AIAA 82-4108

Advanced Engine Technology and its Influence on Aircraft Performance

Ulf Olsson*
Volvo Flygmotor AB, Trollhattan, Sweden

In order to identify problem areas and to investigate the importance of advanced engine technologies, design studies were made on a hypothetical engine for the mid-1980s. The resulting engine has a turbine inlet temperature of 1800 K, overall pressure ratio of 22.9, and thrust-to-weight ratio of 11.5. Technologies involved include boron-aluminum fan blades, a two-stage flash-vaporizing combustor, tungsten-reinforced turbine blades, and full authority electronic control. In fighter mission simulation studies the engine has been shown to result in aircraft superior to those existing today.

Introduction

OVER the last few decades the performance of turbojet engines has steadily increased. If present trends continue, turbine inlet temperatures over 1800 K and thrust-to-weight ratios higher than 10 are to be expected during the 1980s. The present study was undertaken in order to define what type of technologies are needed in order to realize such engines. Another aim of the study was to investigate the performance increases that may be expected from aircraft equipped with such engines as compared to present-day aircrafts.

The studies have been aimed at a hypothetical engine for the mid-1980s, the M85 engine. From a general view of the development tendencies, the turbine inlet temperature of this engine was initially chosen to be 1900 K. This was later modified to 1800 K due to material and cooling problems in the turbine. The cycle of the engine was then obtained from an optimization for a fighter mission. This led to an overall pressure ratio of 22.9, a bypass ratio of 0.5, a total mass flow of 70.7 kg/s, and a thrust at sea level static of 56.0 kN without afterburner (A/B) and 84.4 kN with A/B.

With this cycle as a background, a number of activities were started which were to provide the know-how for the development of such an engine. Examples are computer programs for cycle optimization and compressor and turbine design, and development and testing of combustion chambers, turbine materials, and electronic control systems. The end result of the project was a layout drawing of the engine and performance comparisons with presently existing engines during a hypothetical fighter mission.

Cycle and Systems Analysis

At the outset the M85 cycle was defined against the background of a general judgment of development trends. The aim was to have an engine that was representative of the estimated 1985 technological level. Thus, the turbine inlet temperature was chosen as 1900 K. In order to obtain optimum performance it would then be necessary to have an overall pressure ratio of about 30. For the specific mission chosen, a bypass ratio of 0.5 was considered to be a suitable compromise.

Very early in the studies we found that it would be very difficult to realize such a cycle. With film cooling and directionally solidified materials in the turbine, more than 20% cooling air would be needed. Also, the small dimensions of the blades would make it very difficult to manufacture complicated cooling systems. Cooling of the coolant air in the bypass stream was found to require unproportionally large cooling surfaces. In conclusion, the turbine inlet temperature was lowered to 1800 K.

An investigation as to the influence of the overall pressure ratio (OPR) on engine performance was then made. Increasing OPR gives a lower specific fuel consumption. On the other hand, the weight of the engine will increase due to an increased number of stages and higher wall thicknesses. As a result it was found that the influence on aircraft performance was relatively slight for OPR values between 22 and 27. A value of 25 was chosen as a compromise. It was later lowered to 22.9 as the efficiencies of the compressor and turbine components became better defined.

With lower component efficiencies, the properties of the main and bypass flows changed so that the thrust decreased. In order to achieve maximum thrust while maintaining the rest of the engine process, fan pressure ratio (FPR) had to be lowered from 4.3 to 4.0. OPR then decreased to 22.9. This led to a lower temperature after the compressor and therefore the turbine cooling air could be decreased. The disadvantage of a higher temperature step in the combustion chamber was regarded as manageable.

