SSBBR-X: Candidate Engine for Concorde

Cannon Buechley[1], Greyson Farris[2], Benjamin Ingling[3], Joshua Kincaid[4], and Chase Wallen[5]

The University of Tennessee, Knoxville, TN 37996, USA

Advances in each component of a low bypass ratio turbofan were considered to improve the performance for Concorde's mission. The bypass fan will feature a multi-stage design intended to increase pressure ratio and exit velocity of the bypass air to speeds exceeding cruising speed. The multi-stage fan will allow a variable bypass design to be utilized for optimization in different flight regimes. Preliminary research suggests that using a bypass ratio of around 1.0 will be feasible to implement in the design. The exit nozzle will be a variable converging-diverging nozzle to allow for necessary mass flow at different velocities, which is now an industry standard. Compressors of the past have been high in weight with limited pressure ratios, but by using new technologies currently available and new materials proposed to be available, compressors can be lighter and have higher-pressure ratios per stage. Using a bladed disk (or blisk) alone can lead to a weight savings up to 30% and new materials have been proven to have higher operating temperatures, allowing for higher efficiency and thrust for the entire system. New research has proposed a redesign of the burner-turbine system which would reduce the turbine inlet temperatures as well as increase efficiency and thrust. This new system is the Inter-Turbine Burner (ITB) which adds a second combustion chamber in between the high and low-pressure turbine stages. The effect of an ITB is to burn the fuel from the first burner and use all the remaining oxygen in the system. The ITB system eliminates the need for cooling channels in the turbine blades, which subsequently eliminates the need for a cooling system and bleed valves. This reduces the high complexity and weight of turbines while simultaneously reducing the cost of manufacturing traditional blades. Traditional blades are made of superalloys that are manufactured by casting a one direction crystal structure in the metal. New methods for manufacturing blades have been proposed, consisting of using additive manufacturing and advancements in composite materials. Using composites and additive manufacturing, turbine blades can be made with a high tolerance to temperature, which will decrease the TSFC as well as cost. These systems were tested using parametric cycle analysis implemented in MATLAB, VuCalc, GasTurb13, and AxSTREAM to effectively compare their impact on the entire engine and gauge whether any combination of the new component technologies will be ready for a 2028 entry-into-service date. This analysis is a precursor to a different AIAA design competition with the purpose of designing engines to replace those on the Concorde aircraft. The results of this engine will be compared to Concorde's original Olympus 593 engines to determine if better performance was achieved.

I. Introduction

A. Background

The Concorde had the potential to revolutionize the commercial aerospace field. Flying at a cruising speed more than Mach 2.0, it could make trans-Atlantic flights in as little as 3.5 hours – what in today's aviation takes 6-8 hours to traverse [2]. It was thought to usher in "aviation's second wave," [3] focused on supersonic commercial travel.

[1] Undergraduate Student, Mechanical Aerospace and Biomedical Engineering, Student Member
[2] Undergraduate Student, Mechanical Aerospace and Biomedical Engineering, Student Member
[3] Undergraduate Student, Mechanical Aerospace and Biomedical Engineering, Student Member
[4] Undergraduate Student, Mechanical Aerospace and Biomedical Engineering, Student Member
[5] Undergraduate Student, Mechanical Aerospace and Biomedical Engineering, Student Member

However, the machine was too ambitious for technology of the time. To achieve its speeds, it had to burn upwards of 6770 gallons of fuel each hour. The increasing oil prices during the 1970's as well as concern with the environmental footprint of the byproducts of burning this much fuel with the technology available at the time aided in its downfall. Additionally, due to FAA regulations to prevent sonic booms overland, the only feasible routes for the aircraft were over the Atlantic. Recently, however, with improvements in aeroacoustics, a push for the FAA to revise these regulations has been addressed [2]. Improvements in propulsion technology and materials in the last half century makes re-designing the Concorde's powerplant, the Rolls-Royce/Snecma Olympus 593, a possible and interesting endeavor as interest in commercially viable supersonic flight once again increases.

