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Challenges in the Silent Aircraft Engine Design

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The Silent Aircraft Initiative goal is to design an aircraft that is imperceptible above background noise outside the airport boundary. The aircraft that fulfils this objective must also be economically competitive with conventional aircraft of the future and therefore fuel consumption and mechanical reliability are key considerations for the design. To meet these ambitious targets, a multi-fan embedded turbofan engine with boundary layer ingestion has been proposed. This configuration includes several new technologies including a variable area nozzle, a complex high-power transmission system, a Low Pressure turbine designed for low-noise, an axial-radial HP compressor, advanced acoustic liners and a low-speed fan optimized for both cruise and off-design operation. These technologies, in combination, enable a low-noise and fuel efficient propulsion system but they also introduce significant challenges into the design. These challenges include difficulties in predicting the noise and performance of the new components but there are also challenges in reducing the design risks and proving that the new concepts are realizable. This paper presents the details of the engine configuration that has been developed for the Silent Aircraft application. It describes the design approach used for the critical components and discusses the benefits of the new technologies. The new technologies are expected to offer significant benefits in noise reduction without compromising fuel burn. However, more detailed design and further research are required to fully control the additional risks generated by the system complexity.

Nomenclature

AGMA	American Gear Manufacturer Association
BPF	Blade Passing Frequency
BPR	Bypass Ratio
C_p	Specific heat at constant pressure
ESS	Engine Section Stator
FPR	Fan Pressure Ratio
h_0	Stagnation enthalpy
HP	High Pressure
IP	Intermediate Pressure
LP	Low Pressure
M_{ff}	Fan face Mach number
OGV	Outlet Guide Vane
OPR	Overall Pressure Ratio
R	Rotor
$T_{0, ref}$	Stagnation reference temperature
ToC	Top-of-Climb
U	Blade speed

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V Vane

Greek symbols

Ψ Blade loading

Δ Increment

γ Ratio of specific heats at constant pressure and constant volume

η_p Polytropic efficiency

I. Introduction

For the Silent Aircraft Initiative project noise emission has been used as the primary design driver. The overall goal was to develop a concept aircraft that would be imperceptible above background noise in a typical urban environment outside an airport. In addition to this aggressive noise target, the objective was to produce a viable commercial aircraft that would be economic relative to other passenger aircraft of the future. In terms of the engine design, the aggressive noise target of the project necessitated significant reductions in all components of engine noise relative to today's turbofans. In other words, a design was needed that would simultaneously achieve reductions in jet noise, fan rearward noise, fan forward noise, compressor noise, turbine noise and combustion noise. To be a viable propulsion system, a new engine concept needs to have competitive fuel burn and low pollutant emissions. Furthermore, for a safe and economic operation, the engines must be reliable and robust in all conditions, have low vulnerability to failure, acceptable acquisition costs and require minimal maintenance.

In order to satisfy all the requirements listed above, a novel propulsion system is required, the development of which presents many challenges. The Silent Aircraft engine design started with a series of studies into what overall configuration would have the potential to satisfy the noise target of the project. This work, described in [1], showed that a very high bypass ratio turbofan with some form of variable exhaust was required to deliver ultra-low jet noise during take-off whilst maintaining an acceptable size of engine. Furthermore, to effectively attenuate rearward turbomachinery noise, an embedded engine system was expected to be necessary. This has the advantages of enabling boundary layer ingestion and reducing the drag incurred by the engine installation. However, it was accepted at this stage, that an embedded installation with a variable exhaust would also introduce significant complications into the design. Notably, an embedded configuration suffers from greater inlet duct losses that can have a highly detrimental effect on performance. Also, the engine fan in such an installation has to be tolerant of non-uniform inlet flow conditions as well as being compatible with a variable exhaust.

Subsequent work [2] investigated possible engine thermodynamic cycles and overall mechanical arrangements that could power the first generation of Silent Aircraft airframe designs that are described in [3]. At this stage a 4-engine system was assumed with embedded installations incorporating variable exhaust nozzles. Suitable turbofan engine cycles using expected 2025 engine technologies were optimized for minimum specific fuel consumption and engine diameter. The operation of the engine off-design was also examined to demonstrate the noise benefits of the variable exhaust nozzle. Based on this work, three different engines architectures were proposed for the Silent Aircraft. These are reproduced in figure 1, below. The bare engines were compared in terms of their potential for noise reduction and impact on fuel consumption, but there was also the requirement to minimize the overall size and weight of the propulsion system. From these considerations, a multiple-fan embedded design with boundary layer ingestion was expect to have the greatest potential of meeting the project noise and fuel burn targets. This style of configuration is as shown on the far right of figure 1.

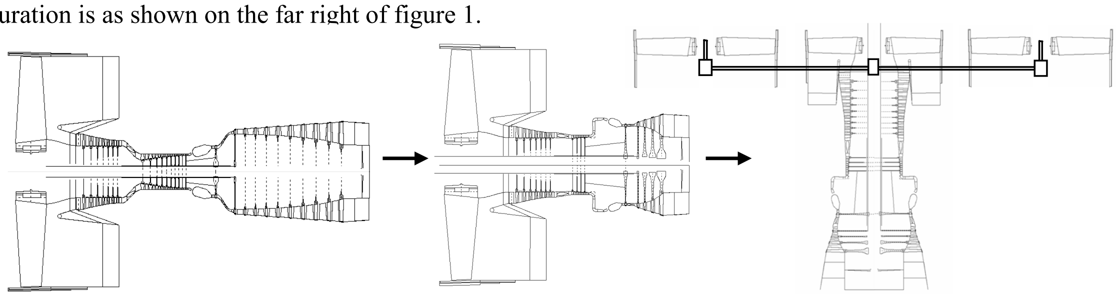


Figure 1: Evolution of the bare engine design, taken from [2]

There are several characteristics of a multiple fan design that can lead to a reduction of the noise level outside of the airport boundary. Firstly, the transmission system between the Low Pressure turbine and the fan enables the turbine to be designed at a more optimum rotational shaft speed. A Low Pressure turbine with a new design philosophy can reduce the source noise, both, tonal and broadband. Secondly, the reduction of the fan diameter leads to narrower exhaust ducts (higher length-to-diameter ratio) that, when lined with acoustic absorbers, are more effective at removing the rearward fan noise. Furthermore, smaller fans have higher rotational speeds leading to an increased blade passing frequency and noise that is more readily attenuated.

Once a multiple-fan embedded configuration of propulsion system was selected for the Silent Aircraft, significant effort was directed at improving the integration of this design with the airframe and in matching this engine to the thrust requirements of the aircraft. These aspects are covered in the companion paper [4]. In addition, work focused on more detailed design of the unconventional parts of the engine, such as the Low Pressure spool system and the distributed fan system. Distributed propulsion has been proposed as a means to improve fuel-efficiency of aircraft in several studies such as [5], [6]. However, these studies were primarily concerned with the overall performance of distributed propulsion and do not consider the design of the engines in any detail. For the Silent Aircraft the distributed propulsion aspect of the engine is fundamental to the design. The current paper will describe the design approach used for the fan system, the LP turbine and the geared transmission system used in the engine. It will show how these components have been optimized for minimum noise, weight and fuel burn within a multiple fan configuration.

The approach to this paper is to start with an overview of the final engine design for the Silent Aircraft. This will describe some of the novel features in the design and the new technologies proposed to achieve both a quiet and fuel-efficient solution. The design approaches used for each of the Low Pressure spool components (the fans, LP turbine and the transmission system) are then described in detail. These components are the most critical for the design as these have a dominant impact on the noise, fuel-burn and packaging requirements. The core of the engine, in contrast, is very similar to today's turbofan technology. The methodology used to specify the characteristics of each of the key components is described and the implications of the design choices on the engine noise, weight and performance are described. Through developing the design of the LP components a range of technical challenges are uncovered and a comparison between two LP spool system designs are presented, which shows the impact of the transmission system on the overall engine characteristics. The paper ends with conclusions of the design studies and includes recommendations for further work that is required to overcome the technical barriers to such an engine becoming reality.

This paper makes a new contribution to the field of novel engine design for future aircraft propulsion. It details a new engine configuration aimed at achieving an integrated low-noise and efficient solution for the Silent Aircraft. It also considers the real-world problems that need to be tackled during the design of a new engine configuration and proposes a way forward for developing some of the technologies further. The work presented is important because features of the Silent Aircraft engine design could have application to other future propulsion systems. The challenges encountered in the Silent Aircraft engine design may be common to other designs and in order for these to be developed successfully it is essential that the main challenges are understood at the outset.

II. Key features of the final Silent Aircraft engine design

The final engine design detailed here combined the findings from the previous engine studies [1], [2], with the optimized airframe characteristics [7] and the operations requirements for minimum noise [8-9]. The key parameters for the final propulsion system configuration are summarized within Table 1. Figure 2a shows the corresponding layout of the engine architecture with some of the principal design features marked on and Figure 2b shows how the engines are installed in the airframe.

