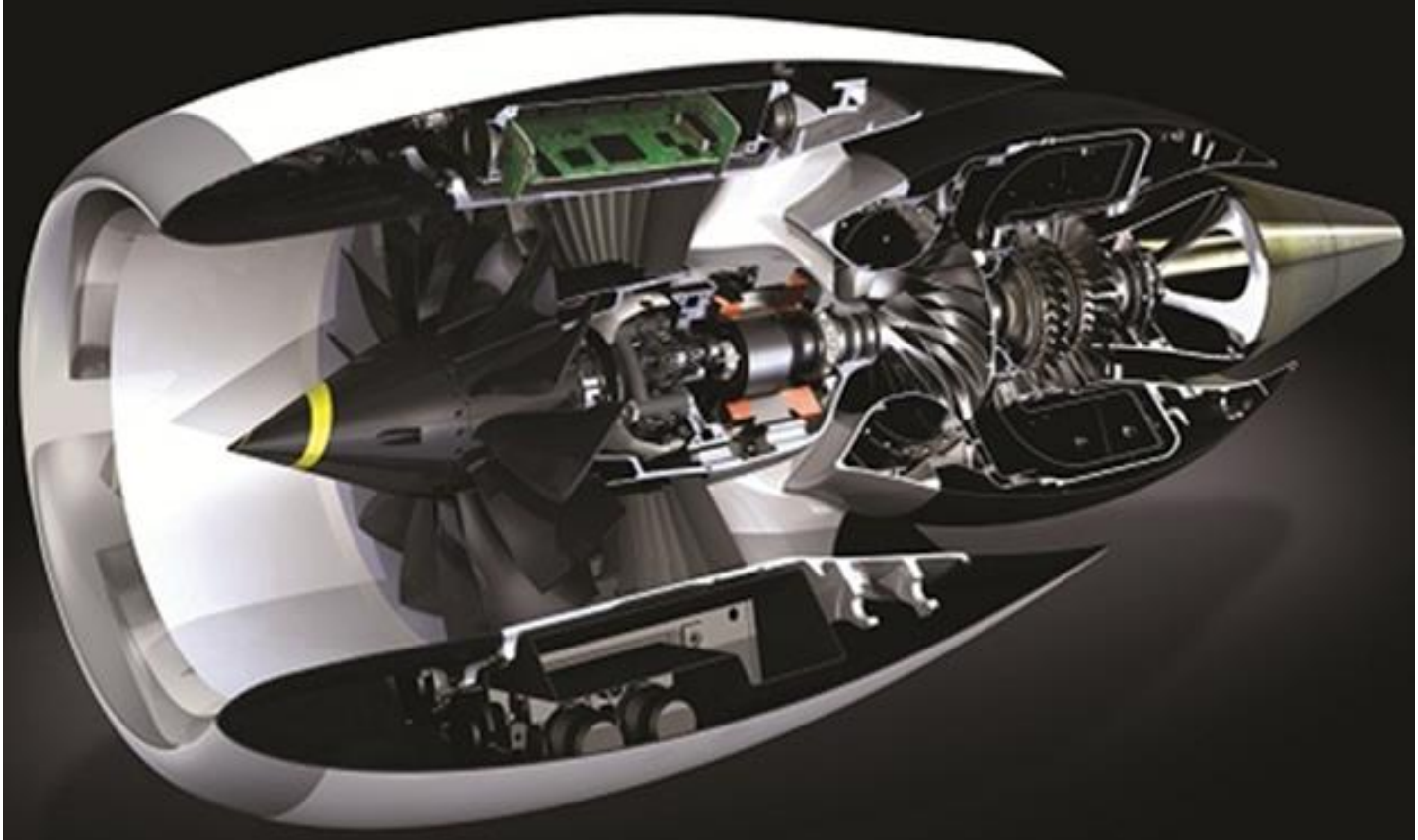
PROJECT POWERPLANT



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Summary

Amsterdam Aircraft Engines (AAE) has received a request to develop new gas turbine engines for the Fokker 70. The Fokker 70 is currently equipped with Rolls Royce TAY 620-15 engines, which have a comparatively low efficiency and therefore high fuel consumption. In order to offer the same payload-range capabilities as the Fokker 70 with centre fuel tank modification, AAE's conceptual design for the new engine needs to be more powerful and more fuel-efficient. The conceptual design of the engine has to comply with some requirements. Firstly, the engine has to be efficient enough to reach a range of 1200NM at MZFW under ISA conditions with zero wind, long range cruise, EU-OPS 1.255 reserves and a 100NM alternate. Secondly, the engine should provide a thrust to guarantee a 2.4% gross second segment climb at MTOW at an elevation of 5000ft under ISA conditions.

By adding modern improvements to the New Engine Design 2F (NED 2F), it is proved that it is possible to deliver an engine which has the required specifications needed for the range and thrust. A comparison is made with two comparable engines to see how the conceptual engine performs in the critical factors: Thrust Specific Fuel Consumption (TSFC), thrust, thrust/weight ratio and range. The conceptual design comes out as an overall best. Designing this new engine will require some investment, but due to the very modern techniques in this design, it is capable of delivering the same performances as the Fokker 70 with centre fuel tank modification.

For the required thrust a free body diagram of the specific climb situation is made. The minimum thrust needed for this climb can be calculated using this diagram, the moment's equilibrium and the lift-formula. This gives a required thrust of 40549N, which can already be delivered by the RR TAY 620. This thrust is validated by inserting the data at 5000ft in the calculations of the RR TAY 620.
The RR TAY 620 has a range of 550NM, according to the payload-range graph of Fokker. This means that the new engine has to be more than twice as efficient. This is verified by the range equation of Brequet for jet aircraft. By inserting the new TSFC of the NED 2F, it is proved that this engine is capable of reaching a range of 1200NM.

In order to reach these requirements with the calculated values, the NED 2F engine has to be designed with improvements. Firstly, computer improvements are analysed. Since the Fokker 70 is equipped with FADEC, it is highly recommended to equip the new engine with this Full Authority Digital Engine Control for improved reliability, performance and efficiency. A TAPS II combustor is recommended to decrease emissions. Secondly, improvements can be found in the thermodynamics and aerodynamics. By increasing the CPR through better compression, the engine's efficiency is consequently increasing as well. The twisted and swept fan blades permit a higher efficiency through a lower effective speed along the chord. Finally, material improvements will also help to reach the requirements. By adding materials that are capable of handling higher temperatures, the Turbine Inlet Temperature (TIT) can be increased, which results in a higher thrust.

In order to produce the NED 2F with its improvements, the RR TAY 620 is analysed. This analysis is preceded by a description of the thrust- and efficiency-formulas, this in order to know what variables they contain for further design. Then the construction and operation from beginning to end of the engine is described, this in order to know which parts can be further improved. Between and in these different sections, the thermodynamics such as the increase and decrease of pressure, temperature and velocity are of importance for the engine's efficiency.

Contents

Introduction

Amsterdam Aircraft Engines (AAE) has received a request to develop new gas turbine engines for the Fokker 70, at today's standards. Development of the Fokker 70 stopped after Fokker closed in 1996. The aircraft however is still in operation mainly due to its superior characteristics like short take-off ground roll, high rate-of-climb and benign stall characteristics, compared to other regional jets in the 70- to 100-seat class.

The current Fokker 70 engines have a comparatively low efficiency, therefore high fuel consumption, and do not meet the current emissions standards. The newly to be developed engines need to be more powerful, but also more fuel-efficient. This is a challenging task to perform. The assignment for the project group is to produce a conceptual design for a new gas turbine engine for the Fokker 70.

The report exists of four chapters.
The first chapter, analysis, is chosen by the team because of the importance to have knowledge about the old engine and Fokker 70. The team has to know which parts of the old engine can be improved and the calculations for the new engine can be validated by using the data from the old engine. Also this chapter is chosen to analyse the Fokker 70. This knowledge is required to investigate which improvements in the conceptual engine can be applied and to perform this with the least modifications needed regarding the F70. The construction, operation and the different systems and sub-systems of the RR-TAY are aspects needed to analyse. For the F70 the currently available systems and the construction of the engine mounting will be analysed. Also there are a few demands from the client. **(1)**
The second chapter, performance requirements, is chosen by the team because these need to be calculated before the engine can be designed. There are two hard requirements given for the new engine. Firstly the engine should have an increased efficiency so the range will increase to 1200 NM, secondly a 2,4% gross second segment climb is required. The range will be increased by reducing the TSFC, and to make sure that the aircraft could climb with the given slope the minimal thrust will be calculated. **(2)**
The third chapter, engine design, is chosen by the team because this chapter describes the modern improvements that can be applied on the new engine. Also it will describe the new engine after these improvements have been analysed. The calculations in chapter two will be applied on the new engine and the performance and concept of the new engine will be determined. **(3)**
The fourth chapter, comparison engines, is chosen by the team because it is important to show how our engine differs from the currently existing engines. This comparison is based on a few components which are most important to the team, TSFC, thrust, weight, thrust/weight ratio and range. From this comparison a conclusion can be made about whether the new designed engine is most suitable for the Fokker 70. **(4)**

One of the most important resources for this assignment were the presentations given by Mr. Bremer. At the end of this paper there is a bibliography for further information about the resources that have been used. The appendices are also at the end of this paper.

1 Analysis

In order to develop the conceptual design engine, knowledge about the RR TAY 620-15 and the F70 is required. This knowledge is required to investigate which improvements in the conceptual engine can be applied and to perform this with the least modifications needed regarding the F70. The construction, the operation of the engine and the different systems and sub-systems cooperating with the RR TAY are aspects needed to analyse **(1.1)**. For the F70 aspects like; currently available systems and the construction of the engine mounting are needed to analyse **(1.2)**. Also there are a few demands from the client **(1.3).**

1.1 Rolls Royce Tay-620 Analysis

The thrust and efficiency of the RR TAY 620-15 engine can be calculated using thermodynamic and aerodynamic formulas **(1.1.1)**. The RR TAY components & operation **(1.1.2)** and the Systems **(1.1.3)** will help understand the RR TAY turbofan. To validate the calculation method a test calculation **(1.1.4)** is made with the RR TAY.

1.1.1 Thermodynamics

Like any other turbofan engine the main goal of the RR TAY 620-15 is to deliver thrust **(1.1.1.a)**. A second requirement of the turbofan engine is to reach the highest possible efficiency **(1.1.1.b)**.

1.1.1.a Principle of thrust

A couple of factors play a role in the emergence of thrust. The first factor is the acceleration of the airflow within the gas turbine. Force equals mass times acceleration, so accelerating a certain mass of air causes a force, which in this case is the thrust. Within the gas turbine, a certain mass of fuel is added, and also accelerated. This means the acceleration of the fuel also assists in the establishment of thrust. In most cases, after the exhaust a higher pressure is present than the pressure of the surrounding atmosphere. The difference in pressure seen over the outflow surface is expressed in a forward force, or thrust. These three factors can be combined in the formula of thrust **(Equation 1-1)**.

Equation 1-1		
Thrust		
$$F_{thr} = \dot{m}_a * (C_j - C_i) + \dot{m}_f * v_j + A_j * (P_j - P_{am})$$		
Symbol	**Variable**	**Unit**
F_{thr}	Thrust	N
$\dot{m}_a$	Mass airflow	Kg/s
C_j	Velocity of airflow after the exhaust	m/s
C_i	Velocity of airflow before the inlet	m/s
$\dot{m}_f$	Mass fuel flow	kg/s
v_j	Velocity of airflow after the exhaust	m/s
A_j	Outflow surface exhaust	m^2
P_j	Static pressure after the exhaust	Pa
P_{am}	Static pressure surrounding atmosphere	Pa

The first law of thermodynamics states that energy can never be created or destroyed, but can only be transferred from one type of energy to another. In the gas turbine this principle is used to generate thrust. First the kinetic energy from the incoming airflow is transformed into potential energy. By adding and combusting fuel, extra potential energy is added to the air gas stream. After the combustion, the potential energy is again converted into kinetic energy, which delivers the thrust.

1.1.1.b Efficiency

In the best case scenario all the power of the ignited fuel would be converted into the propulsion of the aircraft. However, losses can never be circumvented. To see in which manner an engine transfers the power given by the fuel into wanted thrust, different types of efficiency can be calculated:

- Thermodynamic efficiency
- Propulsion efficiency
- Total efficiency

Ad 1. **Thermodynamic efficiency**

The useful power in a gas turbine is the acceleration of air. The higher the percentage of power that comes out of the accelerated air compared to the power added by the fuel, the more efficient the engine is. This kind of efficiency is called the thermodynamic efficiency **(Equation 1-2)**.

Equation 1-2		
Thermodynamic efficiency		
$$\eta_{th} = \frac{C_j - C_i}{Q}$$		
Symbol	**Variable**	**Unit**
η_{th}	Thermodynamic efficiency	%
C_j	Velocity of airflow after the exhaust	m/s
C_i	Velocity of airflow before the inlet	m/s
Q	Added energy	Joule

Ad 2. **Propulsion efficiency**

The engines accelerate the air. However, in the end it is not the air that we want to accelerate, but the aircraft. This is achieved by the third law of Newton, which states that the airflow accelerated by the engine exerts the same but opposite thrust given by the engine, to the engine. The higher the velocity of the aircraft compared to the velocity of the outgoing airflow, the more propulsion efficient the system is **(Equation 1-3)**. However, the delivered thrust of the engines is partly based on the difference between the aircraft its speed and the outgoing airflow. This means that when the thrust increases, the propulsion efficiency decreases, and vice-versa.

Equation 1-3		
Propulsion efficiency		
$$\eta_p = \frac{2}{1 + \frac{C_j}{Ci}}$$		
Symbol	**Variable**	**Unit**
η_p	Propulsion efficiency	%
C_j	Velocity of airflow after the exhaust	m/s
C_i	Velocity of airflow before the inlet	m/s

Ad 3. **Total efficiency**

Combining the thermodynamic efficiency with the propulsion efficiency, the total efficiency is formed **(Equation 1-4)**

Equation 1-4		
Propulsion efficiency		
$$\eta_{tot} = \eta_{th} * \eta_p$$		
Symbol	**Variable**	**Unit**
η_{tot}	Total efficiency	%
η_{th}	Thermodynamic efficiency	%
η_p	Propulsion efficiency	%

1.1.2 Rolls Royce TAY Components & Operation

The RR TAY 620-15 uses a number of stages to deliver the thrust. First, the incoming air is slowed down and compressed in the inlet **(1.1.2.a)**. The airflow passes the fan **(1.1.2.b)** where after it splits into two sections, the core and the bypass section **(1.1.2.c)**. The airflow through the core is further compressed in the compressor **(1.1.2.d)**. The airflow is then mixed with fuel and ignited in the combustion chamber **(1.1.2.e)**. The airflow then passes the turbine **(1.1.2.f)** which powers the fan and the compressor, and is then directed to the exhaust **(1.1.2.g)** where the velocity of the airflow is further increased.

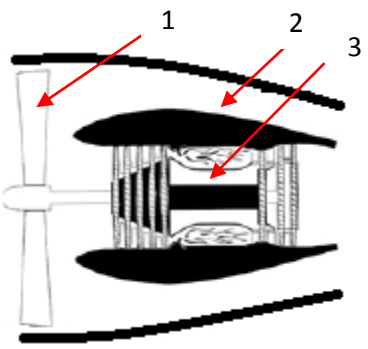
1.1.2.a Inlet

The first stage of the gas turbine is the inlet. Because the inlet is divergent, the velocity of the incoming airflow is decreased before hitting the fan and compressor **(Equation 1-5)**. As a result of the decelerating airflow, kinetic energy is converted into potential energy. Because of this, air pressure and temperature rise. When air temperature rises, the speed of sound rises. For this reason the fan can rotate at higher speeds before developing shock waves which occur when the speed of sound is exceeded. The rise in pressure within the inlet also provides the first step in compressing the airflow.

Equation 1-5		
Continuity Equation		
$$v_1 * A_1 = v_2 * A_2$$		
Symbol	**Variable**	**Unit**
v_1	Inlet velocity	m/s
A_1	Inlet surface	m^2
v_2	Velocity at location 2	m/s
A_2	Surface of location 2	m^2

1.1.2.b Fan

The first stage of the compressor is called the fan **(Figure 1-1)**. The fan is a rotor blade, but has a larger diameter then the rest of the rotor blades found in the compressor. The other thing that apart the fan from the rest of the rotor blades is the dichotomy of the airflow after passing the fan. The air near the core of the fan is also passed to the core of the engine, of which the first step is the compressor. The air more near the tip of the fan is passed to the bypass section.



1. Fan
2. Bypass
3. Core

Figure 1-1 – The fan

1.1.2.c Bypass section

The relation between the airflow passing through the core and the airflow passing through the bypass section is called the bypass ratio **(Equation 1-6)**. The air that passes through the bypass section does not meet the same stages as the air that passes through the core section. In the core section the air is being compressed, mixed with fuel, ignited and expanded. In the bypass section the airflow is only accelerated by a diverging passage, where after it is mixed with the core stream in the exhaust. The acceleration of air in the bypass contributes for a big part of the thrust of the engine, especially at lower speeds and lower altitudes.