The turbine inlet temperature and the overall pressure ratio were now kept fixed at 1800 K and 22.9, respectively, while varying the bypass ratio and the total mass flow so as to optimize the engine in a hypothetical fighter mission. This fighter mission, which will be used throughout this study, was of the F-18 type but with shorter range, no drop tanks, and a different arms load. Aircraft performance was measured as the fighter endurance, defined as the number of turns the aircraft could perform at 11 km and Mach 0.9, maximum A/B, and with the fuel remaining after the mission. In the mission, maneuverability at transonic speeds is of importance.

In order to compare the performance of different engines a system of computer programs was developed which calculated steady-state engine performance, pressures, temperatures and mass flows, engine weight, length and diameter, aircraft performance, requisite amount of fuel, weight, and size. In all, 16 different combinations of bypass ratios and total mass flow were studied. Performance, weights, and dimensions of the 16 different engines were calculated and the fighter endurance of the corresponding aircraft compared. This led to a bypass ratio of 0.5 and a total mass flow of 70.7 kg/s. The corresponding thrust at sea level static (SLS) conditions is 56.0 kN without and 84.4 kN with A/B.

Received Jan. 20, 1981; revision received Nov. 25, 1981.

*Research Engineer. Member AIAA.

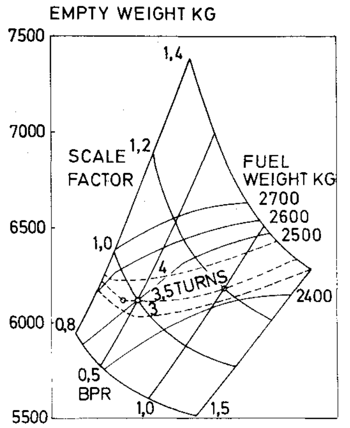

Fig. 1 Results of optimization study.

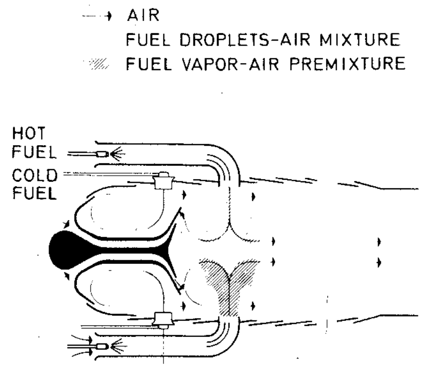

Fig. 2. The M85 combustion chamber.

The results of the optimization study are summarized in Fig.1. It may be seen from Fig. 1 that fighter endurance, as measured by the number of turns of the aircraft, increases with the size of the engine but decreases with increasing bypass ratio. If 3.5 turns are demanded an aircraft with the least empty weight requires an engine having a bypass ratio of 0.4 and a relative size of 0.97. On the other hand, an aircraft with minimum fuel weight requires a bypass ratio of 1.0 and a size of 1.2. If the total starting weight is to be minimized, a bypass ratio of 0.5 and a relative engine size of 1.0 is very nearly optimal. It should be observed, however, that the sensitivity to variations in the bypass ratio is rather small. The total aircraft weight decreases by less than 1% when increasing the bypass ratio from 0.5 to 1.0.

With this defined cycle as a background, detailed design studies of the different engine components were made. These studies and the resulting engine design are described in the following sections.

Fan and Compressor

The M85 fan and compressors are designed so as to meet the cycle specifications outlined previously.

Profile losses are calculated according to NASA SP-36. Shock losses are supposed to be caused by a normal shock at the entrance of the cascade. For the two-thirds of the flow entering the swan neck the efficiency is estimated to be 0.85 and for the one-third entering the bypass duct 0.71. This estimate is in line with test values of existing fans.

There are no shrouds on the blades or other mechanical obstacles in the flow. Therefore, blockage of the flow is caused only by the boundary layers at the hub and casing. It is assumed that blockage increases by 1% per cascade until 5% is reached and then remains constant through the compressor.

The temperature increase in the first fan stage is lowered relative to the second and third stages to give an acceptable hub loading. The tangential velocity is zero in front of the first rotor and behind the last stator. Between the stages a corotation is used to redistribute the aerodynamic loading along the span of the blades. Otherwise, the loading in the hub region is unacceptably high.

