re-introduce the idea of using catalytic combustion in aero engines, in order to obtain ultra-low NO_x emissions.*
2. *Introduce an idea of how catalytic combustion could be implemented in turbofan engine despite the very high engine temperatures used in modern engines.*

In comparison to other well-known aero-engine combustor concepts for achieving ultra-low levels of NO_x emissions, the catalytic combustion concept was proven already in the early 1980s to have the potential to reduce NO_x by some 70% relative the ICAO CAEP/2 regulation as described in section 8 in this thesis. Also concluded at that time was the problem with the long-term high-temperature stability of the catalytic reactor materials which made the concept not viable in a near future.

In the patent and the related paper, an idea is presented of how catalytic combustors still could be introduced by the application of a variable cycle engine design that would use the catalytic reactor only for the colder cruise phase of the flight mission.

The results indicated major NO_x reductions as expected, but at the expense of a fuel burn increase. Although the engine cycle proposed here is rather complex requiring a number of bleed valves for switching between cruise and maximum thrust settings, it is still an interesting and possible way of implementing catalytical reactors in aero-engines for lowering NO_x emissions substantially. The performance analysis of this particular engine cycle is not completed at this time, further analysis would be necessary to determine the particular benefit of this engine cycle.

11.5 Derated Climb Trajectories for Subsonic Transport Aircraft and their Impact on Aero Engine Maintenance Costs

Paper V. This paper was presented at the 19th ISABE meeting in Montreal, Canada in 2009. The purpose of this study was to quantify the effect of applying derated thrust setting during the climb phase of the flight mission. The particular research question to be answered was formulated as;

1. *What is the potential of applying derated thrust setting during the climb phase, in terms of aero engine life (maintenance costs), if any?*

The author's contribution to this study was modeling of the wide-body aircraft used for this particular study, the performance and mission analysis and modeling of the low cycle fatigue and creep in the critical components of the engine.

In general the hot section parts of an aero engine constitute the critical parts with respect to component design life, typically the combustor and the high-pressure turbine. The quantification of a particular aero engine component's design life is a complex task involving modeling of both thermal and mechanical loads such as centrifugal forces, gas bending moments, cyclic loading, metal temperature, thermal gradients to mention a few. For the purpose of this particular trade-off study it is assumed that the HPT is the critical component with respect to engine life, i.e. total engine cycles as a function of LCF and creep, and that the life is limited by HPT blade metal temperature and the centrifugal forces at the blade root.

By applying first-order empirical methods for quantifying LCF- and creep life together with publicly available tensile data for the CMSX-4 superalloy the study concluded a 7% engine life increase for a 0.7% increase in climb fuel, or 0.1% increase in total flight mission fuel consumption. It is pointed out that the results should be on the conservative side due to the fact that the effects of lowered blade metal temperature on oxidation and hot corrosion of the blade surface layers are not taken into account.

11.6 Aeroacoustics and Performance Modeling of a Counter-Rotating Propfan

Paper VI. The paper was presented at ASME Turbo Expo 2010 in Glasgow, U.K. The purpose of the study was to develop a method for design and analysis of counter-rotating propellers with respect to performance and aero-acoustics.

As the search for more fuel efficient aero engines is intensified the open rotor or propfan engine, once more seems to constitute a possible future solution. This was observed by the research group and a design and analysis methodology was initiated. The purpose of this work was to apply the group’s available methods and to develop new methods for designing and especially analyzing the performance and aero-acoustics of a counter-rotating propfan.

The author’s contribution to this study was a methodology to design and analyze the performance of modern counter-rotating propellers intended for flight-speeds normally utilized by jet engines. This was done by applying the well-established methods by Theodorsen for determination of optimum ideal propeller designs that gives an optimum blade loading. The ideal results are then corrected for profile and compressibility losses at various operating points so that the propeller performance can be estimated.

The methods developed by Theodorsen for prediction of optimum propeller designs (Crigler, 1949, Theodorsen, 1948, Theodorsen, 1944a, Theodorsen, 1944b, Theodorsen, 1944c, Theodorsen, 1944d) are rather old but still very useful in modern propeller design problems.

In order to evaluate this proposed design and analysis method, data was collected from the publicly available literature from various tests of the GE36 counter-rotating propeller (UDF). The GE36 propeller was designed and analyzed for the cruise flight condition showing an agreement within 3% in terms of net thrust and 1.5% in propeller efficiency.