B. Proposal Guide

Equipping a supersonic aircraft with efficient turbofan engine has many difficulties which will all be addressed in this proposal. Firstly, for the internal components of the turbofan engine to operate efficiently at supersonic speeds, the incoming air must be slowed by passing through an oblique shock to less than Mach 1 (roughly Mach 0.6). Then for the bypass flow to provide any thrust, the exiting air must be traveling faster than the travel speed of the aircraft. Luckily, because of principles of compressible flow, one only needs to compress the airflow to a high enough pressure ratio, and then, as the air expand through a diverging nozzle, it will accelerate to a more appropriate velocity. For instance, if one wants the exiting air to be traveling at Mach 2.2, the pressure ratio of the compressed air to the ambient air must be 10.7. How would one achieve this pressure ratio with one fan blade? The answer is simple: do not use one fan blade use many. The initial design plan for this engine is to widen the low-pressure compressor to fit into the full bypass nacelle to create a great enough pressure ratio to create this supersonic bypass flow. This however creates another problem; having the low stage compressor draw this much energy out of the flow has the potential to "overdraw the bank." To prevent this, another extreme change will be added. This paper will also propose the addition of inter-turbine burners or ITBs. An ITB is just that, a secondary and/or tertiary burner placed between turbine stages. With this system, nearly all the core streams available oxygen can be used, maximizing the energy added to the core stream, and with advanced ceramic turbine materials, this system could remove the need for turbine blade cooling which would reduce the price of these engines by removing manufacturing complexity.

II. Case Studies

A. Inlet

The design utilized the same inlet geometry as the original Concorde, notably a variable two-dimensional ramp inlet. Likewise, the entranceway was the same as that of the nacelle for the Olympus 593 engine. While the project specified that the original Concorde inlet should be utilized and the turbomachinery downstream should be optimized for existing inlet conditions, the pre-fan Mach number of 0.549 was deemed to be far below what could be reasonably implemented today. Re-designing engine inlet geometry while maintaining the variable-geometry ramp aspect would also allow for greater pressure recovery than the specified method using military specification MIL-E-5007, which used technology from the infancy of supersonic flight [4]. Therefore, the inlet geometry was re-designed to provide a pre-fan Mach number of 0.7, a figure commonly cited as ideal to maximize mass flow and compressor pressure ratio while remaining below transonic at compressor blade tips [5].

B. Fan

i. Fan Blade

Fan design consisted of a hybrid low-pressure compressor and bypass fan. The use of three blade stages in this part of the engine was decided upon as a base number to provide a fan pressure ratio high enough to allow for acceleration to an appropriate bypass exit velocity. Test cases will be analyzed to determine optimum configuration and the optimum fan pressure ratio, as it is a function of multiple things including specific thrust and loosely bypass ratio and can therefore be optimized for this design [6]. Likewise, although expected to be optimized at a value of around 1, tests will be run to determine optimum bypass ratio.

ii. Bypass

To increase the efficacy of the engine across its multiple-flight-regime operating range, new technology for a variable bypass turbofan developed by the United States Air Force was selected for implementation on the powerplant. The Air Force design features a bypass fan mated to the low-pressure turbine shaft with an intermediary clutch to allow for disengagement and the integration of varying pitch fan blades to allow for pivoting to a "pseudo-inlet guide vein" position when decoupled, effectively allowing the design to shift from a low-bypass turbofan to a turbojet across the flight envelope [7]. In the Concorde redesign, this varying bypass ratio design will allow for higher efficiency at takeoff while not sacrificing the ability to reach the Concorde's design cruising speed of about Mach 2.0. A clutch located between each fan stage to enable each individually to be shut down as needed will optimize the transition even further. Additionally, the new design utilized mixed bypass-core exhaust upstream of the nozzle due to its improvements of 2-3% in thrust generation and specific fuel consumption over separated exhaust over a range of bypass ratios [8].