The loading and temperature increases are very evenly distributed throughout the compressor. The stage pressure ratio decreases whereas the pressure coefficient increases. The tangential velocity distribution is of the free-vortex type in front of each rotor. A streamline computer program is used to calculate velocities at the inlet and outlet of the cascades.

A blade profile has been defined which is closely related to the MCA profile but more easily used in flowfield calculations. It has the DCA profile as a special case. As the shape of the blades has been defined, calculations of the detailed flowfields within the cascades may be performed. Finite-element methods for compressible flow including shocks have been developed to this extent.

The theory is essentially a matrix method, but contrary to other related methods, it does not rely on an incompressible solution as a first approximation. Instead, an initial solution is generated which satisfies conservation of mass and energy. This solution is then successively improved by iterations so as to meet the remaining condition of irrotational flow.

The concluding calculation that needs to be made is an estimate of the compressor map. This is made by a stage-stacking method. The M85 fan has three stages and an overall pressure ratio of 3.96 in the design point at a temperature increase of 163 K. There is no pressure profile at the fan outlet. The specific inlet mass flow is 204 $kg/m^2 \cdot s$ which corresponds to an inlet flow Mach number of 0.62. The axial velocity increases somewhat over the fan, giving an outlet Mach number of 0.51. These values are in accordance with today's standard.

The pressure coefficient at the hub of the first stage is 0.85 which is below the recommended maximum value. A measure of the mean axial diffusion is the de Haller number. The smallest values occur at the hub of stage 2 with 0.67 for the rotor and 0.74 for the stator. These values should be large enough to insure a stage diffusion factor less than recommended 0.5.

The compressor consists of five stages giving a pressure ratio of 5.72. The efficiency is estimated to be 0.85 and the temperature increase is 322 K. The axial Mach number is 0.60 at the inlet and 0.38 at the outlet. The axial velocity is reduced through the whole compressor. The tip velocity of stage 1 is 441 m/s, corresponding to a relative flow Mach number of 1.25 toward the rotor blade tip.

The largest absolute angles occur at the hub of stator 1 where the entrance angle is 50.7 deg. The loading as measured by the de Haller number is evenly distributed. The lowest value 0.66 is found at the hub of rotor 3. The diffusion factor will be 0.52 at that point, which is slightly higher than the recommended maximum value of 0.5.

The Combustion Chamber

For high-temperature combustion chambers, cooling of the flame tube is of paramount importance. This is so not only because of the high temperature itself. To produce the high temperature, more air is needed in the combustion zone. Less air will then be available for cooling and for dilution. A resulting deterioration in temperature traverse quality can be expected.

The M85 combustor as shown in Fig. 2 has a two-stage fuel injection with flash vaporization of the secondary fuel. The fuel is heated above the dew-point temperature by means of a heat exchanger which takes heat from the hot compressor air. Vaporization is prevented within the fuel system by means of a high pressure level. As the fuel is injected into the premixing tubes, the pressure decreases and an instantaneous flash vaporization is obtained.

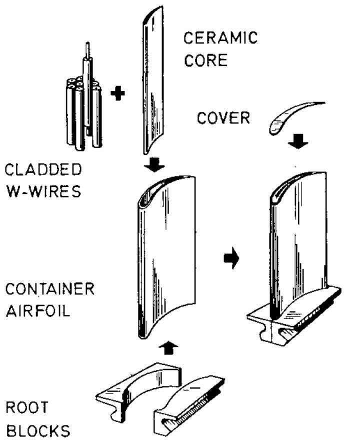

Fig. 3 Production of high-temperature composite material.

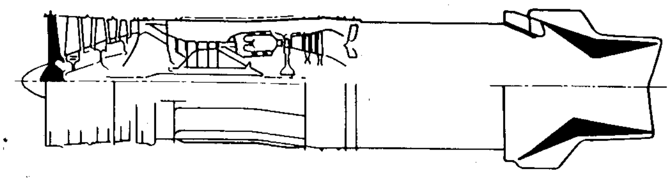

Fig. 4 Design of the M85 engine.