Number of engines	3
Number of fans	9
Fan diameter	1.20 m
Bare engine length	2.46 m
Mass per engine	2985 kg
Max. Thrust per engine	150 kN

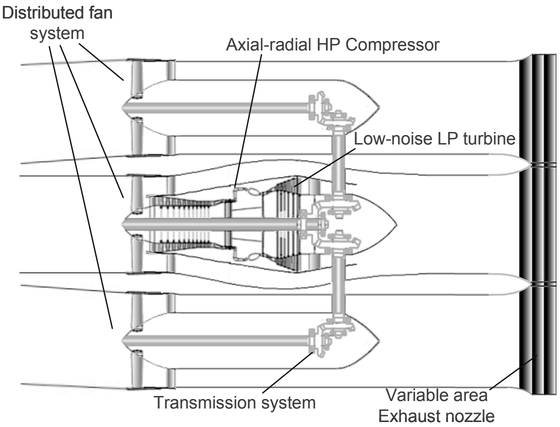
	ToC	**Take-Off**
FPR	1.50	1.19
BPR	12.3	18.3
OPR	48.8	24.2
Nozzle area	+0%	+45%

Table 1. Key Silent Aircraft engine design parameters

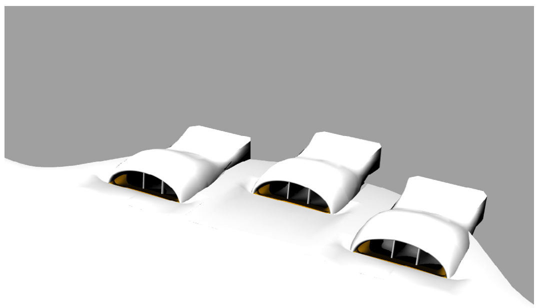
A 3-engine configuration with 9 fans was chosen as the best compromise between a practical design and a system that would achieve the benefits of distributed propulsion. A 3-engine system can be easily packaged at the rear of the all-lifting airframe. More engines would have a negative impact on the design of the airframe control surfaces and there is a standard design rule for engine turbine disk burst trajectories that would be very difficult to satisfy with more than three engines. A design with three engines enables greater freedom in the power management of the engines during take-off than would be possible with two-engines. This is due to the climb requirements following the loss of an engine [8]. As the application of power management was critical to the design to enable reduced fan and jet noise, three engines was the minimum number that could be chosen.

With each engine core driving three fans, the fan diameter is reduced by a factor of $\sqrt{3}$ relative to a single core driving a single fan. Thus the installation wetted area is reduced by a factor of about $\sqrt{3}$, and the proportion of airframe boundary layer air that can be ingested by the engines is increased by $\sim\sqrt{3}$ [1]. The fuel burn of a 3-engine system with each core driving 3 fans was therefore expected to be significantly better. In terms of noise, the available length-to-diameter of exhaust ducts available for acoustic liner was increased by $\sim\sqrt{3}$ and the fan blade passing frequency should also increase by the same factor. The expected bare engine weight benefit of having three fans per core, neglecting the installation and transmission system weight, was estimated as a 20% reduction [1]. However, as shown in section V, the additional weight of the transmission system and its associated lubrication system was found to negate a large part of this benefit.

Each core driving three fans gives a more practical overall arrangement than driving 2 or 4 fans per core primarily because the layout is symmetrical. Having a single core drive more than 4 fans could be possible, but the transmission system would become more complex. As indicated by figure 2, and as described later in section V, the transmission system design for a single core driving 3 fans has shaft and gear wheel sizes that can be comfortably accommodated within the engine layout. The fan diameter of 1.20 m, with an engine length of 2.46 m leads to a relatively long and narrow bypass duct that is ideal for installing acoustic liners on both the hub and casing. A single fan system driven by a single core would have a bypass duct that would be much less effective at acoustic absorption. Downstream of the engine components, there is still sufficient space to fit two fan diameters worth of acoustically lined exhaust duct where mixing also takes place between the core and bypass flow in the central duct.



a) Layout of the engine



b) Packaging of the 3-engines with the airframe

Figure 2. Engine for the Silent Aircraft

Each engine has a variable exhaust nozzle at the duct exits. This is a common two-dimensional nozzle arrangement that is carefully controlled to optimize the fan working line and to minimize take-off jet noise. One key novel aspect of the engine is that the fan has been designed from the start for optimized operation with the variable area exhaust nozzle. This has had a strong influence on the fan design as it has to operate at a much wider range of off-design operating points with minimal noise at take-off and approach, and maximum efficiency at cruise. The operation of the fan with the variable area nozzle is described in [10] and the impact on the design is discussed further in section III below. The nozzle used in the final engine design is continuously variable during take-off between the minimum area at ToC and a maximum area that is 46.9% larger.

Note that the engine cycle is an ultra-high bypass ratio turbofan matched to the variable area nozzle. 2025 estimates of components efficiencies and metallurgical temperature limits were used for the cycle specification and a pressure recovery of 0.946 at ToC and cruise has been applied based on an S-shaped inlet duct design [4]. The engine for the Silent Aircraft is a bleed less engine with power off-take extracted from the High Pressure turbine. Each engine is designed to deliver 722 kW for power-off take, which is comparable to the requirements of today's latest aircraft designs. In terms of the maximum thrust from each engine, 150kN is fairly small by today's turbofan standards (a Trent 800 produces more than 400kN). This is primarily because the maximum take-off weight of the Silent Aircraft is only 151 tonnes. This is significantly less than a Boeing-767, which is powered by only 2 engines. The low thrust requirement of the engines, at such high bypass ratios (BPRs), leads to a small engine core size. This is a further reason for having 3 engine cores rather than 4 or more because more cores would further reduce the size of the engine core components and increase losses due to low Reynolds numbers and relatively large clearance gaps in the rotating compressor blades and engine seal flows.

The Fan Pressure Ratio (FPR) at ToC is the most critical design parameter for the engine cycle (see also [2]). The choice of design FPR is a compromise that is driven by several factors: A lower value leads to higher propulsive efficiency at the cost of a larger engine size, which increases the total installed drag. In terms of noise, as FPR reduces it becomes easier to meet the jet noise target and the nozzle area change needed between take-off and ToC is minimized [10]. Fan source noise also tends to reduce with FPR. Unfortunately, a lower FPR design is heavier and more sensitive to inlet distortion and to installation pressure losses. A ToC FPR of 1.5 was chosen as the lowest possible value that would be achievable with a robust mechanical design, see section III. It is worth pointing out that a fan pressure ratio of 1.5 at ToC is comparable to current technology standards. It was initially thought that ultra low-noise would require much lower FPR values and thus higher fan diameters and bypass ratios. However, this is avoided by combining the fan with the variable area nozzle that effectively creates a variable cycle engine that is optimized for both low-noise at take-off and high fuel efficiency at cruise, see also [1], [2].

The Overall Pressure Ratio (OPR) of the engine was constrained by material temperature limits and optimization of the core cycle to minimize fuel consumption. The strategy used to derive the optimum engine cycle is described in detail within [2]. Note that the very low core size and high OPR leads to the requirement for an axial-radial high-pressure compressor. This is added to remove the requirement for S-shaped ducts in the annulus and to increase the minimum blade height in the HP compressor, thus making the engine core easier to manufacture. Overall, this makes the engine more compact, although there is a weight penalty for the large centrifugal compressor disk and impeller.

The transmission system of the engine introduces various complications and challenges into the design and these are explored in detail within section V. However, the transmission system enables the distributed fan system that leads to large potential noise and performance improvements. In addition, the transmission system introduces a degree of freedom into the Low-Pressure (LP) turbine design that would not be possible with a fixed shaft between the LP turbine and the fan. Overall, it is found that the transmission system and the distributed fan system are critical in order to reach the aggressive goal of the Silent Aircraft project.

III. Fan Design

A. Challenges / Requirements

The fan is a critical component of the engine. In terms of performance it provides the majority of the engine thrust and it has the greatest influence on overall fuel consumption. In terms of noise, fan source noise is one of the dominant sources on take-off and, with the fan providing the majority of the thrust; fan operation determines jet parameters and the resulting jet noise. Before discussing fan design in detail, the key challenges and requirements for the fan stage are summarized.

The top of climb point is defined as the maximum aerodynamic operating point with the fan operating at 100% corrected speed. This is the key design point with the fan having to be able to provide the required thrust with adequate operability margins. At cruise the thrust requirement is reduced but the fuel consumption is critical requiring the fan to operate at peak efficiency. For conventional aircraft and engine designs with a fixed nozzle area the take-off working line is closer to stall than the cruise working line especially at reduced fan speed. This potentially limits the operating range of the fan at take-off and requires design compromises so that take-off stall margin is adequate. For the Silent Aircraft design, nozzle area is increased at take-off to reduce jet noise. This moves the operating point on the fan map to higher mass flow and lower pressure rise; away from stall. Whilst this makes maintaining adequate stall margin easier it does introduce new challenges and requirements not present for conventional designs. At high fan speed the amount the nozzle area can be increased and the jet noise reduced is limited by fan rotor choking. At lower speed negative incidence onto the OGVs becomes limiting. Therefore, a fan designed to deliver low jet noise with a variable area nozzle must support low pressure rise at high capacity with as much margin available as possible before encountering choking at high speed or excessive negative incidence onto the OGVs at low speed.