Equation 1-6		
Bypass Ratio		
$$BPR = \frac{\dot{m}_b}{\dot{m}_c}$$		
Symbol	**Variable**	**Unit**
BPR	Bypass Ratio	
$\dot{m}_b$	Mass flow through bypass section	Kg/s
$\dot{m}_c$	Velocity at location 2	Kg/s

1.1.2.d Compressor

To compress the airflow before reaching the combustion chamber, a compressor is used. The RR TAY uses an axial compressor, meaning there is an axial airflow through the compressor parallel compared to the core of the engine **(Appendix I)**. The compressor consists out of a number of stages, and each stage consists out of a rotor and a stator **(Figure 1-2)**. The rotors are rotating parts, which are driven by the turbine. The stators are stationary parts of the compressor, consisting of vanes which are mounted to the casing. As a result of the rotation of the rotors, energy is passed to the airflow, leading to an acceleration of the airflow. Because of the divergent passage between the rotor blades, the air is also compressed before being passed to the stators. Like the rotors, the stators have a divergent passage. In the stators however, no energy is added to the airflow, but the kinetic energy of the airflow given by the rotor blades is transferred into potential energy, resulting into a rise of air pressure. Repeating this cycle results in an overall increase of pressure and temperature through the compressor **(Figure 1-3)**.

1. Stator vanes
2. Rotor blades

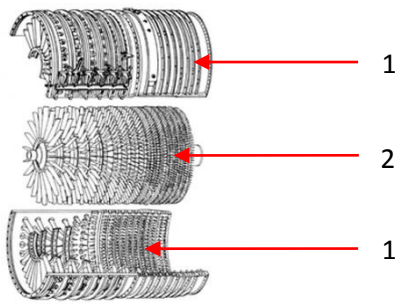
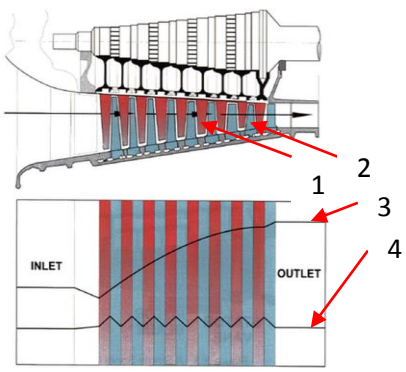


Figure 1-2 - Stator vanes and rotor blades



1. Stator vanes
2. Rotor blades
3. Air pressure
4. Air velocity

Figure 1-3 - Pressure and velocity

The rotors give the airflow a resulting direction corresponding to the direction in which the vanes of the stator blades are faces, and vice-versa **(Appendix I)**. This however, is an ideal situation. In reality, changes in airflow velocity or changes in rotor rotation affect the direction of the resultant airflow, and therefore the equality of direction of the airflow compared to the vanes. When this occurs, the angle of attack of the airflow compared to the blades can either be too high or too low. When the angle of attack is too high, a separation can occur which causes the blades to stall. When the angle of attack is too low, the blades push the airflow backwards which disturbs the compression process severely. For this reason bleeding valves are installed in the compressor of the RR TAY 620-15. The bleeding valves can dump air out of the compressor into the bypass section so that the airflow is accelerated. Since the bleeding valves can be opened and closed in a variable manner, they can affect the airflow in such a way so that the direction of the airflow again corresponds with the direction of the vanes. However, the air being dumped is already compressed. Energy has been used to accomplish this compression, and dumping the air is a waste of energy. For this reason, variable stator vanes have been installed to get the airflow aligned with the rotors and stators with relatively low disparities **(Figure 1-4)**. Variable stator vanes are movable stator blades which can be moved in such a manner so that its vanes face the same direction as the incoming airflow. This way, less energy has to be wasted to prevent a compressor stall or choke. The input for the position of the blades is given by the FADEC.

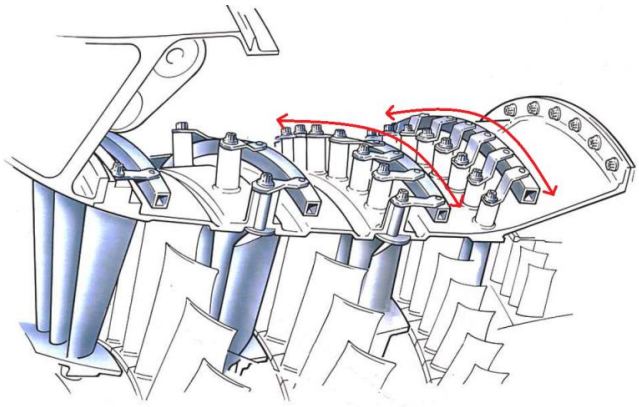
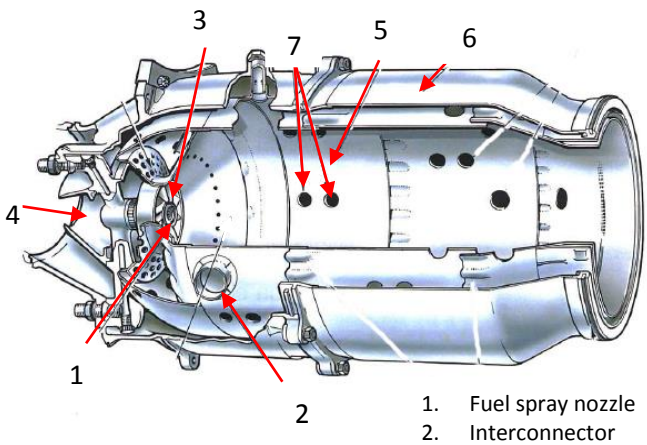


Figure 1-4 - Variable Stator Blades

1.1.2.e Combustion Chamber

After the airflow has been compressed, it will enter the combustion chamber **(Figure 1-5)**. The task of the combustion chamber is to burn large quantities of fuel, supplied through the fuel spray nozzles **(1)** to convert the energy saved in the fuel into heat energy (Q), which will deliver the desired thrust. The heated air has to be released in a manner that the air is expanded and accelerated smoothly and uniformly. The combustion section of the RR TAY 620-15 consists out of ten liners and are all supplied with a fuel spray nozzle. These liners are installed in an annular casing and are all interconnected **(2)**. Every engine has two igniter plugs, where an electric spark initiates the combustion. Because of the high entrance velocity of the air, the first thing that it has to do is to decelerate the air to a velocity capable of combustion; this is done by swirl vanes **(3)**. Beside the deceleration of the air flow, the swirl vanes are also causing turbulent air for the mixing of fuel and air. Approximately 20% of the air flow is taken in by the snout **(4)**. The other 80% of air, which is not burned, flows into the annular space between the flame tube **(5)** and the air casing **(6)** and will cool



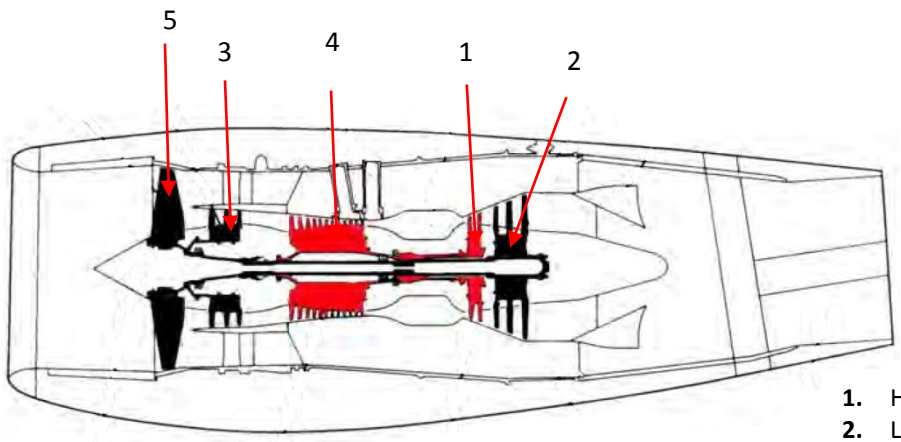
1. Fuel spray nozzle
2. Interconnector
3. Swirl vane
4. Snout
5. Flame tube
6. Air casing
7. Secondary Air

Figure 1-5 - Combustion Chamber

the gas temperature before it will enter the turbine. The secondary air holes **(7)** have the function to create a low velocity recirculation in the flame tube.
The combustion process is an isobaric process, this means that during the combustion the air keeps the same pressure. During this isobaric process, the temperature of gas is ranging from 1800C° to 2000C°. The inside casing of the combustion chamber is therefore constructed with alloys which can cope against high temperatures. In the case of the RR TAY 620-15, chrome and nickel alloys are used.

1.1.2.f Turbine

After the burning in the combustion chamber, the gas will now reach the turbine. The turbine has the task of providing the power to drive the compressor and accessories. By extracting energy out of the hot gases released from the combustion chamber, it has the power to drive these components. The RR TAY 620-15 engine has two turbines **(Figure 1-6)**: the high pressure turbine (HPT) **(1)** and low pressure turbine (LPT) **(2)**. The high pressure turbine, which consists out of 2 stages, drives the high pressure compressor (HPC) **(3)** and unit's necessary for the operation of engine and aircraft systems. Their rotation speed is called N1. The low pressure turbine, which consists out of 3 stages, drives the low pressure compressor (LPC) **(4)** and fan **(5)**. Their rotation speed is called N2. The LP shaft passes through the HP shaft. Each shaft rotates independently of the other and in a clockwise direction, viewed from the front of the engine.



1. High Pressure Turbine
2. Low Pressure Turbine
3. Low Pressure Compressor
4. High Pressure Compressor
5. Fan

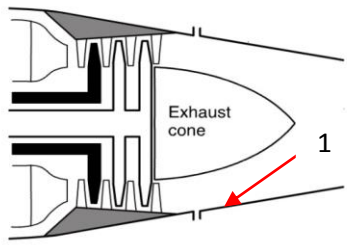
Figure 1-6 - High Pressure and Low Pressure Turbine

The turbine depends for its operation on the transfer of energy between the combustion and the turbine. This energy is taken from the expansion of the hot gases. The temperature just before the high pressure turbine will be maximal **(Appendix II),** also known as the Turbine Inlet Temperature (TIT). As long as there is combustion, the turbine will keep spinning. This spinning is caused by the high velocity blowing gases. The whole process in the turbine is isentropic. Because the gas is expanding, pressure and temperature will decrease and volume will increase. To handle this increased volume of gas, the turbine is made divergent.

The turbine consists out of a number of stages. Every stage is a combination between a rotating rotor blade and a static stator vane. These blades are made out of alloys that are capable of handling high temperatures. This is done so that the Turbine Inlet Temperature can be increased, which increases the engine's thrust.

1.1.2.g Exhaust

The last part of the engine is the exhaust. The purpose of an engine's exhaust is to increase the thrust produced by an engine by accelerating the exhaust gases to a higher speed. This is done by the convergent part of the exhaust, which is formed between the exhaust cone and exhaust pipe **(Figure 1-7) (1)**. The RR TAY 620-15 has a bypass and that means that this engine has two sorts of exhausts: one exhaust for the hot gases and one for the cold airflow from the bypass. These two gases and air flows are then mixed **(2)**. The mixer reduces the velocity of the outgoing air, which results in a reduction of the amount of noise produced. Like the turbine, the exhaust must also be capable of withstanding the high gas temperatures and is therefore manufactured from nickel or titanium.



1. Converging exhaust
2. Mixing of hot gas and cold air

1.
2.
3.
4.
5.
6.
7.

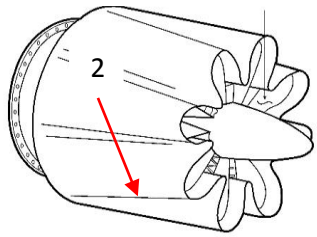


Figure 1-7 - Exhaust

1.1.3 Systems

There are multiple sub systems in the engine. There is the gearbox **(1.1.3.a)** which powers pumps and control systems. The engine can be started by the starting system **(1.1.3.b)** or by the ignition system **(1.1.3.c).** The engine has to be protected from ice and fire. Therefore the aircraft is equipped with a anti-icing system **(1.1.3.d)** and fire protection system **(1.1.3.e).**

1.1.3.a Gearbox

An aircraft has systems, like hydraulic, electric and pneumatic systems that need power. The accessory units deliver this power **(Figure 1-8)**. The accessory units do not only reinforce the pumps and the control systems, it also lets the engine work efficiently. The accessory units are built in gearboxes. The gearboxes regulates that the drive is equally separated over the accessory units. The internal gearbox **(1)** is mounted in the core of the engine, in the compressor. The internal gearbox is positioned in the core, because to create a minimum of thermic fatigue and to minimize the decrease of thrust. Detected on the internal gearbox a radial driveshaft **(2)** is placed. The function of the radial driveshaft is to get the drive or rotation from the internal gearbox to the external gearbox. It is possible that the radial driveshaft cannot be directly connected to the external gearbox, so an intermediate gearbox **(3)** is made between the gearboxes. The intermediate gearbox is mounted on the high-pressure compressor and redirects the drive. The last component of the gearbox is the external gearbox **(4)**. The external gearbox can hand turn the engine; this is made for the maintenance. It also works with a starter/driven gear shaft that makes sure that the drive is divided in two sections: one for low power accessories and one for high power pressure accessories. Because of the combination between small and big gears, in the external gearbox, the gearbox has low weight. In some case an accessory unit can fail, so the external gearbox is geared with a safety system. This system makes sure if one accessory unit fails, the external gearbox fails. This protects the gearbox for damage.

1. Internal Gearbox
2. Radial driveshaft
3. Intermediate Gearbox
4. External

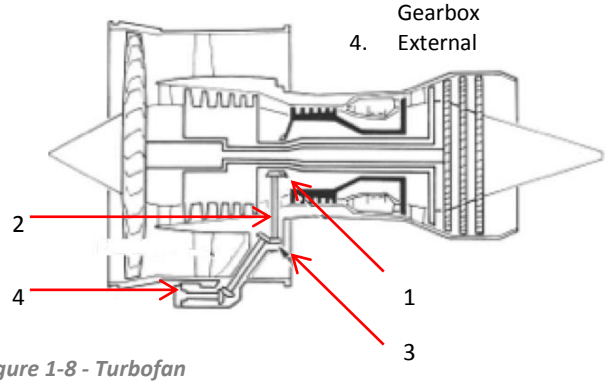


Figure 1-8 - Turbofan

1.1.3.b Starting

The air starting system **(Figure 1-9)** ensures that the engine of the RR-TAY 620 starts. This starting system is used in most of the commercial aircraft, because the system is light, simple and economical. As the engine wants to start, the air system will work. Air will flow through the inlet **(1)**, stator to the turbine rotor **(2)**. The air, originating from extern supply of the ground, will make the turbine rotate. The extern supply of air is regulated by the APU or cross-feed. The turbine rotates faster and faster. At a certain point the speed is fast enough and the sensors give a signal to close the air outlet **(3)**, which are electronically maintained. The outlet close at a speed that is predetermined. The air starter motor transmits the power, which is created, to the reduction gear **(4)** and the clutch **(5)**. These parts are connected with the engine. When the cycle is completed the engine will start.

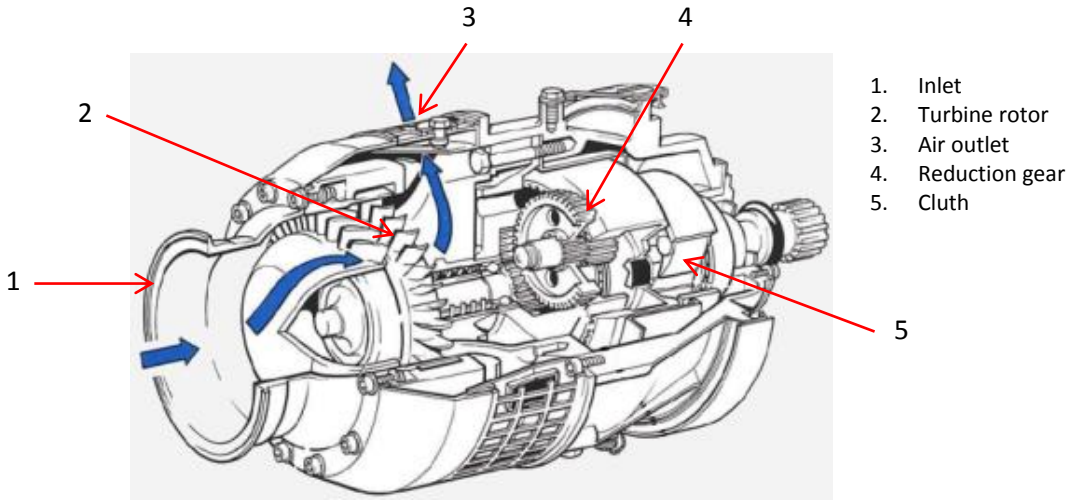


Figure 1-9 - Air starting system

1.1.3.c Ignition

It is not always possible to get enough extern air supply to start the engine. The combustion system is the alternative to start the engine. The system is positioned in the engine and works practically the same as the air starting system. The starter has a little combustion chamber. In this chamber the pressure is increased. Now fuel from the aircraft is added with the air supply in the chamber. The valves provide the air supply. The air/fuel mixture can be ignition. The engine got the supply and can start the engine.

1.1.3.d Anti-Ice

During flying or waiting on the airport, ice can form on the front edges of the engine. This is a problem for the engine, because the ice can change the aerodynamics in the engine and it can damage the engine. It is dangerous and has to be solved, for this reason there is an anti-icing system. The anti-icing system ensures that surfaces, where ice is created quickly, is heated. The surfaces are heated by the hot air of the compressor stages, by using valves. The valves are controlled electrically.

