A Parametric Cycle Analysis of a Separate-Flow Turbofan With Interstage Turbine Burner

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A PARAMETRIC CYCLE ANALYSIS OF A SEPARATE-FLOW TURBOFAN WITH INTERSTAGE TURBINE BURNER

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Today’s modern aircraft is based on air-breathing jet propulsion systems, which use moving fluids as substances to transform energy carried by the fluids into power. Throughout aero-vehicle evolution, improvements have been made to the engine efficiency and pollutants reduction. This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (ITB). The ITB considered in this paper is a relatively new concept in modern jet engine propulsion. The ITB serves as a secondary combustor and is located between the high- and the low-pressure turbine, i.e., the transition duct. The objective of this study is to use design parameters, such as flight Mach number, compressor pressure ratio, fan pressure ratio, fan bypass ratio, linear relation between high- and low-pressure turbines, and high-pressure turbine inlet temperature to obtain engine performance parameters, such as specific thrust and thrust specific fuel consumption. Results of this study can provide guidance in identifying the performance characteristics of various engine components, which can then be used to develop, analyze, integrate, and optimize the system performance of turbofan engines with an ITB.

I. Introduction

In most common air-breathing propulsion engines, fluid air is used as a medium to convert air-fuel mixture into kinetic energy. The energy is used for different applications. These engines convert high pressure and high temperature gas from the combustion chamber into work through the turbine to power the fan for turbofan, shaft for turbo-shaft, and propeller for turboprop, in addition to drive the compressor and accessories. Increases in pressure and momentum across the engine produce sufficient thrust to power the aircraft.

Throughout aero-vehicle evolution, scientists and engineers have attempted to improve engine efficiency, to make it smaller, lighter, require less fuel consumption, and yet more powerful. This type of engine is suitable for combat military aircrafts and long-range commercial aircrafts, since most of their designs are constrained by the weight of the engines and the distance of the flight. Lighter engines mean the aircraft can carry more payload and fuel for long combat operation or long flight time. Scientists have proposed solutions of how to achieve these goals, and one of them is introducing an Interstage Turbine Burner (ITB) into the engines.

Almost all commercial aircraft engines have transition duct between the high-pressure (HPT) and the low-pressure turbine (LPT). The ITB considered in this study, which is a relatively new concept in gas turbine engines, is to use the transition duct as a secondary combustor. By doing so, no new component is added to the existing system. One should note that ITB being studied here is equivalent to the 1-ITB engine in Liu and Sirignano, where the 1-ITB can be conveniently located between HPT and LPT or in the stator of the turbine stages.

The major advantages associated with the use of ITB are an increase in thrust and reduction in NOx emission, as illustrated in Figure 1. In Figure 1a, the inlet temperature of the HPT remains unchanged. As the flow undergoes secondary combustion, a higher specific thrust (ST) results. This implies a smaller and lighter engine and hence, lower cost and higher payload. Figure 1b shows the case in which the peak temperature inside the primary combustor is decreased; therefore the amount of thermal NOx production can be reduced. Furthermore, by lowering the temperature of the primary combustor and the HPT, a smaller amount of cooling air is required. Another advantage is the safety improvement, where flameout of either main burner or the ITB will not shut down the engine.

Sirignano and Liu mention that one major
consequence of increasing engine thrust-to-weight ratio is that, combustion residence time may become shorter than the time required to complete combustion. Therefore, complete combustion process will take place in the turbine passages. This is generally undesirable because it is too technically difficult to burn fuel in a turbine rotor. Constructing what they called M-ITB, between turbine stages remedies this problem, where M represents a number of ITBs. The study in Liu and Sirignano also shows that turbofan base engine and turbofan base engine with afterburner do not compete much with a turbofan base engine with M-ITB. Introducing an afterburner to the engine does increase specific thrust but at the expense of the thrust specific fuel consumption (TSFC). One expects to see an increase in ST and a decrease in TSFC in the final design of the engine. Unfortunately, an increase in ST always results in an increase of TSFC, for constant thermal efficiency. Nevertheless, a turbofan engine with M-ITB results in a tremendous increase in ST and only small increase in TSFC. Turbofan base engines with ITB increase its engine performance further by varying engine design parameters, such as compressor pressure ratio (CPR), fan pressure ratio (FPR), and bypass ratio (BR), in addition to the new material that can sustain higher pressure and higher temperature.

This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (ITB), which is also known as the on-design analysis.