To reach the noise target of the Silent Aircraft fan source noise must be minimized. This is discussed in more detail in [8], but to summarize, the fan should be operated at high efficiency for low noise with blade speed set low enough so that rotor alone noise is cut-off at the critical flyover position but not so low that the blade is overloaded which can result in increased noise.

Finally, the engines ingest the airframe suction surface boundary layer to improve cruise fuel burn [4]. Due to this, flow distortion is persistent at the fan face for the full duration of the aircraft mission. Whilst the inlet can be designed to minimize the distortion [4], supporting the required distortion level without reducing efficiency or excessively reducing stall margin will remain a significant challenge. Whilst this is discussed briefly in [4] and corrections are made to account for the impact in take-off noise [8], the fan design discussed below has been optimized for clean inlet flow.

B. Selection of key parameters

With jet noise a dominant noise source at take-off and significant jet noise reduction only possible through modification of the engine cycle the design (ToC) FPR was selected so that jet noise would be within the Silent Aircraft target of 60 dBA [8]. Figure 3, taken from [8] shows the variation in ToC, cruise, sideline and flyover pressure ratios with jet noise limit at take-off. Whilst the design was iterative and the result of figure 3 are sensitive to airframe performance and take-off weight, the ToC stage fan pressure ratio, which was set to 1.50 before the final airframe parameters were available, is at a level that enables the Silent Aircraft noise target to be met. A lower ToC FPR would allow reduced jet noise but this increases engine size and makes the fan more susceptible to inlet distortion.

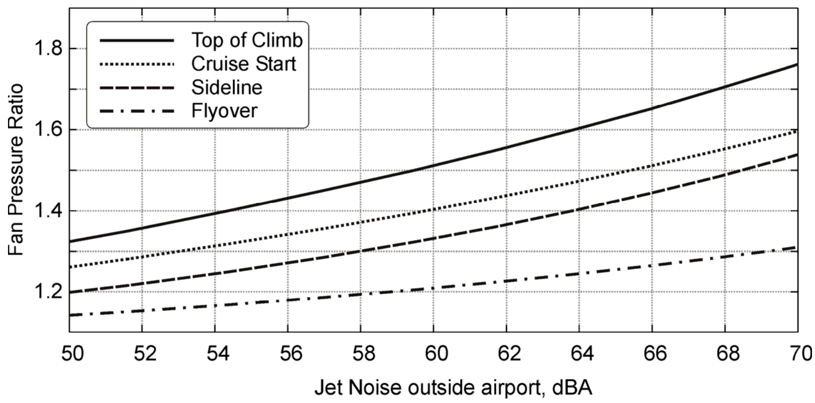


Figure 3. Optimized take-off links jet noise to fan pressure ratio at key operating conditions (from [8])

Rotor blade count was set to 20 to give the required solidity and the hub-tip-ratio was set to 0.29, in the range of existing designs. The OGV blade count was set to 44 so as to cut-off the first BPF tone.

To operate with subsonic relative Mach number at the fan tip during the entire take-off (as discussed in [10]) for the final take-off profile in [8] would require ToC corrected blade speed to be set to 258m/s giving a blade loading of 1.08 for a 0.29 hub to tip radius ratio and 92% efficiency (Eq. 1). This blade loading is significantly higher than current commercial or even research fans and would have poor performance and high noise [11-13]. Meeting this condition at just flyover requires a ToC corrected blade speed of 395m/s corresponding to a loading of 0.46, which is close to the value used in current designs. Based on these results and the noise results reported in references [11-13] the ToC corrected blade speed was set to 371m/s giving a stage loading of 0.52.

$$\psi = \frac{\Delta h_0}{U^2} = \frac{c_p T_{0,ref}\left(FPR^{\frac{\gamma-1}{\eta_p \gamma}} - 1\right)}{\frac{1}{2}\left(1 + htr^2\right)U_{tip,corr}^2} \tag{1}$$

C. Design and Performance

The final fan stage is illustrated in figure 4 with blade counts of 20 rotors and 44 OGV as discussed above. The design presented here is equivalent to the average of the core fan and the two auxiliary fans combined in that the design bypass ratio is 13.5 and the inlet tip radius is 1.033m.

Fan aerodynamic design was performed using MULTIP, a highly developed 3D viscous solver widely used in industry and described in more detail by Denton and Xu [14]. When predicting performance at off-design conditions untwist due to centrifugal and pressure forces was accounted for. The blades were modelled as solid titanium Ti6Al4V and deformation predicted for different rotational speeds. Pressure distribution at approximate peak efficiency for the speed in question was used and the deformation assumed constant for a given speed. Generation of the solid blade mesh from the CFD mesh was performed in Patran and Abaqus was used for the finite element modelling. The resulting deformation at a given speed was converted into a rotation of the blade profile about a fixed point for each of the design stream surfaces and used as input to the CFD code.

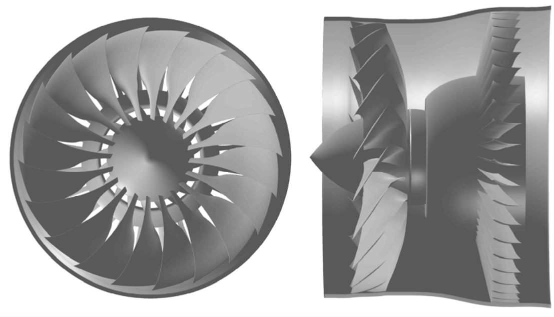


Figure 4. Graphical view of fan stage from head-on and to the side

For conventional fan designs the fan will twist as it accelerates from cold static conditions to hot full speed conditions. This leads to the blade tip metal angle (measured from the axial direction) being greater at part speed than at high speed. With a fixed nozzle area this can be beneficial as the take-off blade metal angle (at reduced fan speed) is greater than the ToC metal angle (at close to 100% speed). This moves the stall line to a higher pressure rise and lower mass flow rate for off-design blade speeds. For the Silent Aircraft fan we want to achieve the opposite – higher mass flow rate and lower pressure rise at part speed. Therefore reducing or reversing the normal blade twist would be beneficial. This can be achieved through forward sweep [15] and this concept was utilized in this design. Figure 5 presents the estimated blade stress and deflection at the ToC design point. Whilst there is up to 6.5mm deflection of the blade at the tip this is the same at both the leading and trailing edges leading to no overall change in blade metal angle.

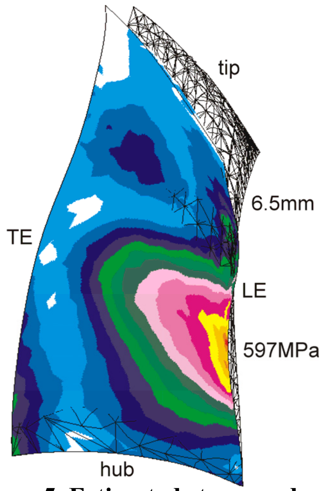


Figure 5. Estimated stress and deformation at ToC design point

The resulting fan rotor characteristic is shown in figure 6, which plots pressure rise and polytropic efficiency versus capacity. The fan face area used in calculating the capacity is that at the fan leading edge whilst pressure rise and efficiency calculations are mass averaged values from 2.4 axial (midspan) chords upstream to 1.1 axial chords downstream. Choking capacity at 100% corrected speed is 1.149 equating to a fan face Mach number of 0.674 whilst the peak rotor only polytropic efficiency of 95.3% occurs at 90% corrected speed. This efficiency value is in line with trends and occurs at the correct speed for low cruise fuel burn.

Figure 7 shows radial variation in fan rotor pressure rise, fan face Mach number and polytropic efficiency at four key operating locations; ToC, cruise, sideline and flyover. The ToC position, specified to enable 300ft/min climb rate at 40,000ft ISA+10K, is at 100% corrected speed providing a mass averaged pressure rise of 1.52. Pressure rise is evenly distributed across the span but the fan face Mach number is slightly higher nearer the hub than the tip. Efficiency peaks near the hub and drops off towards the tip due to shock losses. The cruise point shown here is at 92.5% speed close to the peak efficiency location. Pressure rise is again approximately constant across the span with tip efficiency improved due to reduced shock loss. At the take-off locations of sideline (max climb) and flyover (reduced thrust as airport boundary is crossed) the nozzle area is increased to minimize jet noise. This has the effect of reducing pressure rise at the tip in particular, which also reduces efficiency. The sideline position is at 90% speed whilst the flyover position is at 70%. The reduced speed at flyover gives very low pressure rise across the span.

With forward noise shielded the design of the engine section stator (ESS) was of secondary importance and therefore the downstream duct and OGV design was performed with a specified bleed mass flow used to remove core flow as illustrated in figure 8. This figure also shows streamlines through the fan at the ToC design point.