To relieve thermal stresses in the flame tube, the film cooling system has been improved by the introduction of a double-wall design with impingement cooling of the inner wall and with slots for film cooling air. The design of such cooling systems was studied both analytically and experimentally.

The M85 combustor was designed using a computer program for concept evaluation as well as water-tunnel visualization of the flowfields in a transparent model.

Tests at conditions corresponding to different engine power settings have shown that the two-stage combustion allows a large control area with a high and maintained efficiency. Substantial reductions in emissions were achieved with secondary premixing tubes instead of airblast atomizers. Flash vaporization instead of cold fuel in the premixing tubes did not reduce the emissions substantially. However, prevaporization is favorable for the temperature profile.

Turbine

The M85 turbine comprises a single-stage, high-temperature (HT) turbine and a two-stage, low-temperature (LT) turbine. Attempts were made to reduce the LT turbine to one stage, but then the diameter must be increased in order to achieve sufficient torque while not overloading the blades aerodynamically. With regard to the flow pattern through the turbine the HT-turbine diameter must also increase. The result is an increase of the total weight of the turbines.

Now, the diameter of the LT turbine can decrease if the bypass ratio is lowered. It was found that if bypass ratio was decreased from 0.5 to 0.35 the total turbine weight with one LT stage was lower than with two such stages. However, this results in higher aerodynamic loads on the fan blades and it was found necessary to increase the number of stages in the fan from three to four. This increased the total engine weight, offsetting any gains in the turbine.

The turbine cascades were designed using streamline calculations and a special computer program which determines blade profiles with minimal losses. The flowfield was analyzed in detail with finite-element methods and the blade profiles adjusted accordingly.

Studies concerning the design of cooled turbine blades and the influence of different cooling systems on the aerodynamic and thermodynamic performances of the turbine have been made. Methods to determine velocity and temperature profiles at film-cooled surfaces have been developed.

To meet the very high turbine inlet temperature in the M85 engine, new materials are investigated. A composite material of tungsten-reinforced superalloys is under development. A unique production process is used which seems to lead to creep properties superior to other materials of this type, see Fig. 3.

The nickel-based matrix is obtained in the form of small diameter tubing which is used to clad the tungsten-reinforced wire. The clad wire is then cut into lengths corresponding to the intended specimen length and parallel bundles of these lengths are packed into container tubes. After evacuation and sealing the containers are compacted in hot isostatic pressing.

The total coolant mass flow in the turbine is 10.5%. This value is chosen as a compromise between losses in the cycle and the necessity of maintaining a sufficient coolant flow. It is distributed with 4% in the HT stator, 3% in the HT rotor, 2% in the first-stage LT stator, and 1.5% in the first-stage LT rotor. The second LT stage is uncooled.

The blades of the HT rotor are most critical. Calculations have shown that at one-third the blade length from the root and at the midchord section, where the tensile stresses are highest, the material temperatures with pure convection cooling surpass 1400 K. With 12 rows of film-cooling holes, the material temperature may be decreased to 1200 K. In the forward edge of the blade, where the stresses are compressive, the temperature is 1700 K without and 1450 K with film cooling.

The composite material described previously may endure a temperature of 1350 K. Even with such materials film cooling should be necessary. Ordinary superalloys may endure 1250 K and could not be used here.

It should be noted that the composite material with 50% tungsten fibers is nearly twice as heavy as steel. This means high centrifugal stresses. In the midchord section of the blade, the stress is about 300 MPa. This is still sufficiently below the rupture stress of the composite, which is 2.5 times as high at the given temperature.