II. Aircraft Engine Performance Parameters

There are several of the air-breathing engine performance parameters that are useful in aircraft propulsion design. The first performance parameter is the thrust of the engine for sustaining flight. Thrust is the force produced due to momentum and pressure increases across the engine. It is used to sustain (thrust = drag), accelerate (thrust > drag), or decelerate (thrust < drag) a flight. To increase thrust, one can introduce an additional nozzle to raise jet velocities or add an afterburner.

ST is defined as thrust (T) produced per unit mass airflow rate. In the other words, it defines the amount of fluid air needed to produce a level of thrust. It also means how effective the size of the engine can produce a certain amount of thrust. The specific thrust is described by the following equation:

$$ ST = \frac{T}{m_{\text{air}}} $$

In this paper, $m_{\text{air}}$ is defined as the air mass flow rate of the engine core, $m_c$.

However, one would like to know how significant the thrust increase is compared to the amount of fuel being injected. This leads to the definition of TSFC, which is the second performance parameter. The TSFC defines the rate of total mass flow rate of fuel per unit thrust produced. Accordingly,

$$ TSFC = \frac{m_f}{T} $$

This equation describes how effective a mass unit amount of fuel injected can produce thrust. Small TSFC indicates small fuel consumption for the same level of thrust produced, which is generally sought.
Other useful engine performance parameter is thermal efficiency, which is defined as the net rate of the kinetic energy gain out of the engine divided by the rate of thermal energy available from the fuel:

\[ \eta_{in} = \frac{\dot{E}_{kinetic,gain}}{\dot{m}_f \cdot h_{fg}} \]  

(3)

### III. Parametric Cycle Analysis for A Separate-Flow Turbofan Engine with ITB

In this study, the air stream entering turbofan engine (station number 0) will flow through the fan and the engine core separately. The fan increases the propellant mass flow rate with an accompanying decrease in the exit velocity for a given thrust.

**Figure 2 Station numbering of a turbofan engine with ITB**

The station numbering for the turbofan cycle analysis with ITB is given in Figure 2, in which the ITB (the transition duct) is located between station 5 and 6. The station numbering and the calculation steps follow closely as described in Mattingly3. We assume that (1) the working fluid is air which behaves as a perfect gas with constant properties at three different sections: \( \gamma_a, R_a, C_{p_a} \) for gas upstream of main burner, i.e., station 3; \( \gamma_b, R_b, C_{p_b} \) for gas between station 4 and 5, i.e., across the high-pressure turbine; \( \gamma_c, R_c, C_{p_c} \) for gas downstream of ITB, i.e., station 6, where \( \gamma \) is the specific heat ratio, \( R \) is the specific gas constant, and \( C_p \) is the constant pressure specific heat. (2) all components are adiabatic (i.e. no turbine cooling. However, effects of pressure and work losses due to turbine cooling can be taken into account by using uncooled turbine efficiency reduced by 1 to 3 percent as suggested by Hawthorne3); and (3) Constant polytropic efficiencies3 of compressor, turbine and fan will be used to relate the stage pressure ratio \( \pi \) to their temperature ratio \( \tau \). Cycle analysis is then applied to both the bypass stream and engine core stream separately as listed below.

**Bypass stream**

In this study, we consider only the uninstalled thrust \( (F) \), which depends on the engine alone and hence is independent of the nacelle. The uninstalled thrust of bypass stream \( (F_{fan}) \) is given by:

\[ F_{fan} = \frac{\dot{m}_{fan} (V_{t9} - V_0) + \dot{A}_{to}(P_{t9} - P_0)}{g_c} \]  

(4)

To express equation (4) in term of the free stream Mach number and sound speed \((M_0 \text{ and } a_0)\), temperature \((T)\) and pressure \((P)\), it can be rearranged to give:

\[ \frac{F_{fan}}{\dot{m}_{fan}} = a_0 \left[ \frac{V_{t9}}{a_0} - \frac{M_0}{V_0/a_0} \left( 1 - \frac{P_0}{P_{t9}} \right) \right] \]  

(5)

The velocity ratio \( V_{t9}/a_0 \) can be expressed in term of local Mach number, temperature, pressure, and gas properties as follows:

\[ \left( \frac{V_{t9}}{a_0} \right)^2 = \frac{a_{0}^2 M_{t9}^2}{\gamma a_0^2} - \left( \frac{T_{t9}}{T_0} \right) \left( \frac{T_{t9}}{T_0} \right)^{\gamma - 1} \]  

(6)