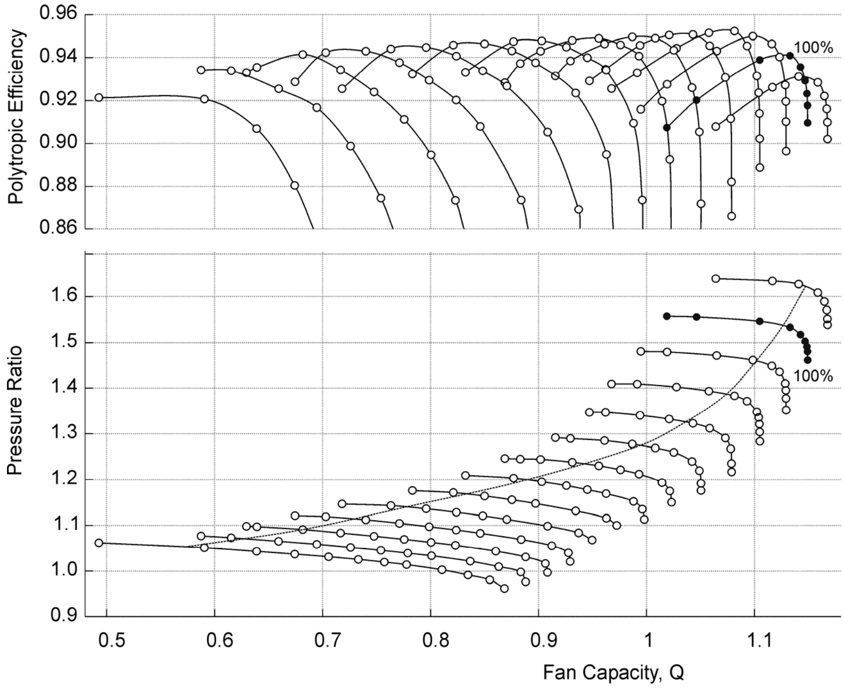


Figure 6. Rotor only characteristic. Lines of constant corrected fan speed from 40% to 105% in 5% increments. Trace of maximum efficiency locus shown on pressure rise plot.

Initial duct designs between the rotor trailing edge and OGV leading edge led to a high incidence range (difference between the flow swirl angle onto OGV leading edge at top of climb and take-off) having to be supported at high Mach number. Modification of the duct enabled a reduction in the incidence range seen by the OGVs. For a constant radius and constant area duct the incidence range that the OGVs needed to support ranged from 8° at the hub to approximately 21° at 90% span, outside of the casing boundary layer as illustrated in figure 9. Maximum positive incidence occurs at the ToC condition where loading is greatest and maximum negative incidence occurs during take-off when loading is least. Also shown in figure 9 is the cruise Mach number variation at OGV leading edge which, for the fixed radius design, varied from 0.95 at the hub to 0.5 at 90% span.

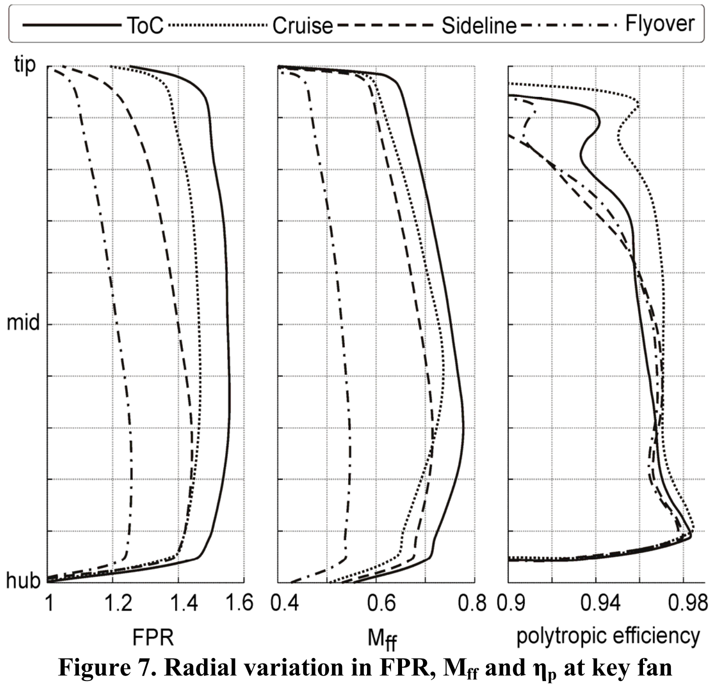


Figure 7. Radial variation in FPR, M_{ff} and η_p at key fan operating points

By increasing the duct radius, whilst maintaining constant duct area, it was possible to reduce the cruise Mach number at the hub and the swirl variation at the tip. This is illustrated in figure 9 with hub cruise Mach number reduced to 0.76 and swirl variation at 90% span reduced to 18° although Mach number is increased slightly here to 0.58. Although the requirements placed onto the OGVs remain a significant challenge, the operational range required is close to that demonstrated in the literature [16-17].

The negative incidence requirement of the OGVs is at lower Mach numbers than indicated in figure 9 because the rotor is at take-off rather than cruise. This difference in the requirements at different operating conditions has been accounted for when designing the OGV. Blade profiles between 10% and 90% span were created using Mises [18] combined with optimization software. The inlet conditions and streamtube parameters for these calculations were taken from initial 3D CFD results. Three conditions were considered for the section design: the cruise point, which was the nominal design condition for the OGVs, the ToC point which determined positive incidence range and a take-off point, between flyover and sideline, which determined negative incidence range. This procedure enabled a fixed geometry OGV design to be produced which, from preliminary analysis, gives acceptable performance at all flight conditions.

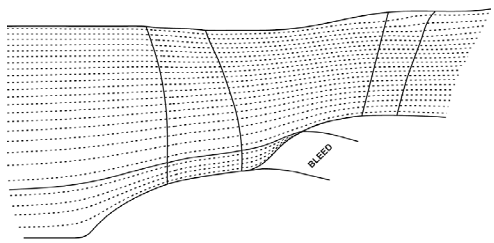


Figure 8. Modelling downstream of rotor

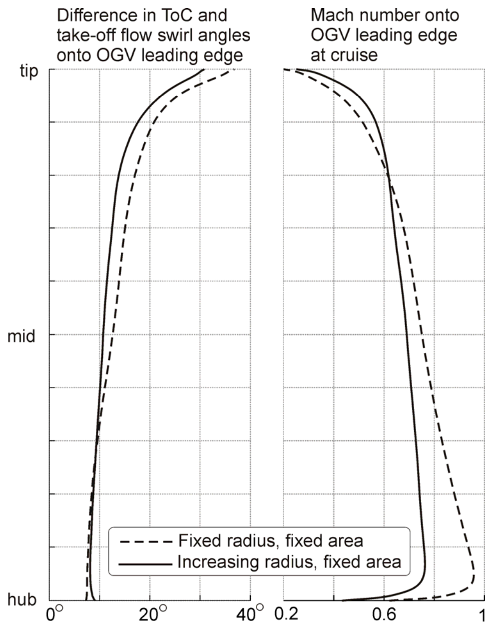


Figure 9. Effect of increasing radius on OGV flow conditions at cruise for fixed area duct

IV. Low Pressure Turbine and transmission system design

The purpose of the LP turbine and transmission system is to transmit the required power to the fan system at the appropriate rotational speed for all flight conditions. With a multiple fan design, as illustrated in Figure 2a, the transmission system is considerably more complex than for a conventional turbofan because the power must be divided and redirected to the separate fans. In such a configuration, the transmission system can also be designed so that the LP turbine rotational speed is different from that of the fans.

This section considers the LP turbine and transmission systems together because their designs are highly coupled. The first part shows how the choice of key parameters for the LP turbine influences the noise, weight and performance of the system. The second part describes the process for designing the transmission system. This shows how a transmission system can be designed for aerospace levels of reliability with minimal complexity, once the LP turbine rotational speed has been fixed. The third part considers the cooling system required. This is critical for providing continuous lubrication and cooling to the transmission system.

In the following section (section V), the design methods developed in this section are applied to two designs of LP system - one with the LP turbine rotating at the same rotational speed as the fans, the other with the LP turbine rotating significantly faster.

A. Low Pressure turbine design

The design of a Low Pressure (LP) turbine is a compromise between different areas, such as, aerodynamics, mechanical design, stress, vibration, weight, cost and noise. The Low Pressure Turbine is one of the major contributor to the engine noise of the Silent Aircraft. The aim for this study was to design a low noise LP turbine whilst minimizing the weight and length.