There are many parameters influencing turbine design. Two of the more important ones are the dimensionless work output ψ and the axial to rotational velocity ratio ϕ. Low values of these parameters imply low gas velocities and longer blades but also reduced friction losses and therefore better efficiency. On the other hand, a low ψ means more stages for a given turbine output and a low ϕ means a larger turbine annulus area for a given mass flow.

The proper choice must be a compromise between all of these aspects. In the M85 turbine the demand to keep the blade long has led to a choice of $\psi = 3.6$ in the HT turbine which is below the generally recommended minimum value of $\psi = 4$. Also the ϕ value chosen is 0.65 which is close to the lowest recommended value of 0.6. In the LT turbines, the demand for a smooth flow path in the turbine channel keeps the ϕ value in the first stage to 0.85 which is close to the lowest recommended value of 0.8 for LT turbines.

The blade length has been governing the choice of reaction rate, i.e., the relative amount of stage expansion which takes place in the rotor. The lowest value in the M85 turbine is 0.15 in the second LT stage which is exactly the lowest recommended value.

In order to avoid outlet guide vanes, the gas angle at the exit from the last stage should be less than 5 deg. In the M85 this angle is 10 deg which is thus higher than current practice. There is also a requirement that the gas angle at a rotor inlet be kept below 70 deg. In the M85 the highest value is 67 deg in the HT stage.

Design of the Engine

The M85 engine is a dual-rotor turbofan with mixed exhaust (Fig. 4). The fan has three stages with variable vanes in the first stage. It is powered by a two-stage turbine with cooling in the first stage. The HT compressor has five stages with variable geometry in the inlet guide vanes and in the vanes of the first two stages. It is powered by a single-stage cooled turbine. The combustion chamber is of the annular type and smokeless. The augmentor has an adjustable exhaust nozzle which is also the tail of the aircraft. The electronic control system is of the full-authority type for both the gas generator and the augmentor and with a simple reserve system. The engine has a modular design.

For the first-stage fan blades, a titanium alloy with FOD-capacity is chosen. For the other fan blades a boron-aluminum composite material is used. The maximum stress in the first-stage fan disk is 380 MPa with blades in Ti 8-1-1 and 290 MPa with boron-aluminum. However, titanium was chosen because of its foreign object damage capacity.

The first-stage blades are precision forged with plasma-sprayed shrouds. All of the blades have axial dovetails. The disks are forged, friction welded, and bolted together. The houses are forged and have abradable seals. Vanes are of Ti 8-1-1, disks of Ti 6-2-4-6, and houses of Ti 6A1 4V. The variable vanes in the two first HT compressor stages are forged in Ti 8-1-1. The rest of the vanes are forged in Waspalloy. The first-stage blades are Ti 8-1-1, the second-stage blades Inco 901, and the rest of the blades Waspalloy. All are electrochemically machined. Houses are Ti 6-2-4-2 and Inco 718. The combustion chamber flame tube is made from a nickel alloy TD Ni-Cr with a coating of zirconium dioxide.

The HT turbine uses blades in tungsten wire-reinforced Fe Cr Al Y alloy and directionally solidified material Cotac DSE in the vanes. The HT turbine disk is made from hot isostatic pressed IN100. The rest of the turbine disk and houses are made from a nickel alloy, Waspalloy. The HT turbine disk is designed so that the hub stress is close to the $\sigma_{0.2}$ limit 1000 MPa. The LT turbine vanes and blades are in Cotac DSE and the houses are electron beam welded from Waspalloy.

The augmentor is constructed with titanium in the front shells and with a Ni alloy in the hotter rear parts. The flame holder is designed with a V-shaped primary ring from which extend radial V profiles which in turn are distributed so that the total blocking effect becomes about 30%. Fixed injectors were chosen. The flame tube is convectively cooled on the outside and film cooled on the inside.

Two different nozzles have been considered, viz a conventional axisymmetric, convergent-divergent nozzle and a two-dimensional nozzle of a new type with a sliding, divided flap. The two-dimensional nozzle has been designed so that the total engine length is the same in both cases. The M85 control system will be a full-authority system, which controls compressor guide vanes, gas generator fuel flow, afterburner fuel flow, and exhaust nozzle area. Experiments with such systems have been made.