The ratio of fan exit temperature to ambient temperature \( T_{t9}/T_0 \) and fan exit Mach number \( M_{t9} \) are calculated by the following equations:

\[ \frac{T_{t9}}{T_0} = \frac{T_{t9}}{T_0} \left( \frac{P_{t9}}{P_0} \right)^{\gamma - 1} \left( \frac{T_{t9}}{T_0} \right)^{\gamma - 1} \]  

(7)

\[ M_{t9}^2 - \frac{2}{\gamma - 1} \left[ \left( \frac{P_{t9}}{P_0} \right)^{\gamma - 1} \right] \]  

(8)

The total-static temperature \( (T_{t9}/T_0) \) and pressure \( (P_{t9}, P_0) \) ratio can be calculated as:

\[ \frac{T_{t9}}{T_0} = \frac{T_{t9}}{T_0} \frac{T_{t12}}{T_{t13}} \frac{T_{t19}}{T_{t19}} = \tau_{r} \tau_{a} \tau_{fan} \tau_{fn} \]  

(9)

\[ \frac{P_{t9}}{P_0} = \frac{P_{t9}}{P_0} \frac{P_{t12}}{P_{t13}} \frac{P_{t19}}{P_{t19}} = \frac{P_{t9}}{P_0} \tau_{a} \tau_{a} \tau_{fan} \tau_{fn} \]  

(10)

where the free stream total/static temperature and pressure ratios \((\tau_r \text{ and } \tau_a)\) are given by:
The total-static temperature and pressure ratio can be calculated as:

\[
\frac{T_{10}}{T_0} = \tau_r \tau_d \tau_{\text{LPC}} \tau_{\text{HPC}} \tau_t \tau_{\text{PR}} \tau_{\text{ITB}} \tau_{\text{IT}} \tau_r \tau_n
\]  

(19)

Assuming isentropic processes in the inlet (diffuser) and exit (nozzle), it yields

\[
\tau_n - \tau_d - 1
\]  

(21)

Others, such as \(\tau_{\text{ITB}}, \tau_{\text{PR}}, \tau_{\text{LPC}}, \tau_{\text{HPC}}, \text{ and } \tau_{\text{IT}}\), are input parameters. The compressor pressure ratio \(\tau_c\) is the product of LPC and HPC pressure ratio and is one of the design parameters

\[
\tau_c = \tau_{\text{LPC}} \tau_{\text{HPC}}
\]  

(22)

**Main burner (station 3-4)**

Application of the steady flow energy equation to the main burner gives

\[
\dot{m}_c C_p T_{13} + \dot{m}_b h_{PR-b} = \dot{m}_4 C_p T_{14}
\]  

(23)

The ratio between total enthalpy of the main burner exit and ambient enthalpy, denoted by \(\tau_{\text{b}-\text{a}}\), is defined as

\[
\tau_{\text{b}-\text{a}} = \frac{(C_p T_{\text{b}})_{\text{burner exit}}}{(C_p T_{\text{b}})_{\text{free stream (ambient)}}}
\]  

(24)

and is an input design parameter. Rearranging and solving equation (23) for \(f_b\) yields

\[
f_b = \frac{\tau_{\text{b}} \tau_{\text{c}} - \tau_{\text{b}-\text{a}}}{\tau_{\text{b}-\text{a}} - \frac{\eta \dot{h}_{\text{PR-b}}}{C_p T_0}}
\]  

(25)

**Interstage Turbine Burner (station 5-6)**

Application of the steady flow energy equation to the ITB gives

\[
\dot{m}_4 C_p T_{15} + \dot{m}_{\text{ITB}} h_{PR-\text{ITB}} = \dot{m}_6 C_p T_{16}
\]  

(26)

Similarly, we introduce \(\tau_{\text{b}-\text{a}}\), which is the ratio between the total enthalpy of the ITB exit and the ambient enthalpy.
\[ \tau_{h-b = \text{exit}} = \frac{\left(C_p T_{h-b \text{ exit}} \right)}{\left(C_p T_{h-b \text{ free stream (ambient)}} \right)} \]  

Rearranging and solving equation (26) for \( f_{ib} \) yields

\[ f_{ib} = \frac{\tau_{h-b} \tau_{lpt} - \left( \frac{C_p}{C_p} \right) \tau_{h-b} \tau_{lpt}}{\left( \frac{C_p}{C_p} \right) \tau_{h-b} \tau_{lpt} \left( 1 + f_b \right)} \]  