C. Nozzle

To maximize overall engine performance over the entire operating range, a fully variable, axisymmetric, converging-diverging nozzle was chosen for implementation. This design has been utilized for decades on aircraft operating over a range of flight regimes as the throat area can be altered to allow more air flow through the turbomachinery as needed. The effect on nozzle efficiency at varying nozzle pressure ratios (which are a function of speed and altitude) can be seen in **Fig 1**; choosing a design which allows for the throat area to alter allowed for the optimization of this operating range. The divergent section allowed for the nozzle pressure ratio to decrease as air is expanded in the section. However, it was not of adequate size to ensure complete expansion; the subsequent loss in thrust potential described in **Fig 2** was deemed acceptable in the design to stay within nacelle bounds and thus prevent increased drag [9]. Further analysis will be done to determine optimal throat area at the design point, nozzle length and area ratio, and lost thrust potential.

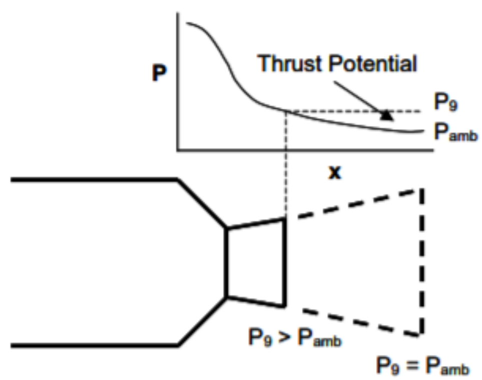

Fig. 1 **Effect of under-expansion on thrust potential [7]**

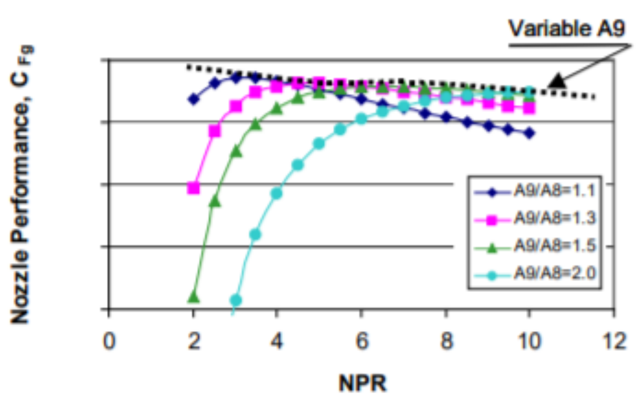

Fig. 2 **Effect of nozzle pressure ratio on nozzle performance by nozzle are ratio [7]**

D. Compressor

i. Blade Type

Compressor sections designed and built in the 1960's, such as Concorde's Olympus 593, were heavy and not very efficient with stages to achieve a desirable pressure ratio. With new technologies available in the machining processes and capabilities of new materials, compressor sections can be lighter and more efficient. A newer technology that has emerged in commercial use is the bladed disk (or blisk) in the compressor section of the engine. A blisk is a blade integrated disk that are highly complex components comprising both rotor disk and blades in one integral part, which can be seen in **Fig 3** [10]. Some advantages to using blisk technology is a reduction in the leakage of compressed air and a lighter weight. Conventional compressor blade rows with axial grooves have some leakage through the root section of the blade. Using blisk technologies allow for this leakage to be eliminated since the entire stage is a single block of machined material [11]. Blisk technology has also been found to be around 20%-30% lighter resulting from the elimination of blade roots and disk lungs. One of the more recognized downsides of using blisk technology is the laborious, and expensive manufacturing and repair processes [12]. If one of the blades gets damaged, then the entire blisk must be removed and replaced.

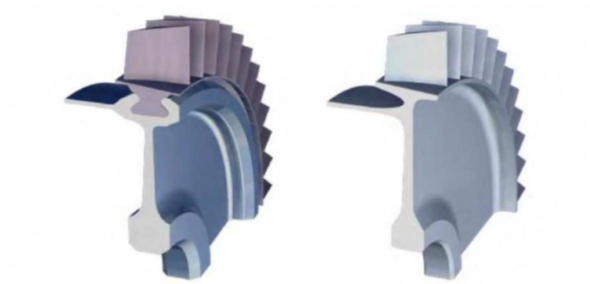

Fig. 3 Visual difference between a conventional compressor stage (left) and a blisk (right) [12]

ii. Materials

Another advancing technology is the materials being used in the compressor. Advances in the use of Titanium within the engine and advances in the ability to experiment with titanium alloys have driven the weight down in materials. A newer titanium alloy, Ti48Al2Cr2Nb, has been shown to have similar strength characteristics as conventional nickel-based alloys, but are significantly less dense. A draw back with using these complex intermetallic is their brittleness. While intermetallic materials have high temperature tolerances and strength characteristics, they become brittle and are more susceptible to damage, causing them to have to be replaced more often [13].