The engine consists of the following separate modules: fan, front casing, core, LT turbine, augmentor, auxiliaries, and bypass duct.

At SLS the fan outlet temperature is 451 K, the compressor outlet temperature 772 K, turbine inlet temperature 1800 K, HT-turbine outlet temperature 1465 K, LT-turbine outlet temperature 1242 K, and mixing temperature 1009 K. The temperature in the afterburner is 2115 K.

Aircraft Studies

From the cycle and design data given previously, the thrust-to-weight ratio of the M85 engine with afterburner at sea level conditions is estimated to be 11.5. As is seen from Fig. 5 (from Ref. 1) this is well within the performance profile expected in the mid-1980s.

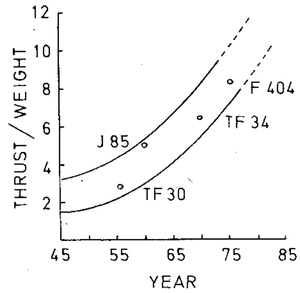

Fig. 5 Increasing trend of thrust-to-weight ratio.

In order to study the influence of future engine technology on aircraft performance, we shall compare the M85 engine to the F404, a current modern engine, in the fighter mission described previously. As a first step in this comparison we will define suitable inlet and wing configurations for the aircraft. Aircraft with three different sizes of each engine will then be compared.

Data for two different types of inlets have been used in a study of their influence on the M85 performance. One is the pitot inlet, which at supersonic flow conditions gives a straight shock. The other is an oblique-shock variable inlet. The M85 engine process is used as a basis for the comparison. Inlet performance is for conventionally body-situated inlets with the engine in the aft fuselage. The inlet areas have been dimensionalized at 11 km, maximum power, ISA atmosphere, Mach 0.8 and drag coefficient 0.97 for pitot inlets, and Mach 2.0, drag coefficient 0.76 for oblique-shock inlets.

Pressure loss and spillover drag characteristics of the two inlets are known. At Mach 2 the oblique-shock inlet has 9% pressure loss as against 28% for the pitot inlet. Below Mach 0.9 both inlets have a few percent pressure loss. For partial power it is assumed that the pressure loss is proportional to the square of the mass flow. The inlet pressure loss decreases pressure and mass flow through the engine, thus reducing the thrust level. Below Mach 2, a 1% increased pressure loss was found to reduce the thrust by about 1.5%. The oblique-shock inlet has a somewhat higher spillover drag below Mach 1.8 than the pitot inlet at all altitudes. For the oblique-shock inlet the spillover drag is comparable to the thrust reduction due to total pressure loss. Pressure losses decrease and spillover losses increase with reduced power level.

In conclusion, as far as pressure and spillover losses are concerned, oblique-shock inlets are superior to pitot inlets at high Mach numbers and high power levels. Where maneuverability and fighter endurance at transonic Mach numbers are important, as in the fighter mission studied here, pitot inlets may be more advantageous. Furthermore, oblique-shock inlets are heavier and larger than corresponding pitot inlets with the same airflow capacity and have a larger wet area which increases external drag. In the present study the inlet areas were 0.416 and 0.325 m^2, respectively. To this must be added arrangements for variable-flow areas. Costs for development, manufacturing, and overhaul are higher. Because of these considerations, pitot inlets were preferred for the present study.

The next part of the study is a choice between three different aircraft configurations: swept wing, delta wing, and canard wing. For this comparison, weight and performance of the M85 engine were scaled to the F.404 total mass flow at SLS maximum power. The aircraft studied were of the single-engine type with internal fuel tanks. They were dimensioned according to the Swedish B3LA attack fuel-dimensioning mission. Aircraft performances were then simulated in the fighter mission.