Applying the energy equation to compressors, turbines and fan, gives

- Low-Pressure Compressor (LPC):
  \[ \dot{w}_{lpc} = m_c C_p \left( T_{12.5} - T_{12} \right) \]  \hspace{1cm} (29)

- High-Pressure Compressor (HPC):
  \[ \dot{w}_{hpc} = m_c C_p \left( T_{13} - T_{12.5} \right) \]  \hspace{1cm} (30)

- Low-Pressure Turbine (LPT):
  \[ \dot{w}_{lpt} = m_c C_p \left( T_{12} - T_{12} \right) \]  \hspace{1cm} (31)

- High-Pressure Turbine (HPT):
  \[ \dot{w}_{hpt} = m_c C_p \left( T_{14} - T_{15} \right) \]  \hspace{1cm} (32)

- Fan:
  \[ \dot{w}_{fan} = m_c C_p \left( T_{13} - T_{12} \right) \]  \hspace{1cm} (33)

For a dual-spool gas turbine engine, HPT and HPC are connected through a single shaft. Power extracted by HPT will be completely consumed by HPC, i.e.,

\[ \dot{w}_{hpc} = \dot{w}_{hpt} \]  \hspace{1cm} (34)

Accordingly, the total temperature ratio (\( \tau_{ht} \)) across the HPT becomes

\[ \tau_{ht} = \frac{1 - \tau_{hpc}}{\left( 1 + f_b \left( \frac{C_p}{C_p} \right) \tau_{hpc \ tau_{lpt}} m_{hpt} \right)} \]  \hspace{1cm} (35)

where \( \tau_{hpc} \) is one of the design parameters.

On the other hand, the LPC, LPT, and fan are connected through another shaft. For a turbofan, they are related by the following equation:

\[ \dot{w}_{lpc} + \dot{w}_{fan} = \dot{w}_{lpt} \]  \hspace{1cm} (36)

Similarly, the total temperature ratio (\( \tau_{lpt} \)) across the LPT becomes

\[ \tau_{lpt} = \frac{1 - \tau_{lpc} + \alpha \left( 1 - \tau_{fan} \right)}{\left( 1 + f_b \left( \frac{C_p}{C_p} \right) \tau_{lpc \ tau_{lpt}} m_{lpt} \right)} \]  \hspace{1cm} (37)

where \( \tau_{lpc} \) is one of the design parameters and \( \alpha \) is fan BR:

\[ \alpha = \frac{m_{fan}}{m_c} \]  \hspace{1cm} (38)

Although the powers are balanced individually ("two-unmixed-spool analysis"), power amounts between the HPC and LPC are not specified directly. Instead, the HPC and LPC total pressure ratios are split based on these two relations shown:

\[ \pi_c = \pi_{lpc} \cdot \pi_{hpc} \]  \hspace{1cm} (39)

\[ \pi_{lpc} = A \cdot \pi_{lpc} \]  \hspace{1cm} (40)

where \( A \) is a user input parameter, the value of which depends on the design. Based on the selected values of (\( \pi_{lpc} = 3.2 \), \( \pi_{hpc} = 12.5 \)), 3.90 is therefore selected for \( A \) in this study. Furthermore, the relation between \( \pi_{lpc} \) and \( \pi_{hpc} \) is not limited to be linear.

**Turbofan engine performance**

As indicated, the \( ST \) is defined as the total uninstalled thrust (through core engine and fan) per unit mass flow rate intake,

\[ ST = \left( \frac{F_c}{m_c} \right) + \alpha \left( \frac{F_{fan}}{m_{fan}} \right) \]  \hspace{1cm} (41)

and TSFC is defined as the total fuel flow rate (main burner and ITB) per unit thrust,

\[ TSFC = \frac{f_b + f_{lb}}{ST} \]  \hspace{1cm} (42)

Rearranging equation (3), the thermal efficiency can be computed as

\[ \eta_{th} = \frac{\alpha_0^2 \left[ \left( 1 + f_b + f_{lb} \left( \frac{V_{10}}{a_0} \right) \left( T_{10} \right)^2 - M_0^2 \right) + \alpha \left( \frac{V_{90}}{a_0} \right)^2 - M_0^2 \right]}{f_b \cdot \left( \theta_{PR-b} \right) \eta_b \left( f_{lb} \cdot \left( \theta_{PR-lb} \right) \right) \eta_{lb}} \]  \hspace{1cm} (43)
The Computer Codes

An Excel program was written in combination among spreadsheet neuron cells, Visual Basic, and macro code to provide user-friendly interface so that compilation and preprocessing are not needed. A Fortran code was also written based on the theory just described. The results computed using both codes were found to be consistent, regardless of the code precision used. The input data (operating conditions) for a two-spool, separate-flow turbofan engine with and without ITB were provided by NASA Glenn Research Center.