1.1.3.e Fire protection

Due to the high temperatures in the combustion chamber the engine is a potential fire risk. For this reason it is important that in this case the fire is traced and that the fire does not spread. The fire protection system solves this problem. This system is equipped with a detection and an extinguishing system. Several components of the engine are made from fireproof material and have the detection and extinguishing system:

- Housing
- Tubes
- Hoods

Ad 1. **Housing**

The flammable liquids are place in a housing in the compressor. This area is the cool area. The housing is made of fireproof materials. If the case is that the flammable liquid are placed in the hot section, than there is a double casing.

Ad 2. **Tubes**

All tubes in with fuel, hydraulic fluid or oil is transported, have to be made of fire resistance material. This material must comply with certain fire regulations.

Ad 3. **Hoods**

The external hoods are equipped with an adequate drainage system to remove the flammable liquids of the nacelle, bay or pod.

1.1.4 Calculation

There is a test calculation made for the thrust and TSFC at cruising altitude from the RR TAY engine to validate the calculation with the future designed engine. The excel sheet for this calculation can be found in **Appendix III**. For this calculation a height of 30.000ft is used as the assignment describes. The temperature, pressure, density and true airspeed can be calculated at this height. The other information was given in the assignment book. The efficiencies were not given and cannot be calculated. The efficiencies of the turbine, compressor and fan were estimated to be quite low. This is because these components of the engine have developed enormously over the past years and these would not be very high during the time this engine was developed. With this information the thrust and TSFC can be calculated by using the formulas described in **Appendix IV**. The thrust is 12442N and this is almost exact the correct thrust of the RR-TAY, which is 12455N. The TSFC is $2{,}5837 \cdot 10^{-5}$Kg/(N·s) and the correct TSFC is $1{,}95446 \cdot 10^{-5}$Kg/(N·s). This difference is caused by a pressure component because the engine is choked. This component will not be taken into account in the calculation of the future designed engine.

1.2 Fokker 70 analysis

The Fokker 70 analysis will give an inside look at the systems of the Fokker 70. These available systems **(1.2.1)** are to operate the turbofan engine. The engine mounting **(1.2.2)** will explain how the turbofan engine is mounted to the fuselage of the Fokker 70.

1.2.1 Available systems

The Fokker 70 has several systems that operate the turbofans. Without these systems the engine cannot operate. The engine control system **(1.2.1.a)** controls the engine and can be controlled automatically or manually. The engine is cooled and lubricated by the oil system **(1.2.1.b).** The fuel system **(1.2.1.c)** supplies fuel flow to the engine. All bleed air in the aircraft is powered by the bleed air system **(1.2.1.d)**.

1.2.1.a Engine control system

All the actions the crew can take related to the engines are located at the engine control panel. This panel is located at the middle section of the cockpit. The cockpit crew can use several subsystems to control and monitor the engines:

- Thrust levers
- Multi-function display system
- Standby Engine Indicator
- FADEC

Ad 1. **Thrust levers**

Two thrust levers are located in the middle of the cockpit to control the thrust of the engines. Each engine can be controlled separately. The crew can move the levers from idle to full throttle manually or use the auto throttle system. The levers give a mechanical movement to the servomotors, which are connected to the fuel flow, which goes to the engine. Under the left and right engine thrust levers an angle scale with degrees is located. The range of this scale is from 0° to 50°. Around 22° a marker for minimum take-off thrust is located when this amount of thrust is selected for take-off, a good cabin pressure, air conditioning, auto shut-off and stall protection is guaranteed.

Ad 2. **Multi-function display system**

The Fokker 70 has two Multi-function display systems (MFDS). The two displays give information about the performance, thrust ratings, air temperatures and primary parameters of the engines. The left display gives information about the selected thrust rating, the N1 and N2 percentage, actual Engine Power Rating (EPR) and the Turbine Gas Temperature (TGT). The right display gives information about the oil pressure and limits, oil temperature and limits, oil quantity and limits, fuel flow, fuel use, fuel temperature and the N1 and N2 vibration numerals.

Ad 3. **Standby Engine Indicator**

The Fokker 70 is supplied with a Standby Engine Indicator (SEI). In case of a failure of the MFDS the crew switches the SEI on, so the primary parameters of both engines like the EPR, the TGT are indicated. A parameter will flash if the maximum or minimum operating limit is exceeded to warn the crew.

Ad 4. **FADEC**

Full Authority Digital Engine Control (FADEC) is a digital engine control system. The system monitors and regulates the engines. This is done by an Electric Control Unit (ECU) that provides data of the engine. The ECU is connected to sensors that measure temperature and pressure of different stages of the engine. In the cockpit the crew can give input as usual, but the signals to the operating system are transported with FADEC. It uses parameters from the cockpit and the engine to check the inputs from the crew for crossing limitations, and then transport the signals to the operating system. The main advantages of the FADEC system are better engine efficiency, redundancy in case of a failure and FADEC can automatic response in case of emergency. The FADEC system is currently installed on the Fokker 70, but the RR TAY 620-15 engine does not have a FADEC connection. So the Fokker 70 has the FADEC system, but cannot use it with the current installed engine.

1.2.1.b Oil systems

The oil system has the purpose to lubricate and cool components of the engine and is pressurized by the oil pumps. The gear pump consists of two steel gears that counter turns in a closed case. The fitting must be tight because only the gear separates the low-pressure oil from the high-pressure oil. When the gears counter turn, low-pressure oil traps between the gears and the case. While turning, high-pressure oil comes out. If the oil temperature is too high, the high-pressure oil is cooled, temperature checked and filtered before it reaches the bearings and the gearbox. In the bearings and gearbox air and dirt is mixed with the oil. The oil is then pumped out of the engine bearings and gearboxes by five scavenge pumps. The next step is to filter the dirt out of the oil. A chip detector checks the oil for any more impurities. When all the dirt is removed from the oil, the de-aerator sets to work. The de-aerator filters the residual air out of the oil. Then the oil reaches the oil tank, which is mounted on the engine. The oil is checked for pressure, quantity and temperature. Oil that is already cold, and means it is also thickened, can pass the cooling system by a by-pass valve. Heated oil can also be cooled by primary air-cooled oil cooler from the cold airflow. In this cooler the oil is pressed through a series of pipes with steel plates attached to it. When air passes through this system it takes away the heat of the steel plates that is formed by the heat of the oil. A secondary cooler is installed to cool it even more. This secondary cooler is called fuel-cooled oil cooler.

1.2.1.c Fuel system

The engine fuel system controls the fuel transfer automatically and provides a rapid and surge-free acceleration or deceleration. Input of the thrust lever position, high-pressure compressor inlet and outlet temperatures will be used by the fuel control system to maintain amount of fuel that will be transferred. The fuel from the aircraft fuel system is transported to the engine fuel system. There the fuel will pass an engine driven low-pressure fuel pump, an oil cooler, a filter and fuel flow transmitter and an engine driven high-pressure fuel pump. Fuel from the high-pressure fuel pump goes to a fuel flow regulator, which measures the right amount of fuel to the spray nozzles. The thrust levers in the cockpit control the fuel flow regulator, which provides a regular fuel flow.

1.2.1.d Bleed air system

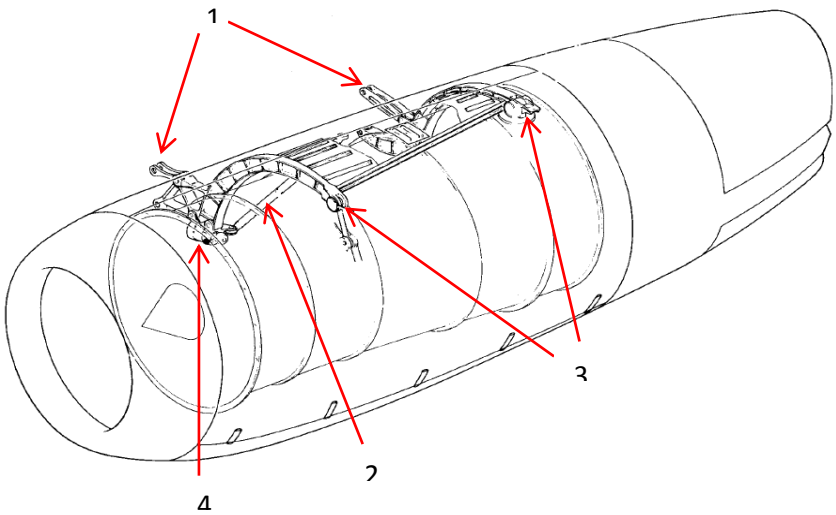
The bleed air system is used for air conditioning, cabin pressurization, engine starting, cooling. Bleed air can be supplied by the engine, when running, or on the ground by the APU. The bleed air system uses air from the compressor stages. Low-pressure (LP) bleed air is extracted from the seventh high-pressure (HP) stage and the HP bleed air is extracted from the twelfth HP stage. During take-off only the bleed air from the seventh HP stage may be used. During normal operation the bleed air from the seventh HP compressor stage will be used. The bleed air of the twelfth HP compressor stage will be used when the bleed pressure of the LP compressor stage is too low or when the temperature is too low.

1.2.2 Engine Mounting

Chances are that the conceptual design engine has to produce more thrust than the RR TAY 620-15. This can only be achieved if the installed engine mounting construction of the F70 can absorb this extra thrust. The F70's engine mounting is constructed out of four key components to absorb forces and vibrations produced by the engine **(1.2.2.a)**. To determine the constructional thrust limit for the conceptual design the F100 engine mounting is examined. Because the engine mounting installed on the F70 and the F100 are identical, the constructional thrust limit can be determined **(1.2.2.b)**.

1.2.2.a Technical Description F70 engine mounting

Two steel engine/nacelle crane beams **(Figure 1-10) (1)** are rigidly mounted to the main structure of the aircraft and curve in an arc around the upper inboard quadrant of the engine. The greater part of the thrust forces are carried by the thrust strut **(2)** between the forward engine mounting and the main structure of the aircraft. Two sets of vibration isolators **(3)** and a trunnion with integrated isolators **(4)** are fitted to the crane beam engine mountings. This provides a flexible engine mounting to minimize the transmission of vibration to the aircraft structure and the passenger compartment.



1. Engine/nacelle crane beams
2. Thrust strut
3. Vibration isolators
4. Trunnion with integrated isolators

Figure 1-10 - F70 engine mounting

1.2.2.b Maximum construction Thrust

The F70 has a less powerful engine than the engine mounted on the F100. The F100 can be provided with both the TAY 620 as the TAY 650 engine. The TAY 650 can produce at maximum a thrust of 67kN **(Table 1-1)**. And since the engine mountings installed on the F70 and the F100 are completely identical the conceptual engine can produce 5kN more thrust. If more thrust per engine is needed, modifications on the construction of the engine mounting are required.

Table 1-1 - Engine Specifications

Engine	Tay 620	Tay 650
Thrust	62kN	67kN
Bypass ratio	3.04	3.07
Inlet Mass flow	186kg/s	189.6kg/s-193.2kg/s
Fan diameter	1,118m	1,143m
Length	2,405m	2,408m
Stages Fan	Fan, 3 LPC, 12 HPC, 2 HPT, 3 LPT	
Applications	Fokker 70/100	Fokker 100

1.3 Demands

To make a concept design engine for the Fokker 70 there are a few demands from the client **(1.3.1)**, which the new engine must meet. The concept design engine must also meet demands from the project group **(1.3.2)**.

1.3.1 Client Demands

The new gas turbine engine for the Fokker 70 must be upgraded to an engine which can perform conform the standards of today. The client, Amsterdam Leeuwenburg Aero Engines (ALAE), has made a few demands for the new engine:

- Gross second segment climb
- Minimum Range
- Efficiency fuel consumption

Ad 1. **Gross second segment climb**

The thrust is not a demand but a parameter, the new engine should provide the Fokker 70 with enough thrust to guarantee a 2,4% gross second segment climb at Maximum Take Off Weight (MTOW) and at an airport elevation of 5000ft under ISA conditions.

Ad 2. **Minimum Range**

The new engine must give the aircraft a minimum range to be at least 1200NM at Maximum Zero Fuel Weight (MZFW) under ISA conditions with zero wind without centre fuel tank.

Ad 3. **Efficiency fuel consumption**

Because the minimum range must make without centre fuel tank, the new engine must be fuel efficient. The thrust specific fuel consumption (TSFC) must be lower to reach the operating range of 1200NM.

1.3.2 Concept Engine demands

To design a new concept engine in such short time, demands are made so the concept design will not be too expensive and the concept engine will be ready on time. This demands are made by the project group who are working on the concept engine:

- Construction
- Noise control
- Emissions

Ad 1. **Construction**

The new engine must fit on the old Fokker 70 construction, so no modification will been made to the construction of the aircraft.

Ad 2. **Noise control**

There are no demands to improve the noise control they must only be within the limitations.

Ad 3. **Emissions**

Since it is still a concept design which will be developed, the amount of emissions concerning the requirements conform CS-E and CS-25 cannot yet be calculated. These will be calculated in a later stage.

1.3.3 Requirements for certification

The new engine must meet the applicable airworthiness requirements, the design and operational regulations. The demands for the engine design are set up by the *European Aviation Safety Association* (EASA). The requirements and regulations that are related to aircraft are written in the *EASA CS-25*, which contain the aircraft airworthiness requirements of the aircraft with the new engine.

For certification related to the aircraft engine design in general, they are divided in a *certification specification for engines (CS-E)*. Which states that all components of the engine must be constructed, arranged and installed so the installation must be accessible for maintenance and necessary inspections. The engine must be isolated from each other part from the aircraft so if the engine system fails or has a malfunction the remaining engine can still operate safely. To assure the designed engine its obtained performance will be approved by the Agency, EASA made performance formulas for *CS-E*. These formulas are provided in **Appendix VI**. The *EASA CS-34* has requirements about the emission requirements *(ICAO Annex 16 – Volume II)* and the Noise requirements are stated in *EASA CS-36 (ICAO Annex 16 – Volume I).* Where the new engine must meet the new noise requirements instead of the noise requirements on the current Fokker 70 engine.

Conform the *Supplemental Type Certificate (STC)* it is possible to have 10% more thrust than the value set by Fokker. This certificate defines the product design change, which explains how the changes affect the existing design and efficiency. This information is relevant for a new designed of modification of a new engine.

2 Performance Requirements

Before designing the engine, the performance requirements need to be calculated. There are two hard requirements given for the new engine: The engine should have an increased efficiency so the range will increase to 1200NM **(2.1)**, the second requirement is the 2,4% gross second segment climb **(2.2)**. Reducing the TSFC will increase the range, and to make sure that the aircraft could climb with the given slope the minimal thrust is needed to be calculated.

2.1 Efficiency calculation

The conceptual engine has to be far more efficient then the currently installed RR TAY 620-15. The conceptual engine should give the F70 without centre fuel tank an operational range of at least 1200NM at MZFW under ISA conditions with zero wind, long range cruise, EU-OPS 1.255 reserves and a 100NM alternate. Under these conditions the F70 with the RR TAY 620-15 installed has an operating range around 550NM following the payload vs. range graph provided by Fokker **(2.1.1)**. This means the conceptual engine has to be more than twice as efficient as the RR TAY 620-15 **(2.1.2)**. The KLM F70-F100 Aircraft Operating Manual (AOM) is used in order to obtain operational values, which are needed for the implementation of the different equations. Operational values like the MTOW, MZFW needed amount of reserve fuel. These operational values are prepared by KLM with the help of their validated calculations methods.

2.1.1 Range Fokker 70 with Rolls Royce TAY 620-15

The Fokker 70 without centre fuel tank at MZFW has an operating range around 550NM **(Figure 2-1)**. The blue dotted line **(1)** is associated with the F70 without centre fuel tank. At maximum payload (MZFW) **(2)** the operational range is around 550NM **(3)**. The graph provided by Fokker is under ISA conditions, zero wind, Long Range Cruise, EU-OPS 1.255 Reserves and with a 100NM alternate.

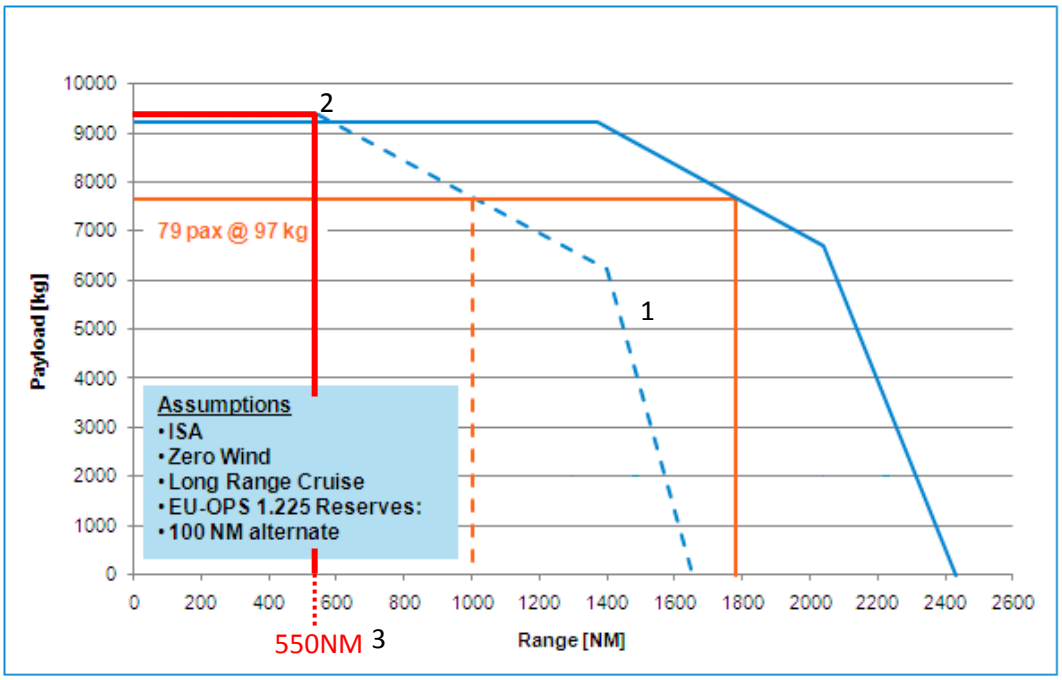


Figure 2-1 - Payload vs. Range

This operational range can be verified by the range equation of Louis-Charles Brequet for jet aircraft **(Equation 2-1)**. This equation can be used to calculate the range without taking the climb and the descent segments of the flight in to account. In other words a straight line from ´point 0´ to ´point 1´ at a constant flight level.