A comparison of different aircraft configurations is very dependent on the choice of the geometrical and aerodynamic data. With these reservations, the results show that the aircraft with canard will be the heaviest one. This depends on a higher level of the drag curve. The delta wing configuration will have least weight but requires the same amount of fuel as the swept wing configuration.

Aircraft with canard wings have the lowest performance which depends on the higher weight. Aircraft performance is best for the delta wing configuration, except for turning performance and maximum altitude at subsonic velocities where aircraft with swept wings are best. The difference is especially large at maximum velocity. Aircraft with swept wings are superior to aircraft with canard wings, except for maximum velocity with A/B where the two configurations are equal due to the more advantageous drag characteristics of the canard-winged configuration. Because of these results, aircraft with swept wings were chosen for the mission studied here.

Aerodynamic data for the swept wing configuration were taken from Swedish studies of new military aircraft. Aircraft with three different sizes of each engine were studied. All of the aircraft had a single-engine installation with afterburner, the engine being situated in the aft fuselage of the aircraft. The aircraft had internal fuel tanks. Weights and dimensions of the aircraft were found according to the Swedish B3LA attack mission. The structure factor and the wing loading at landing were the same for all aircraft.

Engine performance was found from simulations of the M85 and F.404 cycles. The weight of the M85 engine was calculated with computer programs. Elementary aircraft performance as to acceleration capability, maximum velocity, and maximum altitude have been calculated in a flight envelope for a fighter load alternative. The results from the studies of different engine sizes show that the bigger the engine, the heavier the aircraft but the better the performances. The required amount of fuel is about the same for the M85 and F.404 of the same size. An aircraft with the basic M85 size, 70.7 kg/s total flow, has about the same weight and dimensions as an aircraft with the basic size F.404.

If the M85 and F.404 engines are scaled to the same total flow the specific excess power of the M85 aircraft is 20% higher than that with the F.404 at sea level Mach 0.5, 50% better at 11 km Mach 0.9 without afterburner, and 70% better at 11 km Mach 1.4 with afterburner. The maximum velocity at sea level is 5% higher without and 20% higher with afterburner. At 11 km it is 3% higher without and 9% higher with afterburner.

A M85 aircraft climbs 20% more rapidly to 11 km at Mach 0.8 than a F.404 aircraft. Takeoff roll distance is 9% shorter. The turning velocity is 5% better at sea level and Mach 0.5 both with and without afterburner. Empty aircraft weight is 7% and wing area 9% lower. If the M85 engine is scaled to the same total mass flow as the F.404 at SLS maximum power, the fighter endurance of the corresponding aircraft is three times as high.

In conclusion, the results of the fighter mission simulation studies indicate that considerable gains in aircraft performance can be obtained with advanced engine technologies.

Conclusions

Design studies have been carried out on a hypothetical engine for the mid-1980s. Cycle optimization for an F-18 type fighter mission resulted in a turbine inlet temperature of 1800 K, overall pressure ratio 22.9, bypass ratio 0.5, total mass flow 70.7, and a thrust at sea level static of 56.0 kN without and 84.4 kN with afterburner. Thrust-to-weight ratio was found to be 11.5 which is in line with the expected development trend.

The technologies required to realize such an engine involved boron-aluminum fan blades, two-stage recirculating flash-vaporizing combustors, tungsten-reinforced turbine blades, and full-authority electronic control systems. Experiments with such technologies have been made.

In order to study the influence of such advanced engines on aircraft performance the M85 engine was compared to the F.404. Aircraft with pitot inlets and swept wing configuration were found to be best suited to the studied fighter mission. Aircraft with three different sizes of each engine were compared. It was found that aircraft with 1985-type engines were superior to present-day aircraft in all respects.

Acknowledgment

This study was undertaken on grants from the Swedish Air Material Board.

Reference

[1] Quinn, B., "Research Trends and Opportunities in Air Breathing Propulsion," AIAA Paper 78-44, Jan. 1978.