Two configurations were used in this study, i.e. base turbofan engine without ITB and turbofan engine with ITB. When ITB-OFF option is chosen, the program will execute as if there is no ITB added to the turbofan engine. Accordingly, the following variables will be set internally and automatically:

- $C_{pt} = C_{phb}, \gamma_t = \gamma_b$, and gas properties will be the same downstream of the main burner
- $f_{itb} = 0$, i.e., no fuel injected into the ITB
- $\pi_{itb} = 1.0$ and $\tau_{itb} = 1.0$, i.e., no pressure drop and temperature change across station 5 and 6

IV. Results and Discussions

For a range of flight Mach number, four engine design choices, namely (1) CPR, (2) FPR, (3) fan BR, and (4) linear relation of HPC and LPC total pressure ratios, and one design limitation (HPT inlet temperature) are studied. These design parameters are then used to obtain the system performance parameters of ST, TSFC, and $\eta_{th}$. Other design parameters, such as compressor/turbine efficiency, combustor efficiency, and pressure drop/increase across various components, are user-defined input parameters, as shown in Table 1.

Flight Mach number

Figure 3 shows the performance comparison for the turbofan engines with and without ITB at flight Mach number in the range of 0.0 to 3.0. For each engine, the CPR and FPR are fixed at 40 and 1.65 respectively with maximum allowable HPT inlet temperature ($T_{itb}$) of 1500K and maximum ITB exit temperature ($T_{itb}$) of 1900K. Relative to base engines at both BRs (1 and 6), adding ITB will increase ST by 25% with only a small increase in TSFC as flight Mach number increases. In addition, ITB engine allows a wider range of flight operation than conventional base engines. These trends of performance qualitatively agree with the finding of Liu and Sirignano.

As shown in Figure 3, it is clear that a high-BR base engine can operate at supersonic speed with an addition of ITB, without any penalty of very high fuel consumption, as long as the flight Mach number is less than 2.3. However, the situation may deteriorate if the aerodynamic effect is considered. On the contrary, engine at low-BR does not have this problem because of the smaller frontal area.

Compressor pressure ratio

Figure 4 compares the engine performance for varying CPR at a supersonic speed ($M_{f} = 1.5$) with $T_{itb} = 1500K$ and $T_{itb} = 1900K$. The FPR is fixed at 1.65. As CPR increases, all engines at both BRs exhibit a decrease in ST and thermal efficiency with an increase in TSFC. For a conventional base engine, higher CPR limits the heat addition in the main burner due to the higher inlet temperature of the incoming air. The situation is even worse at the supersonic flight when the ram effect introduces at least a pressure rise of 2.0 times the ambient pressure, which raises further the inlet air temperature of the main burner. The consequence is the decreasing trend of thermal efficiency as shown in Figure 4c. ITB remedies this problem by allowing secondary heat addition at a pressure relatively higher than the pressure of an afterburner at some military engines. Nevertheless, ITB engine is superior to base engine because it gains more than 50% increase in ST with only less than 20% increase in TSFC at CPR.
Holding all parameters constant, consider a typical military turbofan engine with a higher $FPR$ of 4.0 (multi-stage fans) and a low $BR$ of 1.0. The results are shown in Figure 5. One can observe the similar trend as discussed above, except that in Figure 5b, where the intersection point of the two engines’ $TSFC$ curves is shifted to a lower $CPR$ value (about 32) when comparing with Figure 4b. Beyond that point, the $TSFC$ of a conventional base engine will increase exponentially whereas ITB engine’s $TSFC$ stays steady as $CPR$ increases. Clearly, ITB can be a potential improvement to the military supersonic turbofan engine performance.

Figure 5 shows the performance comparison for the same types of engines at $M_0 = 0.87$, which is at the subsonic flight. Because of the similar trend of performance, what we have observed and discussed in the supersonic flight conditions can be equally applied to the subsonic flight conditions. However, at subsonic flight, pressure rise due to ram effect is lower, i.e. about 1.6 times the ambient pressure (at $M_0 = 1.5$, pressure rise is 3.67). Therefore, for ITB engines at both $BR$s, when $CPR$ is greater than 30, $ST$ and $TSFC$ are almost independent of $CPR$.