The conceptual design of the Low Pressure turbine has been carried out with the help of Rolls Royce proprietary software. This provides a preliminary layout of the LP turbine with an estimation of the weight. Data from two different flight points are needed for the turbine design. The point on the flight envelope for the aerodynamic design of the LP turbine is Top-of-Climb where the fan pressure ratio is the highest. For the mechanical design it is the Sea Level Static condition where maximum thrust is encountered. The design input data consists of two sets of information - the cycle thermodynamic data and the mechanical data. The cycle data includes the mass flow through the turbine and the total pressure and the total temperature at the inlet and exit of the turbine. The values of the cooling mass flow and its total pressure and total temperature are also specified. Geometry data is specified at the inlet of the Low Pressure turbine by the requirement to match with the annulus of the High Pressure turbine (for a two spool design) or the Intermediate Pressure turbine (for a three spool design). The geometry at the exit, i.e., the axial and the radial positions of the hub and the tip are also specified. The mechanical data input includes information on the turbine speed, some details of the type of turbine construction and some stressing and cooling data.

To achieve a low-noise and low-weight LP turbine design, the absolute rotational speed, the number of stages, the number of blades per row, the axial chord length and the gap between rows are all input parameters that have been chosen carefully.

The use of a geared transmission system in the LP system means that the turbine rotational speed does not have to match the fan rotational speed. The rotational speed is therefore a design variable and a trade study between two LP system designs that differ in speed has been performed, which is presented in section V below.

To minimize the weight and the length of the turbine, it is desirable to reduce the number of stages and to minimize the blade row spacing. However, the turbine must also have acceptable values of blade turning and moderate stage loading values. For the Silent Aircraft engine design, the blade turning was limited to 130 degrees. Higher values would give rise to difficulties in the detailed blade design and a large increase of the endwall losses would be expected thus significantly reducing LP turbine efficiency. Also, for the Silent Aircraft design, a limit of 0.5 was imposed on the ratio of the axial chord to the spacing between blade rows was used to avoid high levels of potential field interaction noise.

To minimize the tonal noise from the low Pressure turbine a large number of rotor blades is beneficial because high frequencies are easier to attenuate. Furthermore, an increase of the number of blades, both vanes and rotors, allows a decrease in the length of the blade axial chord. This is desirable because it reduces the length and the weight of the turbine. This weight reduction is primarily the result of the reduction of the assembly rotor weight, mainly the disc, which is the heaviest component of the Low Pressure turbine. However, the number of blades and the length of the blade axial chord are limited. The maximum number of blades is constrained by three factors. Firstly, the level of centrifugal stresses on the hub is limited by material properties. A maximum of 350 MPa has been applied. Secondly, the blade numbers are limited by the flow blockage on the hub and a maximum ratio of the thickness to the axial chord at the hub of 2.8 was therefore applied. Thirdly, the blade numbers are limited by manufacturing considerations and a minimum hub section thickness of 4 mm has been considered. For the Silent Aircraft design, the ratio of blade numbers between adjacent blade rows has also been chosen, where possible, to avoid tonal noise scattering at low frequencies. This is demonstrated in section V. The minimum length of the blade axial chord is also constrained. This is limited by the value of the Reynolds number and the blade aspect ratio. A minimum value for the Reynolds number of 45000 was applied to ensure good aerodynamic performance. For the blade aspect ratio, a maximum value of 9 was applied to avoid blade vibrations and flutter.

Based on the considerations presented above, the Low Pressure turbine can be designed and its weight estimated. The results of applying these design rules are demonstrated within section V.

B. Transmission system design

As the propulsion system for the Silent Aircraft uses one engine core with one Low Pressure (LP) turbine to drive three fans (figure 2a), a transmission system is needed to transmit and divide the power from the LP turbine into the fans. Furthermore, a cooling system that will provide the lubrication and the cooling of the transmission system is also required. This is a key component of the Silent Aircraft engine as cooling and lubrication are essential for the continuous engine performance.

The aim was to design a transmission system for the Silent Aircraft engine that has low weight and gives rise to low levels of vibration and noise. Furthermore, a highly efficient transmission system is also required so that the amount of heat generated is minimized and the size and the weight of the cooling system can be reduced. The system also needs to be highly reliable, which demands a system with minimal complexity.

There are several different possible ways to transmit the power between two rotating components. For the Silent Aircraft design, gears were used because of their proven high reliability and low power losses. The main elements of the transmission system that were considered are the gears, the shafts and the bearings.

The design of the transmission system has been based on AGMA (American Gear Manufacturer Association) specifications (references [19] and [20]) and reference [21], using current limit for aerospace applications. The transmission system was designed at maximum power take-off conditions because this is the point in the flight envelope when the highest loads are encountered and for 30000 hours of operation at cruise conditions. For the estimation of the loads applied on the gears, poor load sharing has been assumed in order to include a safety factor into the design. The gears are therefore designed to be able to run under loads 10% higher than those expected at take-off conditions.

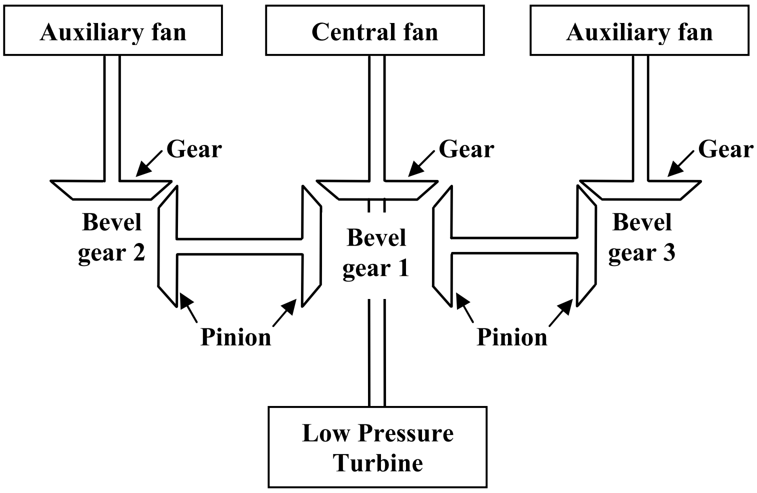


Figure 10. Schematic of the transmission system

Figure 10 shows a schematic of the transmission system. Bevel gears are needed so that the transmission system can redirect the power from the Low Pressure Turbine through 90 degrees to the auxiliary fans. The proposed transmission system consists of three bevel gears. The bevel gear 1 divides the power between the three fans and consists of one gear and two pinions. The other two bevel gears (bevel gear 2 and 3) are identical and are used together with the bevel gear 1 to redirect the power to the auxiliary fans. They each consist of one gear and one pinion. The Low Pressure turbine in a modern turbofan rotates at a high-speed and as a consequence, high values of pitch line velocities on the gears are expected. The pitch line velocity is the peripheral speed of the gear at the pitch diameter (figure 11). Spiral bevel gears have been chosen for the design because they allow the high pitch lines velocities that are encountered in this application. Furthermore, spiral bevel gears have higher contact ratios between mating teeth so they have a smoother operation leading to lower levels of noise and vibration. Spiral bevel gears due to their tooth form (figure 11) transmit axial as well as radial loads to the bearings. Therefore, a ball bearing and a roller bearing were chosen for each of the gears and pinions on the transmission system, where the ball bearing is used to handle the axial load generated by the spiral bevel gear. The bearings are straddle mounted as shown in figure 12a. This type of layout is more beneficial than an overhung one (figure 12b) as it gives a better gear support reducing misalignment, vibration and noise that are requirements for the Silent Aircraft transmission system.

For the design of the gears the values of some geometry parameters such as, the gear ratio, the shaft angle, the pitch angle, the helix angle, the pressure angle, the number of teeth, the gear diameter and the face width, the gear material and the heat treatment need to be specified. The bevel gear 1 behind the LP turbine has a gear ratio of 1. The gear ratio of the bevel gears 2 and 3 changes depending on the rotational speed of the Low Pressure turbine. Figure 11 shows some geometry parameters of a typical bevel gear. The shaft angle is 90 degrees for the three bevel gears, the pitch angle depends on the gear ratio and the values of the helix angle and the pressure angle are 35 degrees and 20 degrees respectively for the three bevel gears. Those are considered standard for aerospace applications. The number of teeth, the gear diameter and the face width together with the gear material and the heat treatment required are determined by the gearbox rating. In addition, the design of the gear has to satisfy that the design rule that the ratio between the face width and the cone distance is lower than 0.3. The gears are enclosed to diminish any vibrations and noise, and the housing material is aluminium for low-weight.