Equation 2-1		
Brequet's range formula		
$$R = \frac{2}{C_T} \cdot TAS_0 \cdot \frac{C_L}{C_D} \cdot \left(1 - \sqrt{\frac{W_1}{W_0}}\right)$$		
Symbol	**Variable**	**Unit**
C_T	Specific Fuel Consumption	$\left[\frac{N}{N \cdot S}\right]$
TAS_0	True Airspeed at the start of the cruise	m/s
$\frac{C_L}{C_D}$	Lift coefficient – drag coefficient ratio	-
W_1	Weight of the aircraft at the end of the cruise	kg
W_0	Weight of the aircraft at the start of the cruise	kg

The C_T at cruise level is calculated at $2{,}53098 \cdot 10^{-4}$ (N/N·S). The TAS_0 is calculated at 221,54 (m/s). The C_L/C_D is assumed to be 15. W_1 is calculated at 35.754,9 (kg). W_1 is the MZFW plus needed reserve fuel, following the *KLM AOM* the MZFW is 33.565 (kg) and the needed amount of reserve fuel is 2189,9 kg. W_0 is the MTOW, following the *KLM AOM* the MTOW is 37.995 (kg). This results in an operational range of 424NM. The exact calculation of the range of the F70 with the RR TAY 620-15 engines as well as the determination of the different variables used in Breguet's equation are provided in **Appendix VII**.

An operational range of 424NM is 126NM less than the operating range provided by Fokker (550NM). This can be explained by the fuel policy wielded by KLM. KLM wields a rather safe fuel policy, by adding extra reserve fuel to the mandatory EU-OPS reserve fuel. For example; KLM adds an extra company fuel for holding purposes of 300 kg **(Appendix VII)**. This 300kg extra fuel is not mandatory by EU-OPS 1.255. If all the extra fuel is withdrawn from the equation the operating range varies between the 500NM – 550NM.

2.1.2 Efficiency conceptual engine

Given the demand the conceptual engine should give the F70 without centre fuel tank an operational range of 1200NM, the minimal specific fuel consumption (SFC or C_T) in the range equation can be calculated. However in the range equation **(Equation 2-1)** there are two variables; the C_T itself and W_1. W_1 contains the constant MZFW (33.565 kg) and the variable reserve fuel value. The reserve fuel value is dependent on the C_T, because when the total efficiency (C_T) has to drop the reserve fuel consumption will drop as well. In other words if the operating range has to increase from around 550NM to 1200NM (a factor: 2,18) the fuel consumption has to decrease a factor 2,18. So does the reserve fuel consumption has to decrease a factor 2,18. This causes an infinite calculation loop. This loop can be solved by applying an advanced calculation method (**Appendix VIII)**. Using this method; results in a C_T of $1.3665 \cdot 10^{-4}$ (N/N·s) and a reserve fuel value of 1028,7 kg. W_1 is MZFW plus Reserves, results in: W_1 = 34.593,7 kg. This can be verified by inserting these numbers in Brequet's range equation **(Equation 2-2)**. Results in an operational range of 1203NM.

Equation 2-2		
Verification range conceptual engine		
$$1203Nm = \frac{2}{1.3665 \cdot 10^{-4}} \cdot 221,54 \cdot 15 \cdot \left(1 - \sqrt{\frac{34.593,7}{37.995}}\right)$$		
Symbol	**Variable**	**Unit**
-	-	-

Concluding, the conceptual engine has to achieve a C_T (SFC) of $1.3665 \cdot 10^{-4}$, which results in a TSFC of $1.3930 \cdot 10^{-5}$ this to provide the F70 with the conceptual engine an operating range of 1203NM. This with a reserve fuel of 1028,7 kg for a 100NM alternate under EU-OPS 1.255, without any extra fuel added, under ISA conditions, zero wind and Long Range Cruise performance.

2.2 Thrust

EASA has a requirement that each aircraft can perform a 2.4% gross second segment climb *(CS 25.121)*. The engines should deliver enough thrust that the aircraft can complete this climb with only one engine running. To start with the calculations it is useful to understand which forces are working at the aircraft during the climb and under which angle they are placed. Therefore there is made a situation sketch **(2.2.1)**. When understanding the situation sketch, the calculations **(2.2.2)** can be made.

2.2.1 Situation sketch

The 2,4% (1.3748 degrees) second segment climb will mean that an aircraft climbs 2,4 meters vertical while traveling 100m in horizontal direction. This climb should be completed with MTOW at an airport elevation of 5000ft with only one engine. During the take-off the pitch of the aircraft will be 12 degrees **(Figure 2-2)**.

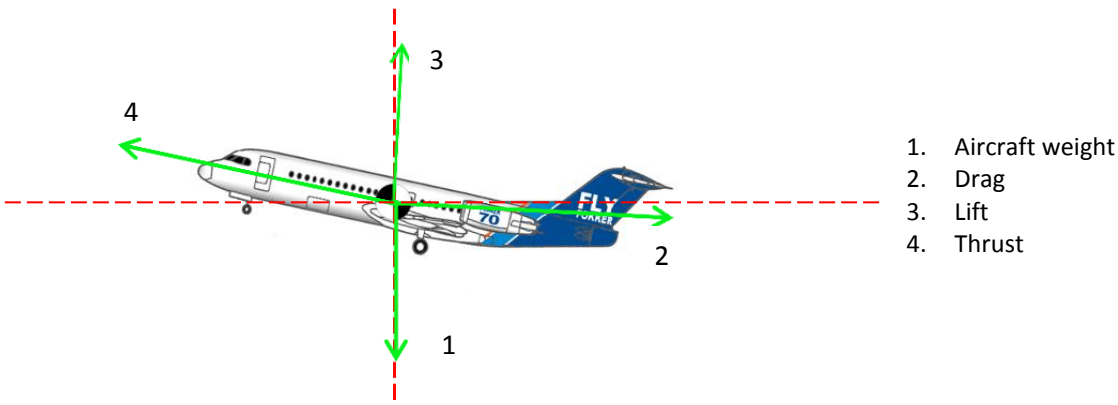


Figure 2-2 - Situation sketch

The weight **(1)** of the aircraft is perpendicular downwards. The drag **(2)** of the aircraft is under an angle of 1.3748 degrees because this is the climb path of the aircraft. Therefore also the lift **(3)** is tilted 1.3748 degrees backwards. The angle of the thrust **(4)** is 12 degrees.

2.2.2 Minimum thrust calculations

The first step is to calculate the C_{D0}, because C_{D0} is assumed to be constant, C_{D0} can be calculated at cruising altitude. To calculate $C_{D0,}$ **Equation 2-3** needs to be filled in. This formula is a substitution of the drag (**Equation 2-4)**, the Lift formula, another form of the drag formula and the formula of the aspect ratio (AR). In **Appendix IX** the substitution and calculations of the formulas will be explained.

Equation 2-3
Drag (D) and C_{D0} calculations

$$D = C_{D0} \cdot 0.5 \cdot \rho \cdot v^2 * S + \frac{{C_L}^2}{\pi \cdot \frac{b^2}{S} \cdot e} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S = \frac{C_D}{C_L} \cdot W$$

$$C_{D0} = \frac{\frac{C_D}{C_L} \cdot W}{0.5 \cdot \rho \cdot v^2 \cdot S + \frac{\left(\frac{W}{0.5 \cdot \rho \cdot v^2 \cdot S}\right)^2}{\pi \cdot \frac{b^2}{S} \cdot e}} = 0.0234349$$

Now C_{D0} is known (**0.0234349**) the necessary thrust can be calculated. Therefore the thrust **(Equation 2-5)** is needed. When the thrust is calculated there is a problem. This calculation does not esteem the vertical component of the thrust due to the twelve degrees angle of attack **(Figure 2-2)**. The result of the vertical component of the thrust is relative simple, the amount of lift needed decreases. As result of the decreasing lift, the C_L coefficient will also decrease. When C_L decreases, the drag and therefore the amount of thrust needed decrease what will result in a lower vertical thrust component, and go further. This loop needs to be calculated a few times. So a new CL formula **(Appendix IX)** with vertical thrust component is needed. This C_L value will be filled in, in the thrust formula. After recalculating this loop for a couple of times the drag will be almost constant.

Equation 2-4
Thrust formula

$$T = C_{D0} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S + \frac{{C_L}^2}{\pi \cdot \frac{b^2}{S} \cdot e} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S + W \cdot \sin(\gamma)$$

Equation 2-5
C_L calculations with vertical thrust component

$$C_L = \frac{\left(W - \frac{W \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (W \cdot \sin 1.3748)\right) \cdot 0.5 \cdot \rho \cdot v^2 \cdot S}{\cos 1.3748}$$

In **Table 2-1**, the C_L and the minimum thrust calculated on basis of the explained equations.

Table 2-1 - C_L and minimum thrust calculations		
Calculation method	**C_L coefficient**	**Minimum thrust (N)**
First calculation (without v. thrust component)	1,614806	43119,72
First Loop	1,536269	40389,67
Second Loop	1,541242	40558,48
Third Loop	1,540934	40548,02
Fourth Loop	1,540953	40548,67
Fifth Loop	1,540952	40548,63
Sixth Loop	1,540952	40548,63

In the fifth and sixth loop the C_L coefficient and the minimum thrust are constant. The minimum amount of thrust that is necessary to take off from an airport with an elevation of 5000ft with maximum take-off thrust is **40549 N**. When filling in the ISA standard values from 5000ft in the Excel-model, the RR Tay 620 generates **50735 N**. So even the current RR Tay is powerful enough to climb with a 2.4% second gradient from an airfield with an elevation of 5000ft under ISA circumstances.

Table 2-2	Corresponds to Equation 2-3	
Symbol	**Variable**	**Unit**
C_{D0}	Zero lift drag constant	-
$\frac{C_D}{C_L}$	Drag/lift ratio	-
W	Aircraft weight	N
ρ	Air density	Kg/m^3
v	True air speed	m/s
b	Wing width	m
S	Wing surface area	93.5 m^2

Table 2-3	Corresponds to Equation 2-4	
Symbol	**Variable**	**Unit**
T	Thrust	N
C_{D0}	Zero lift drag coefficient	-
ρ	Air density	Kg/m^3
v	True air speed	m/s
S	Wing surface area	m^2
C_L	Lift coefficient	-
b	Wing span	m
e	Oswald factor	-
W	Aircraft weight	N
γ	Flight path angle	Degrees

Table 2-4	Corresponds to Equation 2-5	
Symbol	**Variable**	**Unit**
C_L	Lift coefficient	-
W	Aircraft weight	N
ρ	Air density	Kg/m^3
v	True air speed	m/s
S	Wing surface area	m^2

3 Engine Design

Nowadays, new techniques have been developed to make the engines more efficient. These modern improvements **(3.1)** can be made on different parts of the engine. These modern improvements will be taken into account when designing **(3.2)** the new engine.

3.1 Modern Improvements

Since the first run of the RR TAY 620-15 in 1984, many improvements have been made. All this improvements contribute in increasing the thrust and efficiency and decreasing the noise and emissions of the engine. Improvements concerning engine control systems optimise the efficiency **(3.1.1)**. Emissions can be reduced by modern combustion systems **(3.1.2)**. Modern day fan blades can improve the efficiency and the thrust **(3.1.3)**. As well as other thermodynamic and aerodynamic improvements **(3.1.4)**. With the help of new materials higher engine temperatures can be achieved, also the weight of the engine can be reduced with lightweight composite materials **(3.1.5)**.

3.1.1 Engine Control

Modern day jet engines are controlled by fully authorised digital control systems, better known as: Full Authority Digital Engine Control (FADEC). The RR TAY 620-15 is developed in the early eighties and the first engine run was conducted in 1984. At that time, digital engine control systems were not as advanced as today's systems. The most commonly used system nowadays is the FADEC system. So if the decision is made to install a digital engine control system it will be highly recommended to install the FADEC system. In order to make this decision the FADEC system has to be analysed. At first a simple technical description is needed in order to determine if the FADEC system can be included in the conceptual design **(3.1.1.a)**. Thereafter the advantages of the FADEC system will be established in order to determine if it is lucrative to install the FADEC system **(3.1.1.b)**.

3.1.1.a Technical description FADEC systems

Two units operate the FADEC system:

- Engine Control Unit
- Hydro mechanical Unit

Ad 1. **Electronic Control Unit**

The FADEC system is controlled by an Engine Control Unit (ECU), which is mounted on the engine fan case. This is a digital control system that performs complete engine management. Full authority means that this system is placing full authority over the sensors and parameters of the engine in the hands of the computer. If FADEC fails completely, the complete engine will fail. To reduce this potential risk, FADEC has two-channel redundancy, with one active channel and one in standby. If one channel fails, the other automatically takes control, which means that normal engine control and operation with a complete failure of one channel is maintained. Both channels are housed in one assembly, but are physically separated.

The ECU works with two input signals:

1) Pilot inputs: the pilot uses the thrust levers to set the desired thrust. The FADEC in this situation will prevent the thrust from exceeding the limit.
2) From parameters: multiple input variables of the current flight condition as air density, engine temperatures and pressures and many other parameters. These parameters are received from the Air Data Computer and from the Flight Control Computer.

The received inputs are analysed up to 70 times per second. Using these inputs, FADEC then computes the appropriate thrust settings and applies them. Besides this trust setting, FADEC is also

responsible for the control of fuel flow, protection against engine exceeding limits and engine starting.

Ad 2. **Hydro mechanical Unit**

The Hydro mechanical Unit (HMU) controls the fuel flow to the engine's combustion chamber, controls fuel hydraulic signals to actuators and protects against over speeding. The Fuel Metering Valve (FMV) in the HMU transforms FADEC orders through a torque motor and servo valve into fuel flow to the engine nozzles **(Appendix V).**

3.1.1.b Advantages

New engines are adopting FADEC for the benefits offered by digital control, improved reliability and performance and weight reduction. If the engine is equipped with FADEC, fuel efficiency is improved up to 15% due to faster and more accurate engine control. FADEC provides optimum engine efficiency for a given flight condition. Thereby the weight is reduced tremendously, because of the missing heavy mechanical assemblies. Another advantage of the FADAC is the possibility of programming the system. Fuel flow for example can be adjusted.

FADEC also allows the manufacturer to program engine limitations and receive health and maintenance reports. The ECU transmits maintenance data to the Central Fault Display Interface Unit (CFDIU). The interface of the ECU with the CFDIU together with the Multifunction Control Display Unit (MCDU) allows the maintenance crew to quickly access the ECU memory for faults in the engine.

3.1.2 Twin Annular Premixing Swirler (TAPS)

The Twin Annular Premixing Swirler is developed to reduce the NOx emissions. ICAO states NOx Emission Standards. Their environmental activities are largely undertaken through the Committee on Aviation Environmental Protection (CAEP), where new standards on aircraft noise and aircraft engine emissions are formulated. A TAPS II Combustor reduces the NOx emissions by 60% below CAEP/6 (the sixth meeting of CAEP, where new recommendations are made).

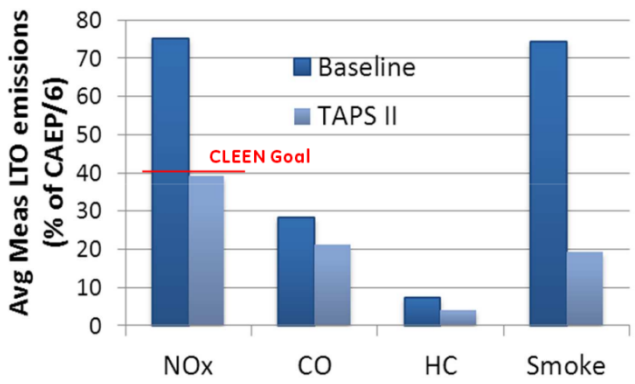


Figure 3-1 - Emission Comparison

In **Figure 3-1** there's shown which improvements in emissions can be made using the TAPS II combustor compared with earlier combustion techniques. Beside the large emission reduction, TAPS is also increasing the performance of the combustion chamber.

The TAPS system consists of the TAPS fuel air mixer and the computer controlled fuel system. The fuel air mixer creates two different flames in the combustor **(Figure 3-2).** The fuel system calculates the amount of fuel and air that needs to be burned and then precisely controls the distribution of fuel in the combustion zone. It provides a process hot enough for burning, but cool enough to prevent high amounts of NOx production.