**Fan pressure ratio**

Increasing $FPR$ is a way to supply more energy to bypass flow. In Saravanamuttoo et al., military engines may have two- or three-stages fan with $FPR$ as high as 4.0 whereas civil engines will always use a single-stage fan with $FPR$ of about 1.5 to 1.8. Now, let us focus our

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**Figure 4** Performances of turbofan engines vs. $CPR$ at $M_0 = 1.5$, $\gamma = 1.65$, $T_{td} = 1500K$, and $T_{ts} = 1900K$.

**Figure 5** Performances of turbofan engines vs. $CPR$ at $M_0 = 1.5$, $\gamma = 4.0$, $T_{td} = 1500K$, and $T_{ts} = 1900K$. 

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attention on the engines with $BR = 1.0$ where most supersonic military aircrafts use to keep the frontal area down. As shown in Figure 7, ITB engine gains the benefit of increasing FPR, where $ST$ increases and TSFC decreases gradually. For base engine, $ST$ increases initially and starts decreasing at $FPR = 2.0$, because more work is extracted from LPT to fan in order to achieve the specified FPR. Unfortunately, lower energy at LPT exhaust stream results in lower average exit velocity of the fan and the engine core stream, and thus lower thrust. However, the secondary heat addition in ITB supplies more energy to LPT to drive the fan with only slight increase in TSFC. Furthermore, as shown in Figure 7b, TSFC of ITB engine is becoming lower than that of the base engine at FPR beyond 2.7, indicating that ITB operates more efficiently.

For subsonic flight, performance trend is qualitatively similar to the supersonic flight condition and will not be shown here.

**Fan bypass ratio**

Figure 8 shows the performance comparison for varying fan $BR$ at two FPR settings. Clearly, as $BR$ increases, base engine with high FPR (i.e., 4.0) exhibits an exponential-like increase in TSFC at a supersonic cruise. It ceases to produce thrust at $BR$ beyond 1.3. However, adding ITB to this engine will not only widen its operation range up to moderate $BR$ (say 3.0), but also gain more than 100% increase in $ST$ accompanied by a decreasing trend of TSFC as $BR$ increases. It may be unfeasible to operate a supersonic engine at a moderate $BR$. Nevertheless, advance in turbine technology (increased $T_{t4}$) will soon allow using a fan with $BR$ larger than those traditionally used in supersonic turbofan engines ($BR = 0.5$ or less) while maintaining a reasonably small frontal area.

For a subsonic flight, Liu and Sirignano clearly indicated the benefit of increasing fan $BR$ on $ST$. Not surprisingly, our results (not shown here) show similar qualitative trend as in Liu and Sirignano. However, as seen on some gas turbine literature, the inlet air mass flow, $m_{air}$, in equation (1) is sometimes defined as the

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**Figure 6** Performances of turbofan engines vs. $CPR$ at $M_0 = 0.87$, $\eta_f = 1.65$, $T_{i4} = 1500K$, and $T_{in} = 1900K$.

**Figure 7** Performances of turbofan engines vs. $FPR$ at $M_4 = 1.5$, $\alpha = 1.0$, $\eta_{pc} = 3.2$, $\eta_{hpc} = 12.5$, $T_{i4} = 1500K$, and $T_{in} = 1900K$. 

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total air mass flow rate of both engine core and fan. Consequently, one will see a decreasing trend of $ST$ as fan $BR$ increases.

**Linear relation of HPC and LPC total pressure ratios**

Comparing to an engine with a parameter $A = 1$, $A$ greater than 1 implies that ‘more power’ is needed to drive the HPC. On the other words, an increase in $A$ results in more power produced at HPT or reduction of the inlet pressure and temperature in ITB. From Figure 9, one can see that both $ST$ and $TSFC$ are proportional to $A$. This is because more fuel can be burned in ITB in order to meet the specified ITB exit temperature requirement. However, an increase in $TSFC$ implies that the rate increase in the amount of fuel injected is greater than the engine thrust produced. For a given large $A$ value, one must notice that fuel is now burned at a much lower pressure. Possibly, the ITB engine is operating like an afterburner engine for a very large $A$. In addition, as $A$ increases, thermal efficiency decreases faster than the engine configuration with lower $A$. It is clear that, for a given turbofan engine configuration, parameter $A$ is an important engine design variable. One wants to maximize and optimize $A$ to have a more control over engine thrust.