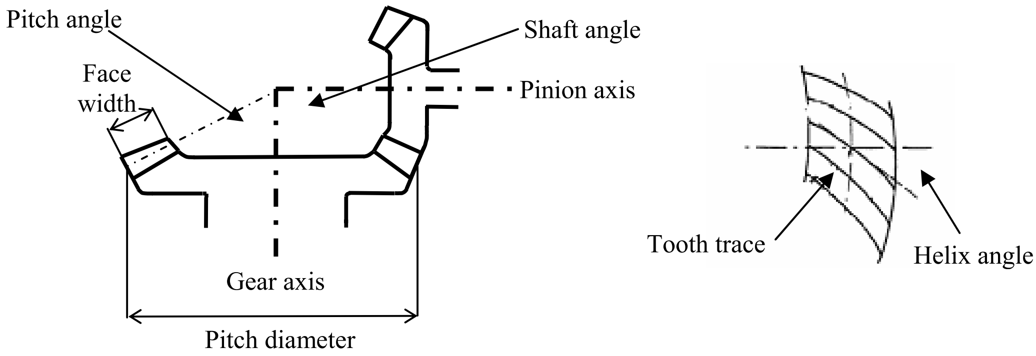


Figure 11. Bevel gear nomenclature

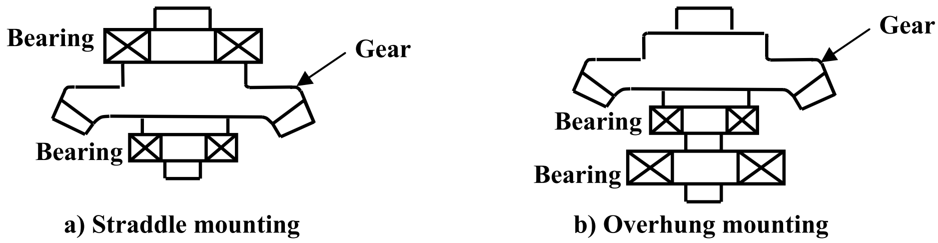


Figure 12. Bearing mounting

The gearbox rating is based on three considerations, i.e., bending stress, compressive stress and scuffing resistance. The gear has to be designed to ensure that it can withstand the bending and compressive stresses at take off conditions. The bending stress and the compressive stress ratings have been performed following AGMA specifications, [22]. Scuffing in a gear takes place when there is a failure in the lubrication that causes metal to metal contact. As a consequence, part of the material on the tooth surface is removed. For this design, a method called the flash temperature index has been used to predict the probability of scuffing. In this method, a critical temperature, i.e., the flash temperature index is calculated. This temperature depends on the operating conditions, the gear material that was chosen to be VASCO-X2M, which is standard for aerospace applications, the gear size and the surface finish (an AGMA Quality Class 13 with 10 micro inches of surface finish was selected). For the Silent Aircraft gear system design this temperature was compared with the results of an aerospace industry survey correlating the probability of scuffing with the flash temperature index. It was concluded that no problem of scuffing should be encountered for the gears.

The torque from the LP turbine into the gears and from the gears to the fans is transmitted by the shafts. The shafts are sized based on the maximum torque and the maximum momentum applied during take-off conditions. The dynamic behaviour of the shafts must also be considered in order to ensure that excess whirling is avoided.

C. Cooling system design

The transmission system generates a large amount of heat that needs to be removed by the cooling system. The heat is transferred to the cooling oil by conduction and then removed from the system by means of a heat exchanger. The estimation of the power loss at take-off condition is a measure of the highest amount of heat generated by the transmission system and this includes the loss produced between the gears and within the bearings and the windage loss. The power loss in the gears has been calculated based on reference [21]. This depends on the working conditions, the gear geometry, the gear ratio and the friction coefficient, which was estimated to be 0.06 according to reference [23]. The power loss on the bearings was estimated based on AGMA specifications, reference [24] and depends on the working conditions, the bearing size and the bearings friction coefficient, which was 0.0015 for the ball bearings and 0.0011 for the roller bearings. The power loss due to windage depends on the gear rotational speed and the gear geometry and can be estimated based on reference [21].

The Silent Aircraft cooling system design is based on that of a conventional modern turbofan. Figure 13 shows a schematic of the cooling system based on reference [25]. The system consists of three parts, i.e., the lubrication system, the scavenge system and the vent and de-aerator system. The lubrication system supplies the oil to the gears and the bearings. The scavenge system returns the excess of oil from the gears and the bearings to the oil tank. The vent and de-aerator system prevents the oil leakage and the oil foaming. There are other elements on the system such as the chip detectors that are located on the gears and the scavenge pipe work, the oil tank, the heat exchanger and the pipe wok.

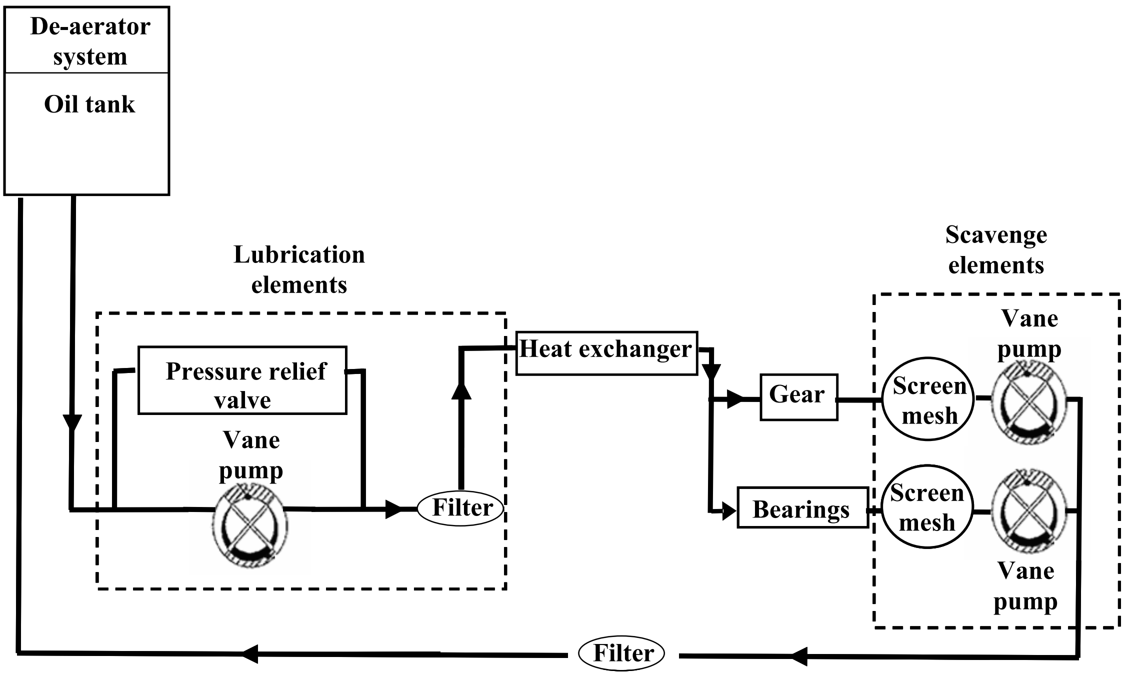


Figure 13. Schematic representation of the cooling system

The main elements of the cooling system that have been considered for the Silent Aircraft preliminary design are the oil- MIL-L-23699 that it is standard in aerospace applications, the oil tank, the oil pumps, the scavenge pumps and the heat exchanger. These are the heaviest elements of the system and therefore have the greatest influence on the overall engine characteristics. The method used for the weight estimation of these elements is included in section V.

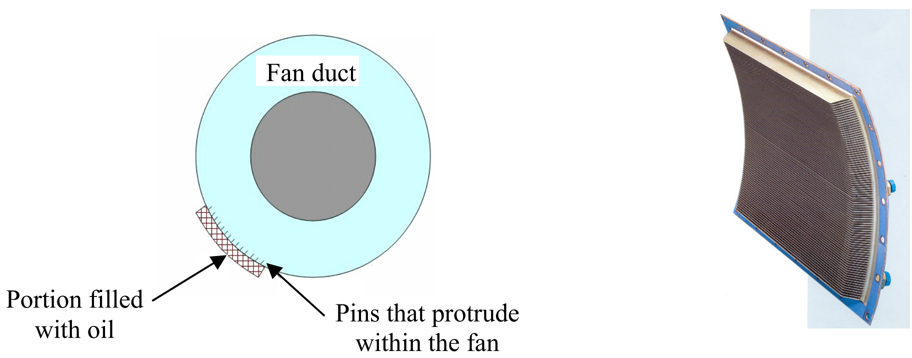


Figure 14: a) Schematic of the heat exchanger b) Heat exchanger sector taken from reference [26]

Two oil pumps were chosen that work simultaneously to provide the amount of oil required during take-off. However, in order to have some redundancy in the system, only one of them is needed during cruise conditions. Thus, in the case of failure of a single oil pump the other one can operate and still provide adequate oil flow rate. There is a scavenge pump for each of the elements of a gearbox, i.e., the gear and the pinion. There is also a scavenge pump per each of the bearings. The pumps chosen are of the vane type because these are more efficient for operation at altitude. The heat exchanger assumed for the design is the surface type and a schematic is shown in figure 14a. This could be located around the fan duct with fins or pins that protrude within the fan duct to increase the surface contact. Figure 14b is a picture from reference [26] of a segment of a surface type heat exchanger used nowadays in aerospace applications. Note that for the Silent Aircraft this type of design would need to be compatible with the liners that are located all around the fan duct to absorb the rearward engine noise. A more detailed analysis on this was not performed for the preliminary design.