The method should be usable in order to generate propeller performance maps of new propellers, i.e. power, thrust and efficiency for use in complete flight mission simulations. The method is not fully evaluated in corners of the performance envelope.

11.7 Potential Benefits of Using Inter-Turbine Reheat in Turbofan Engines

Paper VII. The paper will be submitted to the Journal of Engineering for Gas Turbines and Power during fall 2011.

The purpose of this paper was to study the potential of inter-turbine reheated turbofan engines in terms of energy efficiency.

It is of common and basic understanding the use of a second combustor downstream the first in a Brayton cycle will increase the specific power, or thrust by a substantial amount. It is also a common appreciation that an engine utilizing inter-turbine reheat will have a lower efficiency, and consequently higher SFC, than its conventional counterpart unless any form of waste heat recovery system is applied. However, in this study it is shown that this is not necessarily the case, in fact there exist reheated cycles that for a given maximum turbine temperature will be more efficient that the conventional cycle under the same maximum temperature constraint. The reheat cycle optimum pressure ratio will likely be higher than the conventional optimum pressure ratio. Furthermore it is well known and shown that the maxium specific power, or thrust, occurs when the reheat combustor is introduced at the point during the expansion when the pressure ratio upstream and downstream the reheat position are equal. It is also suggested that the point of reheat for maximum efficiency occur quite early during the expansion.

This hypothesis is applied to the turbofan case and mission optimizations of the reheated engine and the conventional engine are performed.

The potential of reducing NO_x emissions by the use of a secondary combustor is also discussed and quantified to some extent. First order effects on engine NO_x emission levels due to combustion in oxygen depleted air in the secondary combustor are quantified. The results indicated NO_x emissions reductions between 21 and 35% at the max rated thrust setting depending on the maximum design exit temperature that was allowed.

In terms of energy efficiency, the results show that the reheated engines will have small improvements compared to the conventional re-optimized turbofan engine, but still an improvement. The improvement will decrease as the maximum design turbine temperature increases and is a result from the increased cooling requirement from the intermediate-pressure turbine downstream the secondary combustor.

The technology studied here is quite innovative and is assumed to be introduced on the market beyond the year of 2030 which is an important assumption to have in mind when reading this paper. The optimum engine designs reveals for instance overall pressure ratios in the range of 80 to 100 which today would require cooling of the last compressor stages. Same thing for the secondary combustor liner that would have to be cooled using today's metal liner materials. If compressed air for cooling of the secondary would be used, then the high-pressure turbine would have to be by-passed and the overall efficiency would decrease accordingly.

This study does not assume any compressor air for cooling of the secondary combustor, instead it is assumed that the liner could be manufactured in ceramic materials or ceramic composite materials in line with current development and research in the field of combustor liner materials. The quest for ultra-low emissions technologies calls for combustor liner materials that can withstand high temperatures with little or no film-cooling. This has triggered research in the field of advanced combustor liner materials, and in particular the ceramic materials are promising since they are resistant to oxidation, have good mechanical strength at temperatures well above unprotected metals. The silicon compounds are considered most promising with monolithic silicon and silicon carbide exhibiting high strength and stiffness up to 1680 and 1880 K respectively (Lefebvre, 1999). Furthermore, technology demonstrations by NASA and Solar Turbines of silicon carbide ceramic matrix composite liners has resulted in over 9000 hours at elevated temperatures of some 1500 K in laboratory aero engine cycle tests and also in gas turbine tests (NASA, 2011, Roode et al., 2007). If still cooling will be required at the year of 2030 or beyond, there are creative ways of realizing this without bypassing the HPT with compressed air and loosing efficiency; see for instance the patent application regarding an innovative reheat combustor for a gas turbine by GE (Dinu, 2009).

11.8 Propeller Arrangement and Aircraft

Paper IX. This patent application was filed on July 7, 2011. The purpose of this invention is to further develop an idea of how to design and build more efficient air propellers, in particular for the current research state-of-the-art counter rotating propellers.

The conventional propeller limits the flight speed of the aircraft, since the combination of flight speed and propeller rotational speed causes performance losses and noise due to shocks forming at the propeller tip. However, by introducing swept propeller blades with advanced airfoils the propeller can be operated at higher speeds with acceptable performance loss.

This invention seeks to improve the propeller design even more, by introducing non-planar propeller blades, so called box-blades. Initial studies indicate that a box-blade could reduce the propeller blade induced drag by some 20 to 50% of the propeller and also the blade root bending moment by some 20 to 80% depending on the final configuration. The idea originally stems from Prandtl’s work from the 1920s in which he showed that the wing system for minimum induced drag was a box-wing system, i.e. a box-like wing with equal lift and lift distribution over the two horizontal wings.