As the fuel is burned in the combustion chamber, the temperature is thereby increased. If there is too much fuel for the available air, the air mixture is 'rich'. If there is less fuel than available air, the air mixture is 'lean'. In-between the rich and lean zone the maximum amount of NOx is produced **(Figure 3-3)**. Other than conventional combustion chambers, the TAPS II Combustion chamber uses more air at the fuel nozzle for combustion: 70% of combustion air compared to 25% of the combustion air from the conventional chambers. This means that the fuel to air mixture starts out in the 'lean' zone. Conventional chambers pass through the rich and lean zone, where the TAPS II combustor starts in the lean zone and is thereby always lean, producing less NOx.

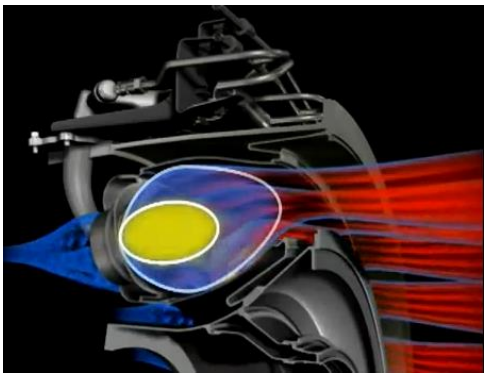


Figure 3-2 - TAPS system

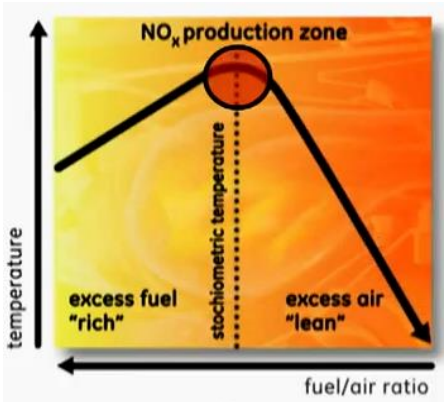


Figure 3-3 - Rich & Lean Zone

3.1.3 Twisted and swept fan blades

A design for improving the performance of a fan blade is to add a twist in the fan blade **(Figure 3-4)**. The speed at the root of the blade is much lower than the speed of the fan blade at the tip. When having a straight fan blade the acceleration of the air by the tip will be larger due to the higher speed of the tip. To give each part of the airflow the same amount acceleration and energy, the root of the blade is twisted much more to increase the acceleration of the air of the root to the same level as by the tip.

Another option to improve the performance of the fan is to give it sweepback **(Figure 3-5)**. The theory behind a swept fan blade is the same as by swept wings. The air will not flow straight over the chord, but due to the sweep or curl in the fan blade the effective speed of the airflow along the chord is v·cos(s) and thus lower than the airspeed of aircraft **(Figure 3-6)**, or in this case of the fan blade. Concluded: the fan is allowed to rotate faster without reaching M=1, therefore the fan operates more efficient and silent.

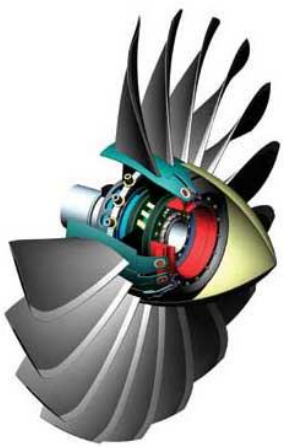


Figure 3-4 - Twisted fan blades

Figure 3-5 - Swept fan blades

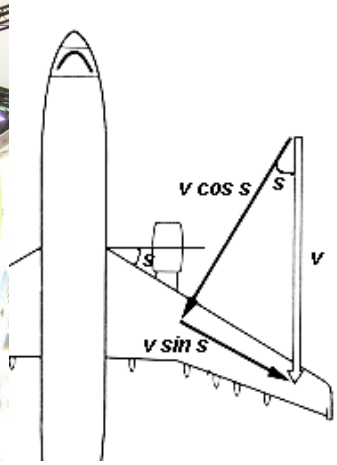


Figure 3-6 - Effective airflow speed

3.1.4 Thermodynamics and Aerodynamics

To improve the old parts of the engine and make the engine more efficient, some elements of the thermodynamic **(3.1.4.a)** have to be rectified. To improve the overall performance of the engine, some changes in the aerodynamics **(3.1.4.b)** need to be made.

3.1.4.a Thermodynamic

It is important to know which of the old parts of the engine needs to be improved. That is why different elements of the thermodynamic have to be rectified to make the engine more efficient. In some matter if one element improve the efficiency of the engine, while other elements efficiency can decrease. The elements to rectify the engine are:

- Specific work
- Thermic Efficiency
- Relations between CPR, TIT, F_{spec} and TSFC

Ad 1. **Specific work**

The specific work is the amount of work the engine produce with the help of one kilo air. During the creation of the engine the objective is to make maximize the specific work. This can be achieved by letting in a lot of air. An option is to make the gas turbine bigger. Another way to make the specific work higher with the pressure increase ratio in the compressor. The increase cannot be too high or too low, so it has to be in between. To calculate the maximum specific work has to be calculated. The equation for the maximum specific work is **(Equation 3-1)**.

Equation 3-1		
Specific Work		
$$W_{spec} = c_p \cdot (T_{04} + T_2 - \frac{T_{04} \cdot T_2}{T_{03}} - T_{03})$$		
Symbol	**Variable**	**Unit**
W_{spec}	Specific Work	J
c_p	Constant pressure	-
T_{04}	Turbine Inlet Temperature[1]	K
T_2	Temperature Compressor	K
T_{03}	Total temperature combustion chamber	K

Ad 2. **Thermic Efficiency**

The thermic efficiency is an important item in the engine. The thermic efficiency is the percentage of fuel converted in the acceleration of air. This efficiency has to be as high as possible, therefore less fuel is needed. Fuel is heavy, expansive, limited in the world and bad for the environment. The get the efficiency as high as possible is by a high increase of the pressure increase ratio. To equation for the thermic efficiency is **(Equation 3-2)**.

[1] TIT = Is the highest temperature outside the combustion chamber and is limited due the materials

Equation 3-2		
Thermic Efficiency		
$$\eta_{th} = 1 - \frac{1}{\left(\frac{p03}{p1}\right)^{\frac{\gamma-1}{\gamma}}}$$		
Symbol	**Variable**	**Unit**
η_{th}	Thermic Efficiency	J
P_{03}	Pressure combustion chamber	Pa
P_1	Begin pressure	Pa
γ	Heat capacity ratio	-

Ad 3. **Relations between CPR, TIT, Specific Thrust and the TSFC**

To know the engines, the relations between these elements are important. The first element is the CPR. The CPR is the pressure ratio in the compressor. The second unknown element is the TSFC. TSFC stands for Trust Specific Fuel Consumption. The TSFC indicates the quantity of fuel is needed to produce 1N trust, 1 hour long. The last element is the specific thrust. The specific thrust is the quantity thrust using 1kg air. In **Table 3-1** the relations between is shown.

Table 3-1 – TIT and CPR

Element	Change	Effect F_{spec}	Effect TSFC
TIT	↑	↑↑	↑
CPR	↑	↑↓	↓

In table is shown that when the TIT increases, the specific work decreases. This is advantageous for the thrust, but the TSFC is high and therefore more fuel is needed. When the CPR increases is first the specific thrust increase and means more thrust. But after an amount of time the specific thrust decrease and that is disadvantageous. While the CPR increase less fuel is needed, because of the TSFC. When creating a new engine, there have to be a consideration what is more important.

3.1.4.b Aerodynamic

To improve the overall performance of the new engine, a number of changes with regard to its aerodynamics will be made compared to the RR-TAY 620-15. The inlet and fan could be larger, the bypass ratio could be higher, and the new inlet could be equipped with a chevron exhaust, a scarf inlet, smoother compressor aerofoils and a blisk fan:

- Inlet and fan enlargement
- Higher bypass ratio
- Chevron exhaust
- Scarf Inlet
- Smoother compressor aerofoils
- Blisk

Ad 1. **Inlet and fan enlargement**

The inlet of the new engine, and the diameter of the fan could be larger than those of the RR-TAY 620-15. By enlarging the inlet, the airflow has more time to compress and heat up. Sound waves move more quickly through warmer air, so hot air means a higher sound barrier, which allows the fan to increase in size while maintaining its rotation speed. Because of this, more air can be sucked into the system. Therefore the thrust is increased while little extra fuel is inserted into the system.

Ad 2. **Higher bypass ratio**

Because of the larger fan, both the core section and the bypass section can be enlarged. However, in the new engine only the bypass section will increase in size. Because of the fact that the TIT is increased in the new engine, more potential energy can be converted by the turbines, allowing the larger fan to maintain operation without an increase in size of the turbine. As a result of the higher bypass ratio, the propulsion efficiency will increase. The extra cold air in the exhaust will also muffle the sound produced by the hot airflow coming form the core, resulting in a quieter engine.

Ad 3. **Chevron exhaust**

The new engine could make use of a chevron exhaust **(Figure 3-7)**. A chevron exhaust makes use of a saw-tooth pattern on the trailing edges of the exhaust, which smoothens the mixture of the high velocity core stream, and the low velocity bypass stream. This way less turbulence is developed, and therefore noise production is reduced.

Figure 3-7 - Chevron Exhaust

Ad 4. **Scarf inlet**

The new engine could be equipped with a scarf inlet **(Figure 3-8)**. A scarf inlet has a longer lower half of the inlet than the upper half. The longer lower half will form a geometric barrier for sound waves wanting to go downwards. This way less noise is heard on the ground during flight. The barrier also prevents debris from being sucked into the engine.

Ad 5. **Smoother compressor aerofoils**

By making the microscopic inconsistencies in the aerofoils of the compressor even smaller. Smaller amounts of grime can accumulate onto the blades. This way the airflow around the blades is smoother, making the compressor more efficient. A more efficient compressor means lower fuel consumption.

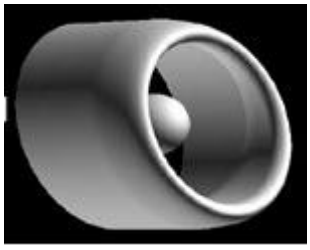


Figure 3-8 - Engine Inlet

Ad 6. **Blisk fan**

When making use of a solid fan instead of a separate blade and hub system. Where the noise production of the fan will decrease, while the efficiency will increase. The so called blisk fan has no separately moving parts, which lowers intern vibration and noise. It also doesn't allow air to leak through, which increases the fan's efficiency.

3.1.5 Materials

Improving the thermal efficiency and power of a gas engine is a demand in this project. One method of increasing both the power and thermal efficiency of the engine is to increase the temperature of the gas entering the turbine. Nowadays the turbine inlet temperature can already be as high as 1500°C, but this temperature exceeds the melting temperature of the metal aerofoils. Therefore it is necessary to look at the different methods of cooling of these aerofoils **(3.1.5.a)** and to look at other or improved high-temperature materials **(3.1.5.b)**.

3.1.5.a Cooling

A typical cooled turbine blade is shown in **Figure 3-9**. As shown, the blade is hollow, so cooling air can pass through the blade internally. The coolant is extracted from the internal channel for impingement and pin-fin cooling. Jet impingement is a very aggressive cooling technique, which very effectively removes heat from the blade wall. However, this technique is not readily applied to the narrow trailing edge. The blade trailing edge is cooled using pin-fins (an array of short cylinders). The pin-fins increase the heat transfer area while effectively mixing the coolant air to lower the wall

temperature of the blades. After impinging on the walls of the aerofoil, the coolant exits the blade and provides a protective film on the blades external surface. Similarly, the coolant traveling through the pin-fin array is ejected from the trailing edge of the aerofoil.
The different cooling methods are:

- Impingement cooling
- Pin fin cooling
- Film cooling

Ad 1. **Impingement cooling**
Impingement cooling is commonly used near the leading edge of the aerofoils, where the heat loads are the highest. With the cooling jets impinging the blade wall, the leading edge is well suited for impingement cooling because of the relatively thick blade wall in this area. Impingement can also be used near the middle of the chord of the blade. **Figure 3-9** shows jet impingement located throughout the cross-section of an inlet blade. Several aspects must be considered when developing efficient cooling designs. The effect of jet-hole size and distribution, cooling channel cross-section, and target surface shape all have significant effects on the heat transfer distribution. Jet impingement is very similar to impingement on a flat plate; however, the sharp curvature at the leading edge of the blade must be considered when using impingement in this region.

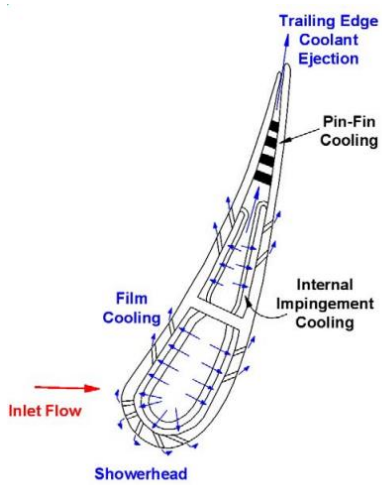


Figure 3-9 - Schematic of a turbine vane cross-section with impingement and trailing edge Pin-fin cooling.

Ad 2. **Pin fin cooling**
Due to manufacturing constraints in the very narrow trailing edge of the blade, pin-fin cooling is typically used to enhance the heat transfer from the blade wall in this region. In a pin-fin array heat is transferred from both the smooth wall and the numerous pins. Flow around the pins in the array is comparable to flow around a single cylinder. As the coolant flows past the pin, the flow separates and wakes are shed downstream of the pin. In addition to this wake formation, a horseshoe vortex forms just upstream of the base of the pin, and the vortex wraps around the pins. This horseshoe vortex creates additional mixing, and thus enhanced heat transfer. Many factors must be considered when investigating pin-fin cooling. The type of pin-fin array and the spacing of the pins in the array effect the heat transfer distribution in the channel. The pin size and shape also have a great impact on the heat transfer in the cooling passage. Because pin-fins are commonly coupled with trailing edge ejection (**Figure 3-9**), the effect of this coolant extraction must also be considered.

Ad 3. **Film cooling**
Film cooling is a major component of the overall cooling of turbine aerofoils. There are holes placed in the body of the aerofoil to allow coolant to pass from the internal cavity to the external surface. The ejection of coolant gas through holes in the aerofoil body results in a layer or “film” of coolant gas flowing along the external surface of the aerofoil. Since this coolant gas is at a lower temperature than the mainstream gas, the heat transfer into the aerofoil is reduced. The adiabatic film effectiveness has a dominant effect in the design of the overall cooling.

3.1.5.b High-temperature materials

The old engine of the Fokker 70, the RR TAY 620-15, consists of modern materials during the development of this engine. Today there are already a number of new materials developed with improved properties. A change in material brings many new features for the components. New materials should always be carefully tested. The materials must be tested for suitability and

durability of the engine. The specific properties must comply with the minimum characteristics mentioned in the materials specifications. Cooling air must be provided so that the fan can run at maximum thrust without damaging the material of the fan blades or changing its properties. In recent years several innovations made to improve the material properties, due developments in materials the thrust to weight ratio of the engine has improved. Designers look carefully what materials are going to be used in the engine and in what improvements can be made to the materials to get even better operating results. The Material innovations are important for different phases of the engine. There is a phase of the compressor chamber, here are stronger materials as necessary. Another reason of the innovation of the materials is the development of materials that can be used under a higher temperature, is used, this is the combustion chamber. Because of the high temperatures in the engine, the designers need a wider range of materials than the airframe designers. The new materials provide improved performance and reliability in the following phases of the engine:

- Compressor
- Combustion chamber
- Turbine

Ad 1. **Compressor**

In the RR TAY 620-15 aluminium was used, but because of the increasingly high temperatures the new engines are developed with titanium. By more renewals there is a composite of titanium alloy with aluminium and vanadium occur. Aluminium aims to strengthen the blades and vanadium is resist against higher temperatures.

Ad 2. **Combustion chamber**

For the combustion chamber various modifications can be made. This is because the temperature in the combustion chamber is higher these days. The engine must comply with a number of new environmental requirements, such as lower emissions and fuel consumption. The RR TAY 620-15 combustion chamber is made of stainless steel, and no longer meets the latest materials and the new environmental requirements. Therefore there is a new material developed specifically for higher temperatures: the alloy HS-188. HS-188 is an alloy of cobalt, nickel, chromium and tungsten. Also this material is resistant to oxidation temperatures below 1095 degrees Celsius and increases the creep limit. In addition a Thermal Barrier coating (TBC) is used to amplify at the higher temperatures.

Ad 3. **Turbine**

The material used in the RR TAY 620-15 is titanium, but tantalum has a higher temperature strength and better oxidation resistance. Today the tantalum material is used for the turbine blades, which consists of single crystals. It is also a more elastic material to absorb the vibrations. The turbine strength can be increased with the addition of rhenium and tungsten to nickel so it becomes a nickel alloy. In addition there are ceramic coatings used to improve a higher operation temperature.

3.2 Conceptual design

With the calculated values of the minimal TSFC for the 1200 NM range and the minimal amount of required thrust during the 2.4% gross second segment climb at MTOW **(Chapter 2)** and the possible modern improvements which can be executed **(3.1)** the conceptual engine can be designed. The designing of the conceptual engine is done with the help of an engine performance model, programmed in Microsoft Excel. The conceptual design is dependent on several variables. Variables like the By Pass Ratio (BPR), the Turbine Inlet Temperature (TIT), the total mass flow (M_L), the Compressor Pressure Ratio (CPR), the Fan Pressure Ratio (FPR) and several efficiencies (η). In order to achieve the minimal TSFC and minimal thrust with the conceptual engine, the different engine variables need to be analysed **(3.2.1)**. With the knowledge about the engine variables, adjustments regarding these variables can be made in order to achieve the efficiency and thrust demands **(3.2.2)**.