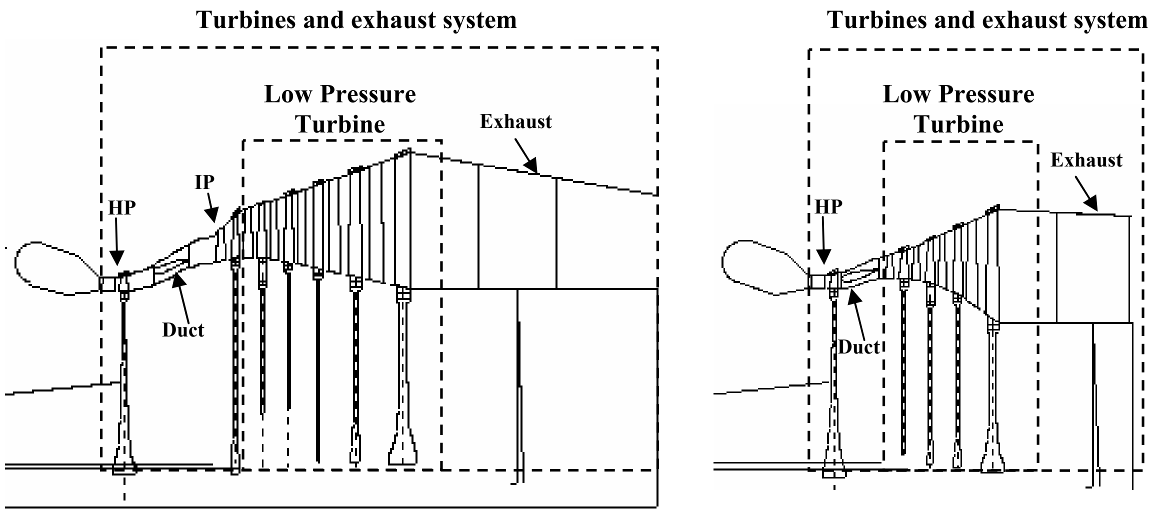
V. Comparison of engine designs with high-speed and low-speed LP turbines

A trade study for two different engine designs was carried out. One design has a low-speed LP turbine running at the same speed as the fan. This design is a three spool engine. The other design has a high-speed LP turbine with a rotational speed that is 1.45 times faster than that of the fans. This is a two spool geared engine configuration. For both design options, the LP turbine, the transmission system and the cooling system were designed and compared in terms of their performance, weight, length, reliability and noise. The analysis and comparison of the two designs is presented below.

A. Comparison of the LP turbines

Increasing the turbine rotational speed for the high-speed design gives rise to a reduction of the stage average loading and the blade turning needed to generate the turbine work. As a consequence, the number of stages can be reduced. Figure 15 shows a cross sectional view of the rear part of the engine for the high-speed and the low-speed design. For the high-speed option, figure 15b shows one stage of High Pressure (HP) turbine, the inter turbine duct, four stages of LP turbine and the turbine exhaust. For the low-speed design, figure 15a shows one stage of HP turbine, the inter turbine duct, one stage of Intermediate Pressure (IP) turbine, five stages of LP turbine and the turbine exhaust. For the high-speed design, the LP turbine is 33% shorter and 58% lighter and the whole system of turbines and exhaust is 43% shorter and 51% lighter when compared with the low-speed design.

Figure 16a shows the values of the midspan exit Mach number for the vanes (V) and rotors (R) for both designs. Figure 16b shows the rotor hub centrifugal stresses. The high-speed Low Pressure turbine has significantly higher values for these two parameters. This suggests that the high-speed design is more challenging in terms of aerodynamic and mechanical considerations.

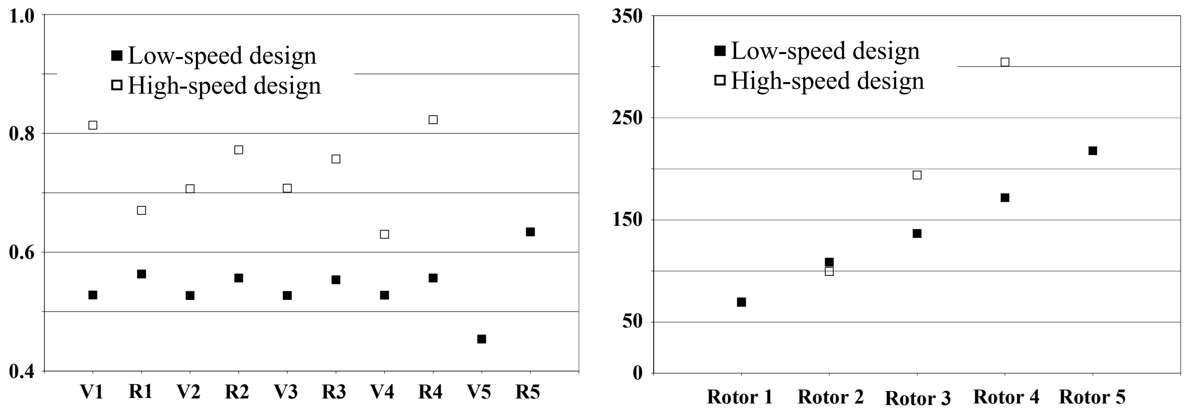


a) Low-speed design   b) High-speed design

Figure 15. Cross sectional view of the turbines and the exhaust system for the low-speed and high-speed designs

The turbine noise is one of the major contributors to the engine noise. Four different noise sources have been analyzed for the turbine noise. These are the tonal noise into the blade passing frequencies, the tonal noise scattered at low frequencies, the haystack and the broadband noise. For the noise analysis, Roll Royce and ITP proprietary software has been used. The frequencies of the tonal noise into the blade passing frequencies are a function of the turbine rotational speed and the number of rotors in the blade rows. For both designs, the first harmonic of the blade passing frequency is out of the audible range at sideline, flyover and approach. Therefore, tonal noise from the blade passing frequency was not considered a problem for either of the designs. The tonal noise scattered at low frequencies is due to the interaction of generated modes and the rotors located downstream. The number off ratio between adjacent blade rows for both designs has been selected when possible around unity to avoid this source noise. Table 2 presents the number of blades for all the rows for the two designs.

Haystack noise occurs due to the propagation of waves through the turbulent shear layer of the jet. This yields to the individual tones to become broadened and therefore, noise into lower frequencies can happen. This source noise was not found to be a problem for any of the two designs. The broadband noise is the most difficult source to quantify for the Low Pressure turbine. This is one of the biggest challenges for the engine design of the Silent Aircraft. The correlations used were based on experimental and analytical studies on Low Pressure turbines with rotational speeds significantly lower than that of the two Low Pressure turbine designs that are being analyzed. Table 3 shows the difference in dBA between the two designs using the existing correlations. The low-speed design has lower levels of broadband noise.



a) Exit midspan Mach number **b) Hub centrifugal stress**

Figure 16. Exit midspan Mach number and hub centrifugal stress for the low-speed and high-speed designs

	Number of blades	
	Low-speed design	**High-speed design**
Vane 1	232	210
Rotor 1	230	210
Vane 2	232	210
Rotor 2	230	210
Vane 3	222	200
Rotor 3	220	200
Vane 4	214	178
Rotor 4	214	176
Vane	198	-
Rotor 5	190	-

Table 2. Number off for the blade rows of the low-speed and high-speed designs

	Low-speed design	High-speed design
Sideline (dBA)	Reference	-5.8
Flyover (dBA)	Reference	-6.8
Approach (dBA)	Reference	-8.0

Table 3. Broadband noise levels of the high-speed design relative to the low-speed design

B. Comparison of the transmission system

The transmission system differs significantly between the two designs. Figure 17 shows a schematic of the transmission systems. For the low-speed Low Pressure turbine, the transmission system consists of three bevel gears as explained earlier, all of them with a gear ratio of 1. The high-speed design requires the reduction of the rotational speed between the Low Pressure turbine and the fans. This consists of three bevel gears and a planetary gearbox. The bevel gear 1 has a gear ratio of 1, the bevel gears 2 and 3 has a gear ratio of 1.45. The planetary gearbox is required to get the speed reduction for the central fan so that it also has an overall gear ratio of 1.45. Table 4 shows the working conditions of the gears for both designs and Table 5 shows relevant design parameters. For the case of the high-speed design, the planetary gear is of the star type where the planets are held stationery. This gear was chosen because a low gear ratio is required and it has the advantage of having lower loads on the planets. The lubrication is also easier because the planets are not moving. The most important parameters of the star planetary gear are shown in table 6. The shafts are transmitting a different amount of torque for the two designs so the shafts diameters differ (table 7). The need of another gearbox for the high-speed design decreases the mechanical efficiency of the system by 0.77%, increases the amount of oil needed for cooling and lubrication by 47%, increases the transmission system weight by 8.5% and the cooling system by 2.6% and reduces the reliability of the system.