This propeller design is interesting from several points-of-view; firstly it applies the Prandtl theory for a minimum induced drag system, secondly it is very interesting from a mechanical design perspective, since the configuration itself will be stronger than an equally loaded single blade and could therefore possibly allow thinner and/or forward-sweep of the propeller blades which in turn have the potential to further increase aerodynamic benefits of the propeller.

This invention will be further evaluated by experiments.

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Appendix A - Orville Wright, diary entry for 17 December 1903

The diary entry by Orville Wright is quoted from the historical NASA publication "The Wind and Beyond" (Hansen, 2003) and gives the reader the opportunity to experience one of the most important events in aviation history, as told by the Wright brothers themselves.

Thursday, December 17, 1903

When we got up a wind of between 20 and 25 miles was blowing from the north. We got the machine out early and put out the signal for the men at the station. Before we were quite ready, John T. Daniels, W. S. Dough, A. D. Etheridge,W. C. Brinkley of Manteo, and Johnny Moore of Nags Head arrived. After running the engine and propellors a few minutes to get them in working order, I got on the machine at 10:35 for the first trial. The wind, according to our anemometers at this time, was blowing a little over 20 miles (corrected) 27 miles according to the Government anemometer at Kitty Hawk. On slipping the rope the machine started off increasing in speed to probably 7 or 8 miles. The machine lifted from the track just as it was entering on the fourth rail.

Mr. Daniels took a picture just as it left the tracks. I found the control of the front rudder quite difficult on account of its being balanced too near the center and thus had a tendency to turn itself when started so that the rudder was turned too far on one side and then too far on the other. As a result the machine would rise suddenly to about 10 ft. and then as suddenly, on turning the rudder, dart for the ground. A sudden dart when out about 100 feet from the end of the tracks ended the flight. Time about 12 seconds (not known exactly as watch was not promptly stopped). The lever for throwing off the engine was broken, and the skid under the rudder cracked. After repairs, at 20 min. after 11 o'clock Will made the second trial. The course was about like mine, up and down but a little longer over the ground though about the same in time. Dist. not measured but about 175 ft. Wind speed not quite so strong. With the aid of the station men present, we picked the machine up and carried it back to the starting ways. At about 20 minutes till 12 o'clock I made the third trial. When out about the same distance as Will's, I met with a strong gust from the left which raised the left wing and sidled the machine off to the right in a lively manner. I immediately turned the rudder to bring the machine down and then worked the end control. Much to our surprise, on reaching the ground the left wing struck first showing the lateral control of this machine much more effective than on any of our former ones. At the time of its sidling it had raised to a height of probably 12 to 14 feet. At just 12 o'clock Will started on the fourth and last trip. The machine started off with its ups and downs as it had before, but by the time he had gone three or four hundred feet he had it under much better control, and was traveling on a fairly even course. It proceeded in this manner till it reached a small hummock out about 800 feet from the starting ways, when it began its pitching again and suddenly darted into the ground. The front rudder frame was badly broken up, but the main frame suffered none at all. The distance over the ground was 852 feet in 59 seconds. The engine turns was 1071, but this included several seconds while on the starting ways and probably about a half second after landing. The jar of landing had set the watch on machine back so that we have no exact record for the 1071 turns. Will took a picture of my third flight just before the gust struck the machine. The machine left the ways successfully at every trial, and the tail was never caught by the truck as we had feared.

After removing the front rudder, we carried the machine back to camp. We set the machine down a few feet west of the building, and while standing about discussing the last flight, a sudden gust of wind struck the machine and started to turn it over. All rushed to stop it. Will who was near the end ran to the front, but too late to do any good. Mr. Daniels and myself seized spars at the rear, but to no purpose. The machine gradually turned over on us. Mr. Daniels, having had no experience in handling a machine of this kind, hung on to it from the inside, and as a result was knocked down and turned over and over with it as it went. His escape was miraculous, as he was in with the engine and chains. The engine legs were all broken off, the chain guides badly bent, a number of uprights, and nearly all the rear ends of the ribs were broken. One spar only was broken.

After dinner we went to Kitty Hawk to send off telegram to M. W. While there we called on Capt. and Mrs. Hobbs, Dr. Cogswell and the station men.