iii. Stages

Future analysis into the compressor stage using GasTurb13 will occur to determine the number of stages needed to get a desired compression ratio. This analysis will lead to a weight estimate based on the material used and the number of stages required. The proposal right now would be to have six stages with a blisk design, the first three with a lower temperature dependent material, followed by three blisk design stages made from a more temperature capable material, such as Ti48Al2Cr2Nb. Most modern military supersonic engines have around seven stages of High-Pressure Compressor stages [14]. The use of a blisk and newer materials could help to eliminate the need for another stage and reduce the weight even more in the engine.

E. Burner

i. Fuel

Considering that a main goal for the redesigned Concorde engine lies in improving the efficiency of the plane, much of the issue comes from the amount of fuel that is consumed. The original Olympus 593 engine failed at using fuel effectively since much of the plane's weight was attributed to the fuel, resulting in higher operational costs which only a portion of the population was willing to pay. This, however, was not the sole reason the Concorde ultimately failed and is contributed to a myriad of other problems. For the jet to reach its target speed and altitude, afterburners were a necessary component. It is hopeful for this design experiment to eliminate the need for reheat at takeoff which will drastically improve the Concorde's fuel consumption rate. Industry standard efficiency in the combustion chamber has risen to the mid-to-high 90%. Some inefficiencies lie in the fact that not all the fuel that is being mixed with the high-pressure air that is leaving the compressor is being burned. This is where an inter-stage turbine burner (ITB) would be used and is visualized in **Fig 4**.

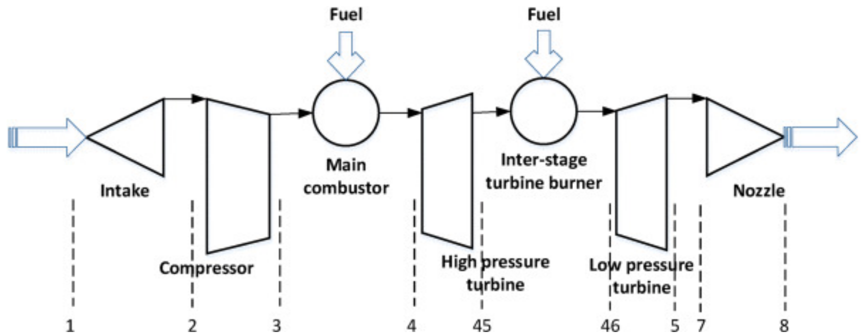

Fig. 4 Inter-Turbine Burner Configuration [15].

ii. Combustor Design

The ITB is used to re-light the already burned air from the main combustor to ensure that the fuel-oxygen mixture has been completely burned. This, in turn, reduces the carbon emissions dispersed into the air according to Robert Jakubowski [16]. Currently, the turbine inlet temperatures hover around 1800K with the limiting factors being the materials used and the cooling system in place. Since a second combustion chamber is being added, not as much heat is needed to burn the air in the main combustion stage. Implementing the ITB allows for the inlet temperature of the low and high-pressure turbines to be reduced to 1300K [16]. This reduction allows for the combustion chamber and turbine blades to be made from more obtainable materials by current or near-future standards. A ceramic coating placed on the lining of the annular combustion chamber combined with a silicon-based CMC turbine blades will easily withstand these temperatures resulting in effective power [17]. Feijia Yin and Arvind Gangoli Rao explain that through these temperature reductions at the end of each combustion process, high amounts of specific power can be extracted [15]. Yin and Rao also compare the power output to that of fighter aircrafts in supersonic cruise [15]. With the design mission of the Concorde and its engine being primarily during supersonic cruise, it shows very promising results with the addition of the ITB.

iii. Pressure Ratio

Early testing shows that the overall pressure ratio (OPR) across the engine can decrease by a substantial amount while also slightly increasing thrust and specific thrust. However, this came at a cost of a slightly increased specific fuel consumption rate [16]. This is the manner of new technologies, in that, most results must come with a tradeoff and the aeronautical industry is far from an exception. Considering that the inlet turbine temperatures for both low and high-pressure turbines are prospective to be much lower than the current standard, lighter materials could be used to shield from the heat. It is also ideal to completely remove the turbofan's internal cooling system. This bleed cooling system adds much more complexity to the manufacturing of the engine, so along with the weight reduction there is a cost reduction of the engine too. This weight reduction, though small, will influence the jet's specific fuel consumption.