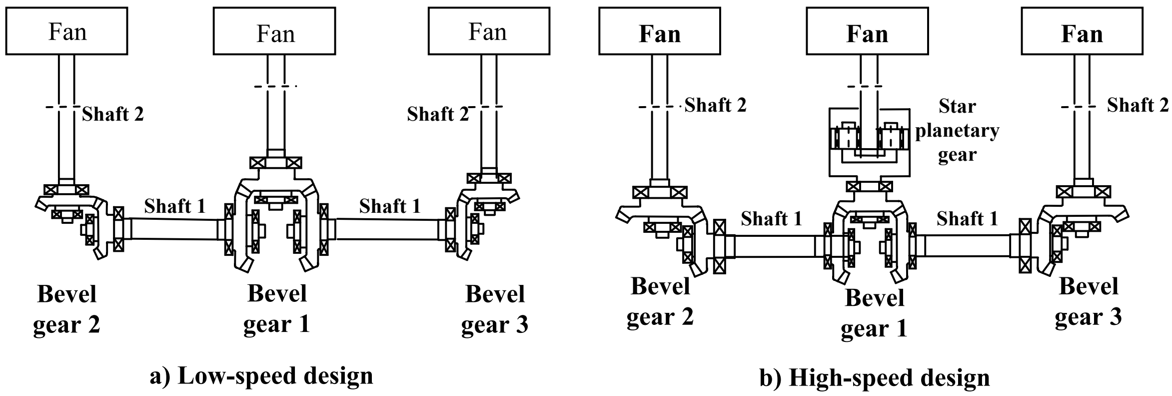


Figure 17. Schematic of the transmission system

	Low-speed design		High-speed design	
	Gear 1	Gear 2 and 3	Gear 1	Gear 2 and 3
Power transmitted (hp)	34866	13946	34866	13946
Rotational speed (rpm)	5480	5480	8000	5480

Table 4. Working conditions for the transmission system design

	Low-speed design		High-speed design	
	Gear 1	**Gear 2 and 3**	**Gear 1**	**Gear 2 and 3**
Pitch angle gear (°)	45	45	45	55.4
Pitch angle pinion (°)	45	45	45	34.6
Pitch diameter gear (in)	16.80	12.28	14.32	18.72
Pitch diameter pinion (in)	16.80	12.28	14.32	12.91
Face width (in)	3.55	2.60	3.03	2.35
Number of teeth gear	60	60	50	87
Number of teeth pinion	60	60	50	60
Pitch line velocity (fpm)	23500	17900	30000	19000

Table 5. Geometry parameters of the bevel gears

	Star Planetary gear		
Gear ratio	1.45		
Number of planets	4		
	Sun	**Planet**	**Ring**
Pitch diameter (in)	14.32	3.22	20.77
Face width (in)	4.5	4.5	4.5
Number of teeth	80	18	116
Pressure angle (°)	20	20	20
Helix angle (°)	26	26	26
Pitch line velocity (fpm)	30000	30000	30000

Table 6. Geometry parameters of the star planetary gear

	Low-speed design	High-speed design
Shaft 1 diameter (in)	6.76	7.60
Shaft 2 diameter (in)	5.76	5.88

Table 7. Geometry parameters of the shafts of the transmission system

C. Comparison of the Cooling system

The heat generated and the amount of oil needed for cooling of the high-speed design are greater than for the low-speed design. Therefore, the cooling system is more complex in the high-speed configuration. The size of the oil tank and the heat exchanger increase for the high-speed design and there are 8 more scavenge pumps because of the star planetary gear. As a consequence, the weight of the cooling system for the high-speed design is considerably higher. Table 8 shows the breakdown of the bare engine weight for both designs normalized by the total bare engine weight of the low-speed design. Although the bare engine when the transmission system is not considered is 16.1% heavier for the low-speed design, the transmission system and the cooling system are much lighter. As a consequence, the low-speed design total weight is only 5% greater than the high-speed design.

	Low-speed design	High-speed design
Bare engine with no transmission	79.6	63.5
Transmission system	14.8	23.3
System for distribution of oil	5.6	8.2
TOTAL BARE ENGINE WEIGHT	100.0	95.0

Table 8. Weight comparison between the low-speed and the high-speed designs

D. Final design choice and overall engine weight

The final choice for the Silent Aircraft was the low-speed LP turbine configuration. This design has a less complex transmission and oil system and the noise levels on the ground are lower. This is achieved with a small impact on the engine weight (~5%) and a minor increase in the bare engine length.

Table 9 shows the breakdown of the engine weight for the final design. The bare engine weight estimation includes the weight of the three fans with the containment made of Kevlar, the splitter, the Intermediate compressor, the axial-radial High compressor, the combustion chamber with the fire walls, the high Pressure, Intermediate pressure and Low Pressure turbines, the turbine exhaust duct and the control system and accessories.

The transmission system weight includes the gears with the housings, the shafts and the bearings. The gearbox weight was estimated from the geometry of the system and the material densities. The weight of the housing was estimated to be 40% of the total gearbox weight. The shaft weight was determined from the shaft diameter and length and the shaft material, which was assumed to be steel. The bearing weight was estimated from a commercial bearing used in aerospace applications with a similar load carrying capability.

The weight of the cooling system includes the oil, the tank, the scavenge pumps, the oil pumps and the heat exchanger. The oil weight is estimated by what would be needed for the lubrication and cooling of the transmission system during 15 seconds of continuous running at take-off conditions. The weight of the oil tank was estimated based on the capacity and weight of a typical tank used for the oil system of modern turbofans with an extra 20% of volume added to account for air space and heat expansion. The weight of the pumps was estimated based on a commercial pump used in aerospace applications. The weight of the heat exchanger is based on values for the commercial heat exchanger shown in figure 14b and scaled with the rate of heat removal required.

The engine installation weight includes the thrust reverser, the variable area nozzle, thee exhaust ducting and actuation, the pylon and engine support structure, the inlet duct with the acoustic liners, the external cowling and the splitters. This weight estimation was based on conservative estimations and reference [27].

WEIGHT (ONE CORE+3 FANS)		
Bare engine with no transmission	5226	43.4%
Transmission system	970	8%
Cooling system	371	3.1%
Engine installation	5492	45.6%
TOTAL ENGINE WEIGHT	12059	100%

Table 9. Engine weight breakdown

VI. Summary and Conclusions

1. The Silent Aircraft Initiative goal was to design an aircraft that is imperceptible above background noise outside the airport boundary and that is also economically competitive with conventional aircraft of the future. To achieve these ambitious targets, a multi-fan embedded turbofan engine with boundary layer ingestion has been proposed. This engine configurations has potential noise, fuel consumption and packaging advantages but it introduces big challenges for the design of the low Pressure spool.
2. A low-pressure system design for the Silent Aircraft engine has been developed that includes: i) a high performance, low-noise fan optimised for use with a variable area exhaust nozzle ii) a low-pressure turbine designed for low-weight that is expected to generate minimal tonal noise iii) a low-complexity transmission and cooling system designed to aerospace standards.
3. For operation with a variable exhaust nozzle the rotor blade of an engine fan must be designed from the outset for "off-design" operating conditions. This includes designing for suitable variations in the mechanical untwist, as well as optimising for high efficiency at cruise and low-noise at take-off and approach.
4. The range of incidence angles perceived by the outlet guide vanes of a fan system is considerably greater when operated with a variable area exhaust nozzle. The vanes need to be carefully designed with the appropriate incidence tolerance in order to maintain good performance and low-noise
5. An embedded engine configuration gives rise to additional duct losses and inlet flow non-uniformity that make the fan design more challenging. These challenges include additional aero-mechanical vibration, reduced stability and non-optimal performance. Further research is required to develop durable, low pressure ratio fan systems for operation within embedded engines.

6. A multiple-fan configuration increases the Low Pressure turbine rotational speed to a more optimum value when compared with a conventional design even without any gear reduction ratio between the LP turbine and the fans. Thus a Low Pressure turbine configuration with minimal tonal noise has been possible by optimizing the number of rotor blades and the aerofoil number ratios between adjacent blade rows. The broadband noise is the most difficult source to quantify for the Low Pressure turbine and further work is required in this field.
7. A multiple-fan engine configuration introduces many complex design implications for the low-pressure transmission system. A design process has been proposed for both the transmission system and the cooling system. The oil system design is critical to the transmission system because it provides cooling and lubrication, both of which are essential for continuous engine performance.
8. A comparison of two low-pressure system designs with different speeds of low-pressure turbine rotation rate shows that the details of the transmission and cooling systems are key in determining the overall weight and viability. A higher speed design at first seems to offer advantages in terms of weight (reduced number of stages). However, a higher speed design demands a more complex transmission arrangement and greater oil flow rates. The final design chosen for the Silent Aircraft engine has the low pressure turbine and all three fans rotating at the same speed.
9. There are many challenges to the further development of transmission and cooling systems for multiple-fan engine configurations. In order to be viable, all the components have to be proven to have comparable reliability to current commercial aero-engines as well as shown to be invulnerable to failure. Developing the details of a dependable cooling system with acceptable heat-exchangers will be particularly difficult and more work is required in this field.

Acknowledgments

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