Variation of $A$ value between 0 and 1 constrains the amount of heat addition in the ITB because of smaller work required to drive HPC, or smaller work produced by HPT. As a result, there is only a small amount of temperature drop across HPT, and thus higher inlet temperature of ITB. Accordingly, restricting $A$ value between 0 and 1 is not desirable.

**HPT Inlet Temperature**

For a base engine with $BR = 1.0$ and $FPR = 4.0$, the minimum $T_{td}$ has to be at least 1400K as shown in Figure 10a. With ITB, $T_{td}$ can be as low as about 1100K while producing more $ST$ and reducing $TSFC$. Further increase of $BR$ to a moderate value, say 3.0, holding other parameters the same, base engine will not produce any thrust at all. However, as shown in Figure 10a, a $BR=3.0$ powered ITB engine at $T_{ts}=1600K$ (not 1900K as used in all other cases) is still able to operate at $T_{td}$ greater than 1500K. Preliminary result (not shown here) indicates that, at higher $T_{ts}$, the same engine can operate at lower $T_{td}$, i.e. 1400K, yet producing more $ST$ with less $TSFC$. From preceding discussion, it is clear that the addition of ITB makes the operation of a supersonic turbofan engine up to a moderate $BR$ ($=3.0$) possible. However, as shown in Papamoschou, in order to keep the frontal area small, it requires advanced turbine technology for a very high $T_{td}$. It may be interesting to investigate whether one can apply the ITB’s advantage (i.e., lowering $T_{td}$ through addition of ITB) to keep the same $T_{td}$ while maintaining a small frontal area.
For subsonic flight, ITB engines benefit equally well from lower $T_t4$ as discussed in supersonic flight.

V. Conclusions

Results of the parametric studies presented in this paper can be summarized as followings:

1) An ITB engine gains more benefit when operating at high flight Mach number. It provides a design basis for high-performance engines applicable to lightweight and/or high-speed aircraft.

2) Allowing heat addition in the ITB without a significant increase of TSFC further extends the operational range of compressor pressure ratio. At subsonic flights, ITB engine maintains almost the same level of ST and TSFC over the operating range.

3) Most low-BR supersonic engine with high FPR benefits from ITB, because it does not have the penalty of very high fuel consumption, yet producing higher ST.

4) With ITB, supersonic engine now can operate at a moderate BR.

5) Value for the linear relation between HPC and LPC total pressure ratios, i.e., parameter $A$, must be greater than 1 and its value requires optimization for a given mission.

6) Through the addition of ITB, HPT inlet temperature can be lower while producing more ST with less TSFC.

Although there are many advantages of using ITB for better engine performance, there are also challenges needed to be resolved. Specific hardware design challenges are:

- High velocities with possible swirl in the transition duct are important issues regarding the flame stability.
- Higher LPT temperatures than the conventional engine will require redesign of the LPT cooling system with a reduction in LPT stages.
- Integration and complexity of a second combustor, including all associated cooling and control requirements, needs to be overcome.

The current computer program is written for a specific engine configuration, namely unmixed two-spool turbofan engine with a separate fan and engine core stream nozzles. Application of an ITB is not limited to this configuration. Therefore, to increase the flexibility