F. Turbines

i. Blade Temperature

The turbine is the most important part of any jet engine and its design conditions set the usefulness of the entire power plant. Specifically, the engine inlet and temperature are of the highest importance because it is the hottest component of the entire system. In the past, turbine inlet temperatures reached to around 2400R (1060C) as given in [1]. This is because that was the limitation on the melting point of the turbine blades at the time, but in the current era turbine inlet temperatures can withstand temperatures up to 3560R (1700C) found in [17]. However, the limiting temperature given in [1] set a ceiling at 3150R (1476C). The melting point for most superalloys is far below what is necessary for reaching these temperatures, even with cooling systems and coatings. This can be seen in **Fig 5** that the material temperature for nickel superalloys is able to withstand only around 2260F (980C), but metal matrix composites (MMC) and ceramic matrix composites are being developed with melting points exceeding 2960R (1400C), which means cooling systems can be eliminated altogether. Furthermore, since only the first stage and guide vain experience the full maximum temperature, then as long as the first stage can withstand the temperature the rest of the turbine blades can too.

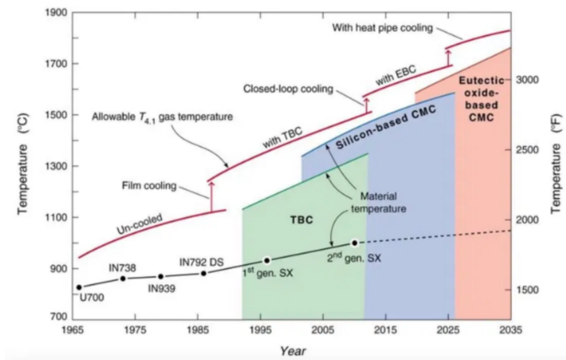

Fig. 5 Future of CMCs and Turbine Capabilities [17]

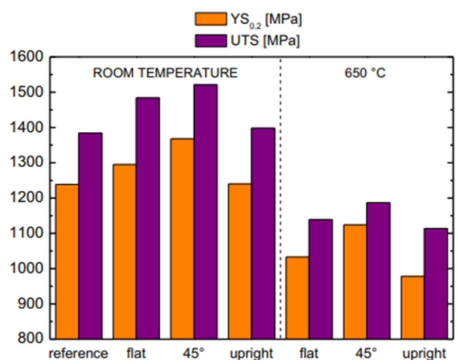

Fig. 6 IN718 Yield Strength (YS) and Ultimate Tensile Strength (UTS) at room temperature and elevated temperature [18]

ii. Manufacturing

Turbine blade manufacturing is an important because it determines the different kinds of materials that can be used as well as change the properties of the blades. The Concorde used metal casting for the manufacturing of the blades. This was done by guiding the crystal structure to grow in a specific direction to increase the strength of the blade and decrease the creep of the blade, which is common knowledge in the aerospace community. A newer technology is using additive manufacturing to manufacture turbine blades [18]. The materials that can be used in the turbine can be expanded to use ceramics and additional metals like titanium. The blades can be affected by how they are printed on the stand, specifically when it comes to the direction of the print. **Fig 6** shows the strength of the exact same material that is printed in different configurations. Those configurations are flat, upright, and 45 degrees. In both yield strength and ultimate tensile strength, the 45-degree orientation exceeded the others. Furthermore, if cooling systems are involved for a different type of engine, additive manufacturing can also include the cooling track inside of the blades instead of having to cut them out after the fact which would significantly reduce the cost. Other benefits of using additive manufacturing are being able to adjust mesh sizes of the print layers themselves which can decrease weight and cost of material being used. An entire blisk can be made with additive manufacturing, but repairability suggests such a hefty price for a component likely to fail means that creating individual blades that can be replaced with new ones regularly. Additionally, it is important that the blades can withstand the forces even at high temperatures. The drop in strength is clear between the two sides of **Fig 6**.