If this is successfully fulfilled, adjustments concerning the different components in the conceptual engine can be executed **(3.2.3)**.

3.2.1 Different engine variables

An engine is dependent on six variables which can improve the performances of the conceptual engine, these variables are:

- By Pass Ratio
- Turbine Inlet Temperature
- Mass Flow
- Compressor Pressure Ratio
- Fan Pressure Ratio
- Different efficiencies

Ad 1. **Bypass Ratio**

If the BPR is increased the TSFC will improve. The BPR can be increased by adjusting the shape and direction of the fan blades. Another option to increase the BPR is to decrease the mass flow through the core or increase the mass flow through the bypass section, this can be done by increasing the fan and bypass diameter.

Ad 2. **Turbine Inlet Temperature**

By increasing the TIT the thrust will increase, however the efficiency will decrease. If the TIT is increased, higher temperature resistant materials have to be installed.

Ad 3. **Mass flow**

An increase of the mass flow will increase the thrust. The mass flow is most oftentimes increased when the BPR gets increased. The mass flow will not affect the engine efficiency.

Ad 4. **Compressor pressure ratio**

If the CPR is increased, the thermal efficiency will increase as well. The pressure ratio per stage can be increased with modern day improvements to a maximum of 1.5.

Ad 5. **Fan Pressure Ratio**

If the FPR increases also the efficiency will increase. However at a certain point the pressure leaving the exhaust will be higher than the critical pressure.

Ad 6. **Different efficiencies**

The engine consist out of many components, each component has its own efficiency. Unfortunately exact numbers about these efficiencies are not provided and these numbers are assumed. One thing is certain if the different efficiencies increase the total efficiency will increase as well.

3.2.2 Engine design

The new engine design needs to be far more efficient than the old RR-TAY, the range should increase from 550NM to 1200 NM. The minimum thrust that is needed to take-off from an airfield with an elevation of 5000 feet and successfully complete 2.4% gross second segment climb is **39468N (2.2)**.
At first the efficiencies of the components are far better than 30 years ago. When improving the efficiencies of the RR-TAY 620-15 the values in **(Table 3-2)** are assumed.

Table 3-2

Efficiencies of the improved engine

η_i	η_f	η_{ldc}	η_{vp}	η_{ldt}	η_u	η_m	η_{vb}
0,97	0,97	0,96	0,97	0,96	0,97	0,99	0,97

The price is not leading to reach this performance. When looking at modern comparable engines a table is set up **(Table 3-3)** with maximum performance characteristics of the current turbofan engines. This is necessary to make a spectrum in which there could be designed.

Table 3-3			
Maximum performance characteristics			
FPR	**BPR**	**CPR**	**TIT**
2	9	32	1700

The calculations made in the performance model **(Appendix X)**.The increased efficiencies makes sure that the TSFC immediately decreases from $2.58 \cdot 10^{-5}$ to $2.15 \cdot 10^{-5}$. This is a huge step forward compared to the RR TAY 620-15. At first the CPR is increased from 16 to 25 this makes sure that P_8 increases and the TSFC will be slightly better. The pressure component of the thrust is now relative high when compared with the ambient pressure. By increasing the bypass, the pressure component will decrease and the efficiency of the engine will be better. The only problem is that it is better to keep the exhaust pressure above ambient pressure so the bypass cannot be increased infinite. A BPR of 8 is possible but the engine will be larger than using a lower bypass, so this will not be the final step that is taken by adjusting the BPR.
However modern day engines dispose of a compressor with a CPR around 30. So there is chosen to improve the CPR to 32. Another step that is taken is to decrease the TIT, the only problem was that there was no margin for the pressure component so there is chosen to decrease the BPR to 6 what will result in a smaller engine and a larger pressure component. The TIT can be decreased to 1160. In spite of the large adjustments to the engine the necessary efficiency is still not reached. By slightly lower the FPR, the pressure component and the exhaust pressure will increase so the BPR can be increased to 6.4.
However the efficiency is not reached so there is decided to lower the FPR with another small step to 1.55. This makes sure that the TIT can decrease to 1150 and the BPR can be slightly increased to 6.7. After this adjustments are done the TSFC is decreased to 1.39 with a range of 1202 NM. Only the thrust is too low, this is because the increased efficiency but the mass air flow is still the same. The BPR is more than doubled and assumed that the core will be slightly smaller, the mass flow will increase with 59%. With this mass flow the engine will deliver slightly more thrust at cruising altitude then the current RR-TAY engine.
Beside the thrust and efficiency at cruising altitude also the thrust at 5000ft needs to be verified. When filling in the ISA values at 5000ft with the V_2 airspeed, the engine delivers **71.0 kN** thrust, way more than the required **40,5kN** thrust.

Table 3-4						
Effects of adjustments						
Step No.	**Adjustment**	**Value**	**P18 (Pa)**	**P8 (Pa)**	**TSFC**	**Range (NM)**
Step 1	Efficiencies	X	36189	78301	2,15	777
Step 2	CPR	25	36189	87199	1,99	841
Step 3	BPR	8	36189	32379,36	1,45	1151
Step 4	CPR	32	36189	31227,04	1,39	1200
Step 5	BPR	6	36189	49370	1,52	1100
Step 6	TIT	1160	36189	30148	1,41	1188
Step 7	FPR	1,6	35092	33545	1,42	1180
Step 8	BPR	6,4	35092	30316	1,4	1196
Step 9	FPR	1,55	33996	34050	1,41	1186
Step 10	TIT	1150	33996	32750	1,41	1190
Step 11	**BPR**	**6,7**	**33996**	**30498**	**1,39**	**1202**

With the maximum performance pushed out of the conceptual engine the aircraft should reach 1202NM and can easily complete the 2.4% gross second segment climb at an airport elevation of 5000ft (**Table 3-5**). The calculations of this engine could be found in **Appendix X**.

Table 3-5				
Engine/Aircraft performance				
Thrust TO SL	**Thrust TO 5000ft**	**Thrust Cruise**	**TSFC**	**Range**
83943 N	**70999 N**	12858 N	$1{,}39 \cdot 10^{-5}$	**1202 NM**

3.2.3 Component Improvements

With the optimal performances of the conceptual engine determined, constructional design aspects have to be applied in order to achieve the optimal design aspects. Of the six engine variables earlier discussed only three of those variables are increased to their limits in the conceptual engine design; the BPR, the M_A and the CPR. The BPR is increased to a ratio of 6,7, the maximum M_A is increased by 59 % and the CPR has doubled. In order to achieve these values, structural adjustments have to be made. The BPR is directly linked with M_A. Increasing these variables can simply be done by increasing the diameter of the fan, however this results in unwanted structural effects **(3.2.3.a)**. The CPR can be increased by applying modern compressor stages which can achieve a pressure ratio per stage of 1,5 **(3.2.3.b)**.

3.2.3.a Bypass ratio

The goal is to increase the BPR and the M_A without increasing the fan diameter significantly. This can be achieved by increasing the mass flow through the bypass and/or by decreasing the size of the core section and therefore also decreasing the mass flow of the core section. Both methods are used in order to increase the BPR and increase the total M_A. The mass flow through the core is decreased by 16.36% and the mass flow through the bypass section is increased by 45,75% compared to the RR-TAY 620-15 **(Appendix XI)**. For example during cruise the mass flow through the core has to decrease to from 17,08kg/s (RR-TAY 620-15) to a value of 14,29kg/s (conceptual engine). And the mass flow through the bypass section is increased from 51,92kg/s to a mass flow through the bypass section of 95,71kg/s. This is done for two reasons; first the total mass flow cannot exceed 110kg/s during cruise

otherwise the engine would get overpowered and the engine fan diameter would increase significantly. The second reason is the fixed BPR of 6,7, the BPR cannot be adjusted otherwise the total efficiency would decrease.

Concluding: the mass flow through the core will decrease and the mass flow through the bypass section will increase. This is can be done by reducing the size of the core section, slightly increasing the size of the bypass section and slightly increasing the speed through the bypass section. This results in a slightly larger fan diameter. The speed of the bypass section is limited by the rotation speed of the fan. The rotation speed of the fan is limited by the speed of sound. The rotation speed can be increased by using swept fan blades. The concept of the swept fan blade is exactly the same as sweepback on transonic aircraft wings. Twisted fan blades are used in order to optimise the mass flow over the total span of the fan blade.

3.2.3.b Compressor ratio

The compressor is divided in two sections the LPC and the HPC. The CPR over the LPC is increased to 3 and the CPR over the HPC is increased to 32 for optimal engine performances. With modern day compressor stage ratios of 1,5 it is possible to decrease the number of stages in the compressor. This results in a 3 stage LPC and a 9 stage HPC. With a pressure ratio per stage of 1,4425 for the LPC and a pressure ratio per stage of 1,4697 for the HPC.

4 Comparison Engines

To compare the newly designed engine with other engines, the currently existing engines **(4.1)** need to be looked at. When the important measurements and other variables are known, the comparison **(4.2)** can be made. Finally a conclusion **(4.3)** can be taken to see which engine is best compared to the others.

4.1 Existing engines

The existing engines need to be comparable with the new designed engine and the currently used RR TAY 620. Two engines were found during the research: the General Electric GF34-8 **(4.1.1)** and the Rolls Royce BR710 **(0).**

4.1.1 General Electric CF34-8

The first of the existing engines is the General Electric CF34-8 **(Figure 4-1).** This engine is one of the possible replacements of the currently used RR TAY 620 on the Fokker 70. The CF34-8 is a high-bypass turbofan engine that is used for medium sized aircraft and suitable for fuselage mounting. Compared to the Fokker 70 RR TAY 620, the looks are quite the same, although the CF34-8 is a more advanced engine, with a higher thrust, Bypass Ratio (BPR) and a lower engine weight. The specifications of the CF34-8 can be found in **Table 4-1**.

Figure 4-1 - GE CF34-8

Table 4-1	
General Electric CF34-8	
Thrust	64.5 KN
Bypass ratio	5:1
Inlet Mass flow	199.6 kg/s
Fan diameter	1.17 m
Engine length	3,25 m
Stages compressor	Fran, 10HPC
Stages turbine	2HPT, 4LPT
Compressor ratio	28:1
Diameter	1.30 m
Weight	1100 kg
Thrust to weight ratio	5.3:1
TSFC	$1{,}81 \cdot 10^{-5}$

4.1.2 Rolls Royce BR710

The second of the chosen existing engines is the Rolls Royce BR710 engine **(Figure 4-2).** This engine is a two-shaft, high-bypass-ratio engine with a one-stage low pressure (LP) compressor and a ten-stage high pressure (HP) compressor. A two-stage HP turbine and a two-stage LP turbine drive this engine. The engine features a single low emissions annular combustor with 20 burners. This engine has even an higher thrust than the GE CF34-8 but a lower BPR and a higher engine weight. Other specification of the RR BR710 can be found in **Table 4-2**.

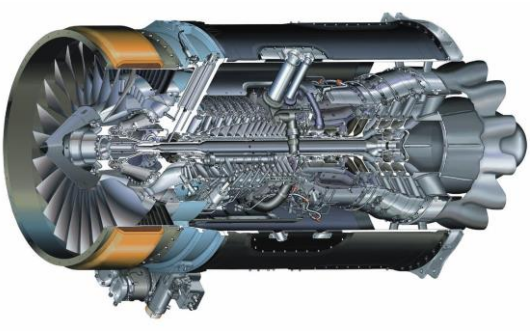


Figure 4-2 - RR BR710

Table 4-2	
Rolls Royce BR710	
Thrust	66.0 KN
Bypass ratio	4:1
Inlet Mass flow	201.8 kg/s
Fan diameter	1.344 m
Engine length	2,21 m
Stages compressor	1LP, 10HP
Stages turbine	2HP, 2LP
Compressor ratio	24:1
Weight	1597 kg
Thrust to weight ratio	41,3 N/kg
TSFC	$1{,}81 \cdot 10^{-5}$ Kg/s/N

4.2 Comparison

The New Engine Design 2F (NED 2F) and the two existing engines ''RR BR710 and GE CF34-8'', suitable as a retro fit for the Fokker 70, have been examined to determine which of the three is most convenient. To make this comparison, a number of critical factors have been chosen and calculated for each engine, together with the old RR TAY 620-15. These factors are the Thrust Specific Fuel Consumption (TSFC), the thrust, the weight, the thrust to weight ratio and the range **Table 4-3**.

Table 4-3					
Engine comparison					
Engine	TSFC [Kg/s/N]	Thrust [kN]	Weight[kg]	Thrust/weight ratio [N/kg]	Range [NM]
RR TAY 620-15	$2{,}58 \cdot 10^{-5}$	61,6	1422 kg	41,3	550
NED 2F	$1{,}39 \cdot 10^{-5}$	79,5	1422 kg	55,9	1202
BR-710	$1{,}81 \cdot 10^{-5}$	66,0	1597 kg	41,3	647
CF34-8	$1{,}81 \cdot 10^{-5}$	64,5	1120 kg	57,6	1178

All of the three substitutes, to the RR TAY 620-15, have got an overall better score when it comes to specifications. However, it is the NED 2F that comes out as an overall best. It has got the lowest TSFC, and the highest range. The maximum thrust is somewhat too high, but can easily be limited to a lower level by an engine control unit. The weight of the NED 2F is assumed to be equal to that of the RR TAY to insure its range is not overstated. Estimated is that due to new technologies and

innovative material use, its weight will be even lower, and therefore its range will be higher. A good second is the GE CF34-8. It has got the same TSFC as the RR BR710, but due to its low weight, more fuel can be taken on board, which increases the range and almost suits the demands.

4.3 Conclusion

As shown in the comparison, the NED 2F is the most suitable engine to replace the RR Tay 620. With a higher thrust and a much lower TSFC, the Fokker 70 will reach the range demand of 1200NM. With a weight as much as the old RR TAY 620-15 it perfectly fits the engine mounting requirements and no changes need to be made to the fuselage. However, the details of this engine have yet to be designed. Designing, testing, and certificating is a process that could take many years, and the costs are going to be high. Nevertheless, the NED 2F its specifications outshine its acquisition cost, and its design is most future oriented. The low fuel consumption will reduce operating costs, and its ability to provide the Fokker 70 with a higher range will generate future income. The high bypass ratio, the chevron exhaust, and the scarf inlet makes the engine one of the quietest in its class, which will improve passenger comfort and airdrome capacity. It is estimated that further designing the engine will take another year, that the testing will take another three years, and that the engine is ready to be produced another year from then. The five years waiting time is a small investment for the innovation and the many years of benefit that the NED 2F will bring.