III. Results

As this project is currently in progress, there are little definitive numerical results for improvements. That being said: it is expected that it will be significantly more efficient while also providing more thrust than the Olympus 593 engines that the Concorde was equipped with. Shown below in **Fig 7** is a 2D rendering of what the new engine should look like if all proposed changes prove beneficial. Notice the positioning of the ITB; this will hopefully make the afterburner no longer necessary. The other point to be made regarding this system is that, in general, as one increases the pressure ratio, the efficiency also rises, and as one increases the bypass ratio, the efficiency also increases. Both are goals with this system. With a highly compressed bypass stream, this engine theoretically will be much more efficient than the previous engine, and with the rise of new technology and the addition of the ITB as a new location to add heat, this will likely be far better than the Olympus.

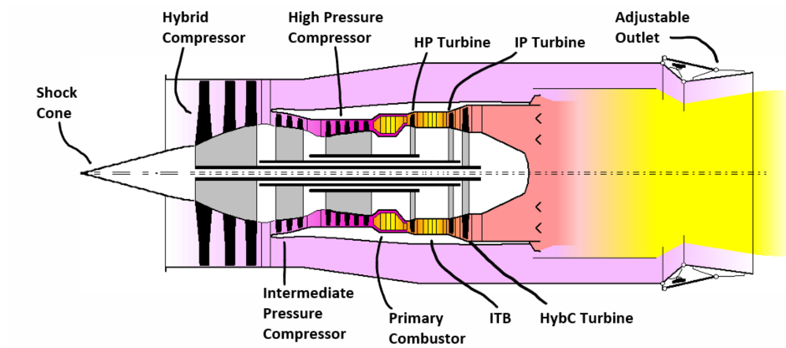

Fig. 7 Rendering of SSBBR Engine

The area under a temperature entropy graph is the work output of an engine. In **Fig 8** shown below, there is a general comparison between the system of the new engine verses the Olympus. As we want to avoid generating entropy, the ITB works better as it is reheating "high-energy" high pressure air, creating roughly the same work output with less entropy generation. In conclusion, we hope that these coming advancements in propulsion technology will create real measurable improvements in supersonic travel. Results will be finished by the paper conference. We hope to achieve a bypass ratio of 1.0, remove turbine cooling entirely, decrease full burn by 20% and decrease emissions.

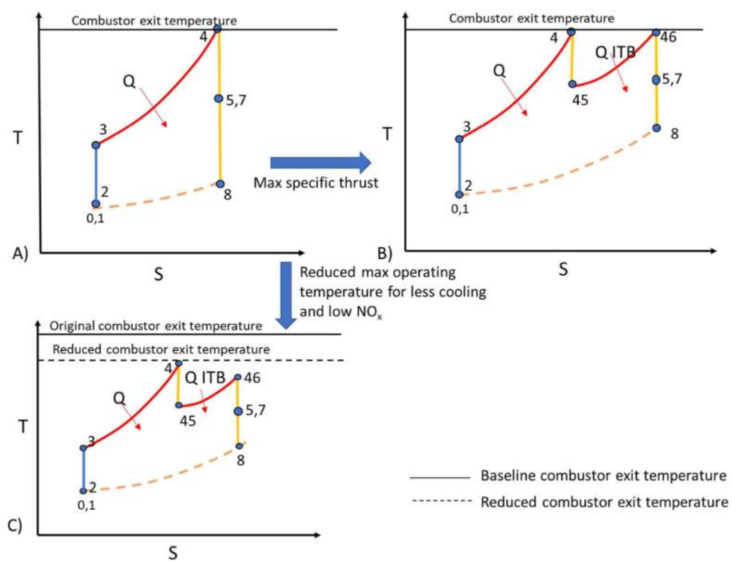

Fig. 8 Projected TS diagram for SSBBR Engine

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