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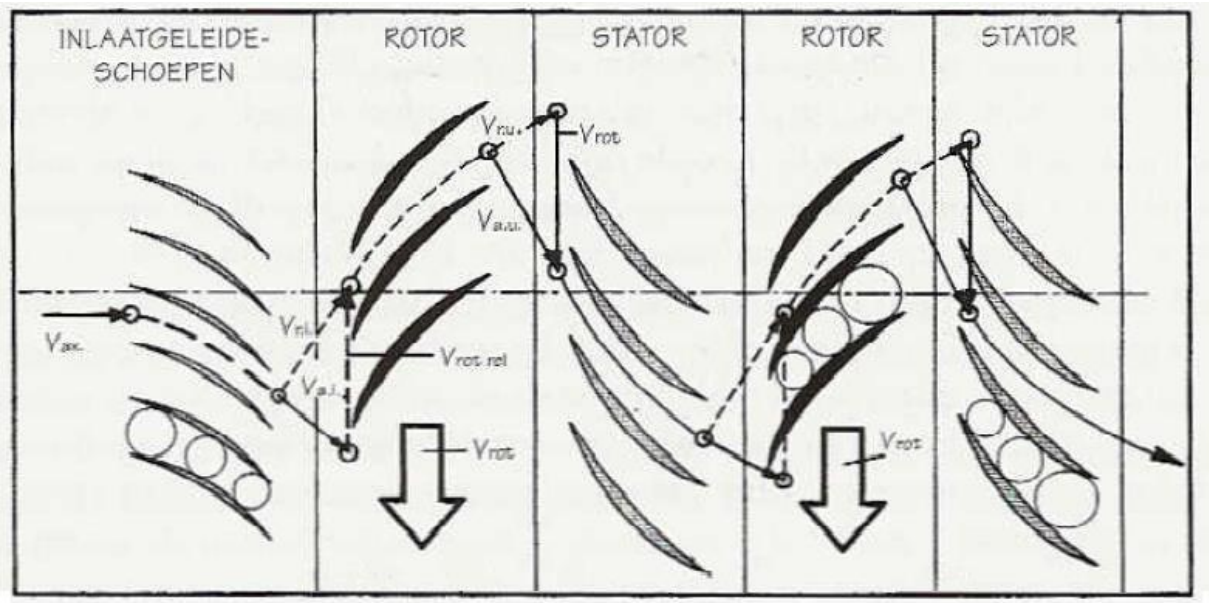
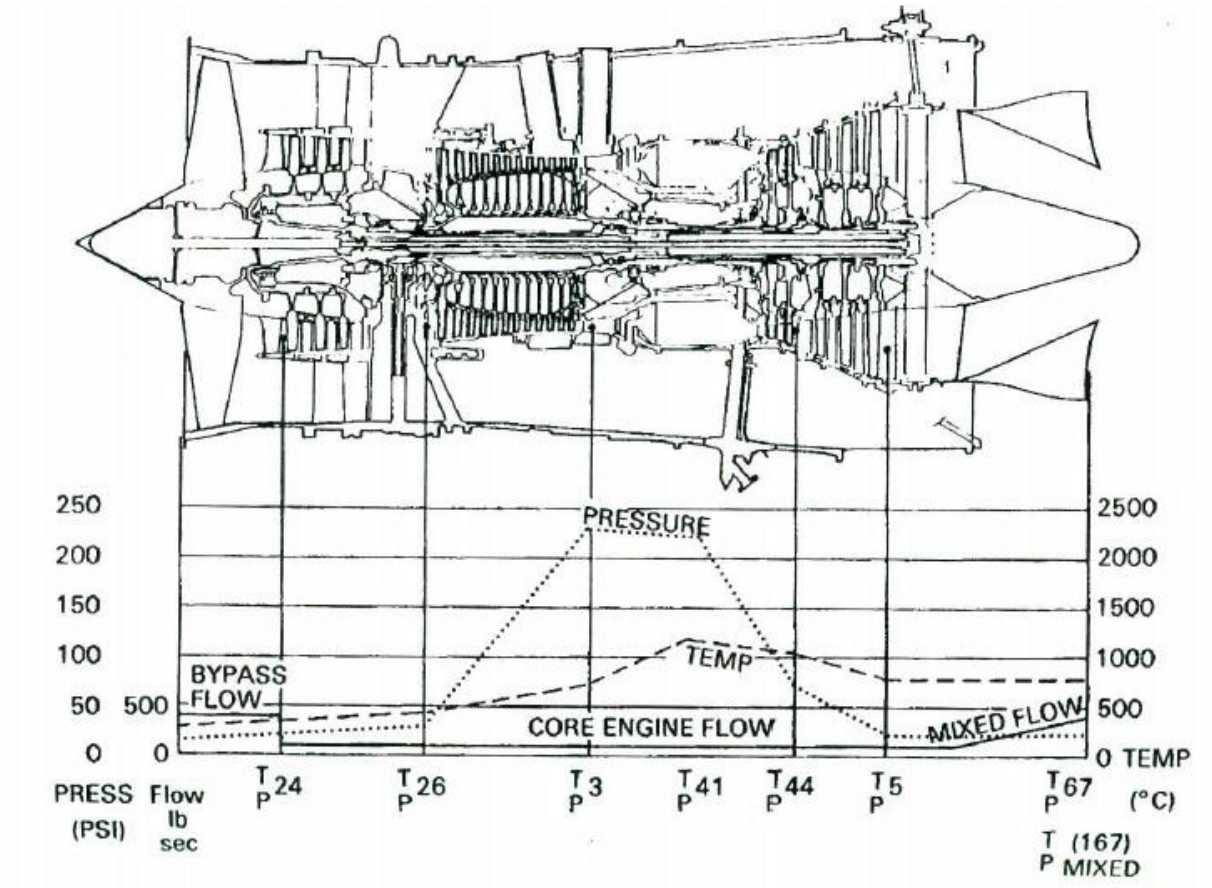
6 Appendices

Appendix I. **Compressor Core**



Appendix II. Engine Specifics



Appendix III. Excel sheet thrust and TSFC Cruising altitude

P_1	30073,86	η_i	0,95	η_m	0,97	BPR	3,04	$C_{p(l)}$	1005
T_1	228,6	η_r	0,88	η_{vb}	0,96	FPR	1,65	$C_{p(g)}$	1148
C_1	221,24	η_{idc}	0,83			TIT	1305	Y_l	1,4
M_L	67	η_{vp}	0,95	η_{hdc}	0,83	CPR_{idc}	2,35	Y_g	1,333
		η_{idt}	0,82			CPR_{tdc}	16	Rs	287
		η_u	0,94	η_{hdt}	0,82				

Cold Part					
T_{01}	252,9518				
T_{02}	252,9518	T_{02}'	251,7342	P_{02}	42142,5
T_{013}	297,1668	T_{013}'	291,861	P_{013}	69535,12
T_{018}	297,1668	T_{018}'			
T_{18}	247,639	T_{18}'	244,4776	P_{18}	35118,85
				P_{cr}	35118,85

C_{18}	315,4383
ρ_{18}	0,494128
m_c	50,41584
A_{18}	0,323454
F_{18}	6380,913

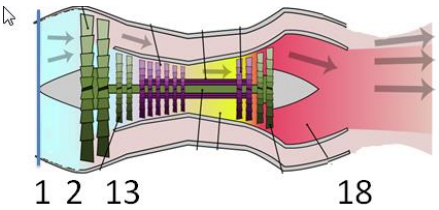
Hot part					
$T_{02,5}$	337,217	$T_{02,5}'$	322,8919	$P_{02,5}$	99034,87
T_{03}	633,7549	T_{03}'	583,3435	P_{03}	674280
T_{04}	1305			P_{04}	640566
$T_{04,5}$	1037,371	$T_{04,5}'$	978,6235	$P_{04,5}$	202399,3
T_{05}	840,0113	T_{05}'	796,6884	P_{05}	70352,89
T_{08}	840,0113				
T_8	720,1126	T_8'	712,4595	P_8	36388,76
				P_{cr}	36388,76

C_8	524,8754
ρ_8	0,17607
m_h	16,58416
A_8	0,179454
F_{tot}	12549,68
F_{spec}	187,3087
H_0	43100000
f	0,019517
TSFC	2,58E-05
η_{tot}	0,199032
R	913,3467

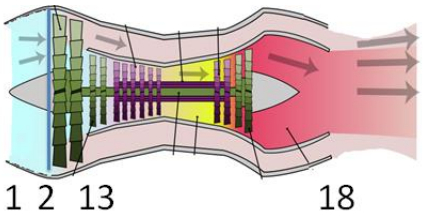
W0	41730
W1	36522

Appendix IV. Formulas calculations Thrust and TSFC

Turbofan – cold



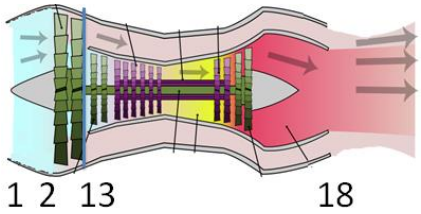
$$T_{01} = T_1 + \frac{{c_1}^2}{2c_{pl}}$$



$$T_{02} = T_{01}$$

$$\eta_i = \frac{T'_{02} - T_1}{T_{02} - T_1}$$

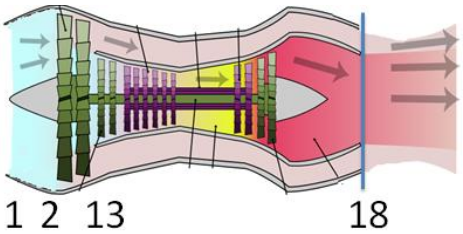
$$\frac{{T_{02}}'}{T_1} = \left(\frac{p_{02}}{p_1}\right)^{\frac{\gamma_l - 1}{\gamma_l}}$$



$$p_{013} = FPR \cdot p_{02}$$

$$\eta_f = \frac{T'_{013} - T_{02}}{T_{013} - T_{02}}$$

$$\frac{{T_{013}}'}{T_{02}} = \left(\frac{p_{013}}{p_{02}}\right)^{\frac{\gamma_l - 1}{\gamma_l}}$$



$$T_{018} = T_{013}$$

$$p_{cr} = p_{013} \cdot \left(1 - \frac{1}{\eta_{u,fan}} \cdot \left(\frac{\gamma_l - 1}{\gamma_l + 1}\right)\right)^{\frac{\gamma_l}{\gamma_l - 1}}$$

$$p_{18} = maximum\ (p_{cr}, p_{atm})$$

$$\frac{{T_{18}}'}{T_{013}} = \left(\frac{p_{18}}{p_{013}}\right)^{\frac{\gamma_l - 1}{\gamma_l}}$$

$$\eta_{u,fan} = \frac{T_{013} - T_{18}}{T_{013} - {T_{18}}'}$$

$p_{18} = p_{atm}$ →

$$T_{018} = T_{18} + \frac{{c_{18}}^2}{2c_{pl}}$$

$$F = \dot{m}_c \cdot (c_{18} - c_1)$$

$p_{18} = p_{cr}$ →

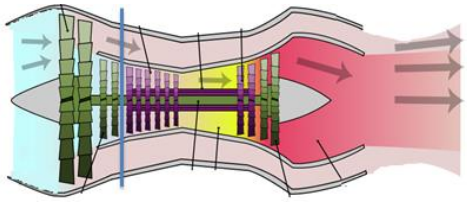
$$c_{18} = c_{geluid} = \sqrt{\gamma_l \cdot R_s \cdot T_{18}}$$

$$\frac{p_{18}}{T_{18}} = \rho_{18} \cdot R_s$$

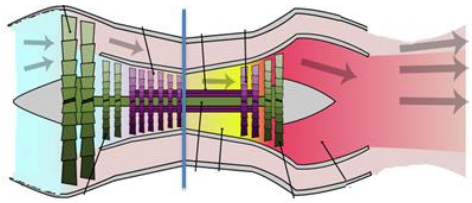
$$\dot{m}_c = \rho_{18} \cdot A_{18} \cdot c_{18}$$

$$F = \dot{m}_c \cdot (c_{18} - c_1) + A_{18} \cdot (p_{18} - p_{atm})$$

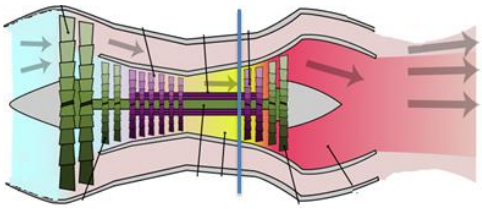
Turbofan – hot



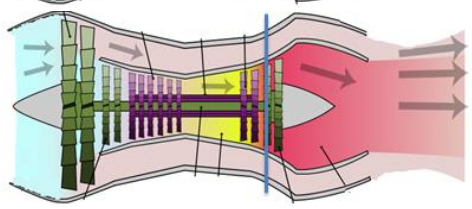
$$p_{02.5} = CPR_{LDC} \cdot p_{02}$$
$$\frac{T_{02.5}{}'}{T_{02}} = (\frac{p_{02.5}}{p_{02}})^{\frac{\gamma_l - 1}{\gamma_l}}$$
$$\eta_c = \frac{T'_{02.5} - T_{02}}{T_{02.5} - T_{02}}$$



$$p_{03} = CPR_{LDC+HDC} \cdot p_{02}$$
$$\frac{T_{03}{}'}{T_{02.5}} = (\frac{p_{03}}{p_{02.5}})^{\frac{\gamma_l - 1}{\gamma_l}}$$
$$\eta_c = \frac{T'_{03} - T_{02.5}}{T_{03} - T_{02.5}}$$

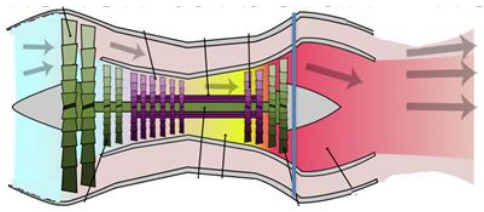


$$p_{04} = \eta_{vp} \cdot p_{03}$$
$$TIT = T_{04}$$

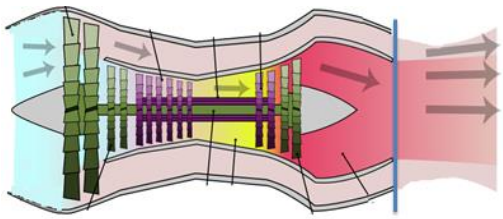


$$\eta_m = \frac{c_{pl} \cdot (T_{03} - T_{02.5})}{-c_{pg} \cdot (T_{04.5} - T_{04})}$$

$$\eta_t = \frac{T_{04.5} - T_{04}}{T'_{04.5} - T_{04}}$$
$$\frac{T'_{04.5}}{T_{04}} = (\frac{p_{04.5}}{p_{04}})^{\frac{\gamma_g - 1}{\gamma_g}}$$



$$\eta_m = \frac{\dot{m}_h \cdot c_{pl} \cdot (T_{02.5} - T_{02}) + \dot{m}_c \cdot c_{pl} \cdot (\dot{T}_{013} - T_{02})}{-m_c \cdot \dot{c}_{pg} \cdot (T_{05} - T_{04.5})}$$
$$\eta_t = \frac{T_{05} - T_{04.5}}{T'_{05} - T_{04.5}}$$
$$\frac{T'_{05}}{T_{04.5}} = (\frac{p_{05}}{p_{04.5}})^{\frac{\gamma_g - 1}{\gamma_g}}$$

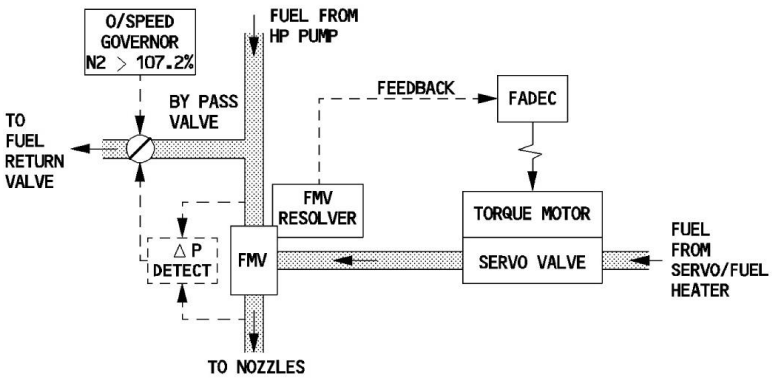


$$T_{08} = T_{05}$$
$$p_{cr} = p_{05} \cdot (1 - \frac{1}{\eta_u} \cdot \left(\frac{\gamma_l - 1}{\gamma_l + 1}\right))^{\frac{\gamma_g}{\gamma_g - 1}}$$
$$p_8 = maximum\ (p_{cr}, p_{atm})$$

$$\frac{T_8{}'}{T_{05}} = (\frac{p_8}{p_{05}})^{\frac{\gamma_g - 1}{\gamma_g}}$$
$$\eta_u = \frac{T_{05} - T_8}{T_{05} - T_8{}'}$$
$$p_{18} = p_{atm} \rightarrow$$
$$T_{08} = T_8 + \frac{c_8{}^2}{2c_{pg}}$$
$$F = \dot{m}_h \cdot (c_8 - c_1)$$
$$p_{18} = p_{cr} \rightarrow$$
$$c_8 = c_{geluid} = \sqrt{\gamma_g \cdot R_s \cdot T_8}$$

$$\frac{p_8}{T_8} = \rho_8 \cdot R_s$$
$$\dot{m}_h = \rho_8 \cdot A_8 \cdot c_8$$
$$F = \dot{m}_h \cdot (c_8 - c_1) + A_8 \cdot (p_8 - p_{atm})$$

Appendix V. HMU



Appendix VI. Performance Formulas (CS-E: AMC E620)

The following corrections from the observed test conditions to the assumed atmospheric conditions of pressure and temperature should be used within the range of conditions appropriate to the particular type of Engine, i.e. taking into account the characteristics of the Engine Control Syst em, and the possible effect of Reynolds Number, unless more accurate or additional corrections for a particular type of Engine have been agreed or required by the Agency.

Gas Pressures:
S.I. Units
$$Pc = P0 \times \frac{1013 \bullet 25}{B}$$
Gas Temperatures:
$$Tc = T0 \times \frac{288}{\theta}$$
Rotational Speed:
$$Nc = N0 \times \sqrt{\frac{288}{\theta}}$$
Thrust:
S.I. Units
$$Fc = F0 \times \frac{1013 \bullet 25}{B}$$
Mass Air Flow:
S.I. Units
$$Wc = W0 \times \frac{1013 \bullet 25}{B} \times \sqrt{\frac{\theta}{288}}$$

In cases where certain proprietary types of air flowmeter are employed, it should be noted that W0 is the actual air consumption of the Engine during test.

Appendix VII. Verification range equation

Operating aspects

In order to verify the range some operating values during ''cruise'' flight have to be examined (Table).

Operating aspect	Value	Reference
MTOW	37.995 kg	KLM F70-F100 AOM
MZFW	33.565 kg	KLM F70-F100 AOM
Fuel capacity MZFW	4430 kg	KLM F70-F00 AOM
CRZ ALT	FL300 / 30.000ft AMSL	Project Assignment
Mach Nr.	0,73	Project Assignment
C_L/C_D	15	Project Assignment
Reserve fuel	-	KLM F70-F00 AOM

Range equation

The determination of the different characteristics of the range formula are listed below.

Thrust Specific Fuel Consumption

Determined in **(1.1.4)** as: $2{,}58 \cdot 10^{-5}\ \frac{kg}{N \cdot s}$

Specific Fuel Consumption

C_T is determined by multiplying the TSFC by the gravitational constant.

$$C_T = TSFC \cdot 9{,}81 = (2{,}58 \cdot 10^{-5}) \cdot 9{,}81 = 2{,}53098 \cdot 10^{-4}$$

True Airspeed

The true airspeed is determined by multiplying the Mach number by the speed of sound. The speed of sound is calculated and has a value of 303 m/s.

$$TAS_0 = M_0 \cdot a = M_0 \cdot \sqrt{\gamma \cdot R_s \cdot T_0}$$

$$TAS_0 = 0{,}73 \cdot \sqrt{1{,}404 \cdot 287 \cdot \big(288 - (0{,}0065 \cdot 30000 \cdot 0{,}3048)\big)}$$

$$TAS_0 = 221{,}54\ m/s$$

Reserves

Following EU-OPS 1.255 the reserve fuel consist out of: alternate fuel plus final reserve fuel plus the contingency fuel. In the KLM F70-F100 AOM the alternate fuel and the final reserve fuel are combined and provided in one table; the alternate table. The alternate table yields a value of 1772 kg with a 100Nm alternate assuming the cruise altitude to the alternate destination will be 7000ft. A company fuel of 300kg is added as well as a contingency fuel of 117,9 kg is added. 117,8 kg is 5% of the initial trip fuel value (2358 kg). The actual trip fuel value is 2240,1 kg.

Reserves:	F70-F100 AOM Alternate Table	: 1772
	Company fuel	: 300
	Contingency fuel	: 117,9 +
	Total reserve fuel	: 2189,9 kg

Weight at end of cruise

The weight at the end of the cruise is calculated by adding the amount of reserve fuel to the MZFW.

W_1 :	MZFW	: 33.565
	Reserves	: 2.189,9 +
	W_1	: 35.754,9 kg

Weight at start of cruise

W_0 :	MTOW	: 37.995

Inserting these values in Breguet's equation yields a value of 424Nm **()**.

-
Range equation
$$424Nm = \frac{2}{2,53098 \cdot 10^{-4}} \cdot 221,54 \cdot 15 \cdot \left(1 - \sqrt{\frac{35.754,9}{37.995}}\right)$$

Appendix VIII. TSFC and reserve fuel calculations

Values		
R	1200	NM
TAS_0	221	m/s
$\frac{C_L}{C_D}$	15	-
$MTOW$	37995	kg
$MZFW$	33565	kg
$Reserves$	1904,9	kg

$$R = \frac{2}{C_T} \cdot TAS_0 \cdot \frac{C_L}{C_D} \cdot \left(1 - \sqrt{\frac{W_1}{W_0}}\right)$$

$$TSFC = \frac{\frac{2}{R} \cdot TAS_0 \cdot \frac{C_L}{C_D} \cdot \left(1 - \sqrt{\frac{W_1}{W_0}}\right)}{9.81}$$

$$Reserves = \frac{Reserves_{old}}{\frac{C_{T\,old}}{C_{T\,new}}}$$

Loop 1

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1904.9}{37995}}\right)}{9.81} = 1.03E - 5$$

$$Reserves = \frac{1904.9}{\frac{2.58}{1.03}} = 760.48$$

Loop 2

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 706.48}{37995}}\right)}{9.81} = 1.509E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.509}{1.03}} = 1114.39$$

Loop 3

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1114.39}{37995}}\right)}{9.81} = 1.36E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.36}{1.03}} = 1004,48$$

Loop 4

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1004,48}{37995}}\right)}{9.81} = 1.407E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.407}{1.03}} = 1038,58$$

Loop 5

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1038,58}{37995}}\right)}{9.81} = 1.392E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.392}{1.03}} = 1028,00$$

Loop 6

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1028,00}{37995}}\right)}{9.81} = 1.397E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.397}{1.03}} = 1031,28$$

Loop 7

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1031,28}{37995}}\right)}{9.81} = 1.395E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.395}{1.03}} = 1030,26$$

Loop 8

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1030,26}{37995}}\right)}{9.81} = 1.396E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.396}{1.03}} = 1030,58$$

Loop 9

$$TSFC = \frac{\frac{2}{1200 \cdot 1852} \cdot 221.54 \cdot 15 \cdot \left(1 - \sqrt{\frac{33565 + 1030,58}{37995}}\right)}{9.81} = 1.396E - 5$$

$$Reserves = \frac{1904.9}{\frac{1.396}{1.03}} = 1030,48$$

Appendix IX. Thrust calculations at 5000ft

Values				
S	93,5	m^2	$\frac{C_L}{C_D}$	15
e	0,79	-	$MTOW$	37995
AR	8,433	-		

Values			
	30000ft	**5000ft**	
Rho	0,4589	1,055	kg/m3
V	221,54	71,51	m/s
Pitch	0	12	Deg

C_{D0} calculation by cruise

$$AR = \frac{b^2}{S} \qquad C_L = \frac{L}{0.5 \cdot \rho \cdot v^2 \cdot S}$$

$$D = C_{D0} \cdot 0.5 \cdot \rho \cdot v^2 * S + \frac{{C_L}^2}{\pi \cdot AR \cdot e} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S = \frac{C_D}{C_L} \cdot W$$

$$C_{D0} = \frac{\frac{C_D}{C_L} \cdot W}{0.5 \cdot \rho \cdot v^2 \cdot S + \frac{\left(\frac{W}{0.5 \cdot \rho \cdot v^2 \cdot S}\right)^2}{\pi \cdot \frac{b^2}{S} \cdot e}}$$

$$C_{D0} = \frac{\frac{1}{15} \cdot 9.81 \cdot 37995}{0.5 \cdot 0.4589 \cdot 221.54^2 \cdot S + \frac{\left(\frac{9.81 \cdot 37995}{0.5 \cdot 0.4589 \cdot 221.54^2 \cdot 93.5}\right)^2}{\pi \cdot 8.433 \cdot 0.79}} = 0.0234$$

Thrust calculations at 5000ft

$$C_L = \frac{W \cdot 0.5 \cdot \rho \cdot v^2 \cdot S}{\cos 1.3748}$$

$$C_L = \frac{9.81 \cdot 37995 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5}{\cos 1.3748} = 1.6148$$

$$T = C_{D0} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S + \frac{{C_L}^2}{\pi \cdot \frac{b^2}{S} \cdot e} \cdot 0.5 \cdot \rho \cdot v^2 \cdot S + W \cdot \sin(\gamma)$$

$$T = 0.0234 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + \frac{1.6148^2}{\pi \cdot 8.433 \cdot 0.79} \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + 9.81 \cdot 37995 \cdot \sin(1.3748) = 43120N$$

$$C_L = \frac{(W - \frac{T \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (T \cdot \sin 1.3748)) \cdot 0.5 \cdot \rho \cdot v^2 \cdot S}{\cos 1.3748}$$

Loop 1

$$C_L = \frac{\left(37995 \cdot 9.81 - \frac{43120 \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (43120 \cdot \sin 1.3748)\right) \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5}{\cos 1.3748} = 1.5362$$

$$T = 0.0234 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + \frac{1.5362^2}{\pi \cdot 8.433 \cdot 0.79} \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + 9.81 \cdot 37995 \cdot \sin(1.3748) = 40390N$$

Loop 2

$$C_L = \frac{\left(37995 \cdot 9.81 - \frac{40390 \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (40390 \cdot \sin 1.3748)\right) \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5}{\cos 1.3748} = 1.5412$$

$$T = 0.0234 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + \frac{1.5412^2}{\pi \cdot 8.433 \cdot 0.79} \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + 9.81 \cdot 37995 \cdot \sin(1.3748) = 40558N$$

Loop 3

$$C_L = \frac{\left(37995 \cdot 9.81 - \frac{40558 \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (40558 \cdot \sin 1.3748)\right) \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5}{\cos 1.3748} = 1.5409$$

$$T = 0.0234 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + \frac{1.5409^2}{\pi \cdot 8.433 \cdot 0.79} \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + 9.81 \cdot 37995 \cdot \sin(1.3748) = 40548N$$

Loop 4

$$C_L = \frac{\left(37995 \cdot 9.81 - \frac{40548 \cdot \cos 1.3748}{\cos 12} \cdot \sin 12 - (40548 \cdot \sin 1.3748)\right) \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5}{\cos 1.3748} = 1.5410$$

$$T = 0.0234 \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + \frac{1.5409^2}{\pi \cdot 8.433 \cdot 0.79} \cdot 0.5 \cdot 1.055 \cdot 68.42^2 \cdot 93.5 + 9.81 \cdot 37995 \cdot \sin(1.3748) = 40549N$$

Appendix X. Calculations modified engines

Performance at sea level (ISA)

P_1	101325	η_i	0,97	η_m	0,99	BPR	6,7	$C_{p(l)}$	1005					C_8	440,4999
T_1	288	η_f	0,97	η_{vb}	0,97	FPR	1,55	$C_{p(g)}$	1148					ρ_8	0,696074
C_1	71,51	η_{ldc}	0,96			TIT	1150	Y_l	1,4					m_h	36,36364
M_L	280	η_{vp}	0,97	η_{hdc}	0,96	CPR_{ldc}	3	Y_g	1,333					A_8	0,118595
		η_{ldt}	0,96			CPR_{tdc}	32	Rs	287					F_{tot}	79213,82
		η_u	0,97	η_{hdt}	0,96									F_{spec}	282,9065
														H_0	43100000
Cold Part								**Hot part**						f	0,011075
T_{01}	290,5441							$T_{02,5}$	402,1428	$T_{02,5}'$	397,6788	$P_{02,5}$	313189,4	TSFC	5,08E-06
T_{02}	290,5441	T_{02}'	290,4678	P_{02}	104396,5			T_{03}	807,0625	T_{03}'	790,8657	P_{03}	3340687	η_{tot}	0,326355
T_{013}	330,4993	T_{013}'	329,3006	P_{013}	161814,5			T_{04}	1150			P_{04}	3240467	R	1063,973
T_{018}	330,4993	T_{018}'						$T_{04,5}$	791,9383	$T_{04,5}'$	777,019	$P_{04,5}$	674577		
T_{18}	290,3655	T_{18}'	289,1243	P_{18}	101325			T_{05}	456,533	T_{05}'	442,5578	P_{05}	65673,04	W0	37995
				P_{cr}	83647,27			T_{08}	456,533					W1	34593,7
								T_8	507,2003	T_8'	508,7673	P_8	101325		
C_{18}	341,5682											P_{cr}	34727,37		
ρ_{18}	1,215877														
m_c	243,6364														
A_{18}	0,586644														
F_{18}	65796,01														

Performance at 5000 feet (ISA)

P_1	84310	η_i	0,97	η_m	0,99	BPR	6,7	$C_{p(l)}$	1005					C_8	439,9956
T_1	278	η_f	0,97	η_{vb}	0,97	FPR	1,55	$C_{p(g)}$	1148					ρ_8	0,580514
C_1	71,51	η_{ldc}	0,96			TIT	1150	Y_l	1,4					m_h	30,38961
M_L	234	η_{vp}	0,97	η_{hdc}	0,96	CPR_{ldc}	3	Y_g	1,333					A_8	0,118977
		η_{ldt}	0,96			CPR_{tdc}	32	Rs	287					F_{tot}	64967,4
		η_u	0,97	η_{hdt}	0,96									F_{spec}	277,6384
														H_0	43100000
Cold Part								**Hot part**						f	0,011737
T_{01}	280,5441							$T_{02,5}$	388,3017	$T_{02,5}'$	383,9914	$P_{02,5}$	260876	TSFC	5,49E-06
T_{02}	280,5441	T_{02}'	280,4678	P_{02}	86958,65			T_{03}	779,2849	T_{03}'	763,6455	P_{03}	2782677	η_{tot}	0,302194
T_{013}	319,1241	T_{013}'	317,9667	P_{013}	134785,9			T_{04}	1150			P_{04}	2699197	R	985,2019
T_{018}	319,1241	T_{018}'						$T_{04,5}$	804,2621	$T_{04,5}'$	789,8563	$P_{04,5}$	599992,3		
T_{18}	280,2889	T_{18}'	279,0878	P_{18}	84310			T_{05}	480,4009	T_{05}'	466,9066	P_{05}	68040,3	W0	37995
				P_{cr}	69675,28			T_{08}	480,4009					W1	34593,7
								T_8	506,0396	T_8'	506,8326	P_8	84310		
C_{18}	335,5892											P_{cr}	35979,16		
ρ_{18}	1,048072														
m_c	203,6104														
A_{18}	0,578896														
F_{18}	53769,26														

Performance at FL300

P_1	30073,86	η_i	0,97	η_m	0,99	BPR	6,7	$C_{p(l)}$	1005
T_1	228,6	η_f	0,97	η_{vb}	0,97	FPR	1,55	$C_{p(g)}$	1148
C_1	221,54	η_{ldc}	0,96			TIT	1150	Y_l	1,4
M_L	102	η_{vp}	0,97	η_{hdc}	0,96	CPR_{ldc}	3	Y_g	1,333
		η_{ldt}	0,96			CPR_{tdc}	32	Rs	287
		η_u	0,97	η_{hdt}	0,96				

C_8	423,204
ρ_8	0,225927
m_h	13,24675
A_8	0,138545
F_{tot}	12885,87
F_{spec}	126,332
H_0	43100000
f	0,013562
TSFC	1,39E-05
η_{tot}	0,368691
R	1201,994
W0	37995
W1	34593,7

Cold Part

T_{01}	253,0179				
T_{02}	253,0179	T_{02}'	252,2854	P_{02}	42466,31
T_{013}	287,8125	T_{013}'	286,7687	P_{013}	65822,78
T_{018}	287,8125	T_{018}'			
T_{18}	239,8438	T_{18}'	238,3602	P_{18}	34025,96
				P_{cr}	34025,96

C_{18}	310,4339
ρ_{18}	0,494311
m_c	88,75325
A_{18}	0,578382
F_{18}	10175,45

Hot part

$T_{02,5}$	350,2026	$T_{02,5}'$	346,3152	$P_{02,5}$	127398,9
T_{03}	702,8236	T_{03}'	688,7187	P_{03}	1358922
T_{04}	1150			P_{04}	1318154
$T_{04,5}$	838,185	$T_{04,5}'$	825,1927	$P_{04,5}$	349110,9
T_{05}	546,1001	T_{05}'	533,9299	P_{05}	57405,41
T_{08}	546,1001				
T_8	468,1527	T_8'	465,7419	P_8	30355,51
				P_{cr}	30355,51

Appendix XI. Mass flow calculation

The mass flow through the core section (hot section, M_h) will decrease by 16,36% and the mass flow through the bypass section (Cold section, M_c) will increase by 45,75% **(Figure)**. The percentages are validated with the help of the data at cruise level.

m_h RR Tay at cruise: $$m_h = m_a \cdot \frac{1}{1+BPR} = 69 \cdot \frac{1}{1+3{,}04} = 17{,}08 kg/s$$

m_h NED 2F at cruise: $$m_h = m_a \cdot \frac{1}{1+BPR} = 110 \cdot \frac{1}{1+6{,}7} = 14{,}29 kg/s$$

Decrease in percentages: $$\frac{17{,}08-14{,}29}{17{,}08} \cdot 100\% = 16{,}36\%$$

m_c RR Tay at cruise: $$m_c = m_a \cdot \frac{BPR}{1+BPR} = 69 \cdot \frac{3{,}04}{1+3{,}04} = 51{,}92 kg/s$$

m_c NED 2F at cruise: $$m_c = m_a \cdot \frac{BPR}{1+BPR} = 110 \cdot \frac{6{,}7}{1+6{,}7} = 95{,}71 kg/s$$

Decrease in percentages: $$\frac{95{,}71-51{,}92}{95{,}71} \cdot 100\% = 45{,}75\%$$

Figure - Mass flow validation

This result in the following values for sea level (SLS) and at 5000ft.

	SLS	5000
m_h	38,12kg/s	32,84kg/s
m_c	255,38kg/s	220,06kg/s

Appendix XII. Procesverslag

Planning en afspraken

Vanaf het begin van het project is door de groep een duidelijke planning gemaakt met de namen van de groepsleden, hun taken, de tijd die hiervoor beschikbaar is en de deadline. Het was de groep helaas niet gelukt om zich hier helemaal aan te houden. Gedurende het project zijn er een aantal momenten geweest waarop de groep de planning via de notulen heeft aangepast. Één van de grootste redenen was dat er hertentamens tussen het project doorliepen, waar soms wel 6 van de 8 projectleden hard voor moesten studeren.
Voor de planning zijn eerst duidelijke afspraken gemaakt tussen de groepsleden en deze zijn opgesteld in het groepscontract. Achteraf heeft de groep veel gebruik gemaakt van de afspraak dat alle wijzigingen in de planning duidelijk in de notulen moesten komen te staan. Ook waren er veel afspraken over te laat komen en/of helemaal niet op de vergadering komen. Hier heeft de groep achteraf weinig gebruik van gemaakt.

Werkproces

De communicatie tussen de groepsleden verliep vrij soepel. Mede hierdoor zijn er geen sancties geweest voor groepsleden. Ieder groepslid die een keer te laat kwam of niet aanwezig kon zijn had hier een geldige reden voor en kon dit vaak op tijd aan de groep melden. De vergaderingen verliepen goed. Soms werden ze wel iets te uitgebreid of te lang. Dit kwam onder andere door het voorzitterschap waar niet iedereen veel ervaring mee had. De voorzitter kon soms de vergadering in betere lijnen leiden dan in deze groep gelukt was.

Groepsleden

Tussen de groepsleden zijn verder geen frustraties of onenigheden voorgekomen.

Verbeteringen

Ieder groepslid had aan het begin van het project een korte lijst opgesteld met verbeterpunten die hij/zij mee had genomen vanuit het vorige project. Voor bijna iedereen was het voorzitterschap een verbeterpunt. Dit hebben we tijdens dit project goed kunnen oefenen tijdens iedere vergadering en elke keer werd er door de groep verbeterpunten genoemd voor de voorzitter.
Een ander verbeterpunt is ook de planning. Deze vonden wij allen beter dan voorgaande keren, maar er was niet goed rekening gehouden met bijvoorbeeld de hertentamens tussen het project door. Dit leidde ertoe dat de planning een aantal keer moest worden aangepast.