<table>
<thead>
<tr>
<th>Description</th>
<th>Input value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Polytropic efficiency</td>
<td></td>
</tr>
<tr>
<td>Fan ($\eta_{fan}$)</td>
<td>0.8961</td>
</tr>
<tr>
<td>Low-pressure compressor ($\eta_{elp}$)</td>
<td>0.9036</td>
</tr>
<tr>
<td>High-pressure compressor ($\eta_{ehp}$)</td>
<td>0.9066</td>
</tr>
<tr>
<td>High-pressure turbine ($\eta_{ehpt}$)</td>
<td>0.9029</td>
</tr>
<tr>
<td>Low-pressure turbine ($\eta_{elpt}$)</td>
<td>0.9174</td>
</tr>
<tr>
<td>Total pressure ratio</td>
<td></td>
</tr>
<tr>
<td>Inlet ($\eta_{in}$)</td>
<td>0.99</td>
</tr>
<tr>
<td>Main burner ($\eta_b$)</td>
<td>0.96</td>
</tr>
<tr>
<td>ITB ($\eta_{ITB}$)</td>
<td>0.96</td>
</tr>
<tr>
<td>Nozzle ($\eta_n$)</td>
<td>0.99</td>
</tr>
<tr>
<td>Fan nozzle ($\eta_{fn}$)</td>
<td>1.3</td>
</tr>
<tr>
<td>Component efficiency</td>
<td></td>
</tr>
<tr>
<td>Main burner ($\eta_b$)</td>
<td>0.99</td>
</tr>
<tr>
<td>ITB ($\eta_{ITB}$)</td>
<td>0.99</td>
</tr>
<tr>
<td>Mechanical</td>
<td></td>
</tr>
<tr>
<td>Low-pressure spool ($\eta_{n-sp}$)</td>
<td>0.93</td>
</tr>
<tr>
<td>High-pressure spool ($\eta_{n-hp}$)</td>
<td>0.92</td>
</tr>
<tr>
<td>Specific heat at constant ratio (kJ/kg-K)</td>
<td></td>
</tr>
<tr>
<td>Region* 0$\rightarrow$3 ($C_{1}$)</td>
<td>1.004</td>
</tr>
<tr>
<td>Region 4$\rightarrow$5 ($C_{2}$)</td>
<td>1.096</td>
</tr>
<tr>
<td>Region 6$\rightarrow$10 ($C_{3}$)</td>
<td>1.089</td>
</tr>
<tr>
<td>Specific heat ratio</td>
<td></td>
</tr>
<tr>
<td>Region 0$\rightarrow$3 ($\eta_1$)</td>
<td>1.399</td>
</tr>
<tr>
<td>Region 4$\rightarrow$5 ($\eta_2$)</td>
<td>1.273</td>
</tr>
<tr>
<td>Region 6$\rightarrow$10 ($\eta_3$)</td>
<td>1.279</td>
</tr>
<tr>
<td>Fuel low heating value ($h_{PR}$)</td>
<td>42798.4 kJ/kg</td>
</tr>
</tbody>
</table>

Table 1 – input parameters

*Region numbers refer to Figure 2.
and usefulness of the program developed, additional options for different engine configurations would be a very desirable feature. Currently, a research on the performance (off-design) cycle analysis of a turbofan engine with an ITB is on-going.

**Nomenclature**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>( A )</td>
<td>cross-sectional area or linear constant</td>
</tr>
<tr>
<td>( a )</td>
<td>sound speed</td>
</tr>
<tr>
<td>( C_p )</td>
<td>specific heat at constant pressure</td>
</tr>
<tr>
<td>( \varepsilon )</td>
<td>polytropic efficiency</td>
</tr>
<tr>
<td>( i )</td>
<td>installed thrust</td>
</tr>
<tr>
<td>( f )</td>
<td>fuel/air ratio</td>
</tr>
<tr>
<td>( g )</td>
<td>Newton’s constant</td>
</tr>
<tr>
<td>( h_{fg} )</td>
<td>low heating value of fuel</td>
</tr>
<tr>
<td>( M )</td>
<td>Mach number</td>
</tr>
<tr>
<td>( \dot{m} )</td>
<td>mass flow rate</td>
</tr>
<tr>
<td>( P )</td>
<td>pressure</td>
</tr>
<tr>
<td>( P_t )</td>
<td>total pressure</td>
</tr>
<tr>
<td>( R )</td>
<td>universal gas constant</td>
</tr>
<tr>
<td>( T )</td>
<td>temperature or installed thrust</td>
</tr>
<tr>
<td>( T_t )</td>
<td>total temperature</td>
</tr>
<tr>
<td>( V )</td>
<td>absolute velocity</td>
</tr>
<tr>
<td>( W )</td>
<td>power</td>
</tr>
</tbody>
</table>

**Greek symbols**

<table>
<thead>
<tr>
<th>Symbol</th>
<th>Description</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \alpha )</td>
<td>bypass ratio</td>
</tr>
<tr>
<td>( \gamma )</td>
<td>ratio of specific heats, ( c_p/c_v )</td>
</tr>
<tr>
<td>( \eta_m )</td>
<td>mechanical Efficiency</td>
</tr>
<tr>
<td>( \eta_h )</td>
<td>thermal Efficiency</td>
</tr>
<tr>
<td>( \pi )</td>
<td>ratio of total pressure</td>
</tr>
<tr>
<td>( \pi_r )</td>
<td>(exception) ratio between total pressure and static pressure due to the ram effect, ( P/P )</td>
</tr>
<tr>
<td>( \tau )</td>
<td>ratio of total temperature</td>
</tr>
<tr>
<td>( \tau_r )</td>
<td>(exception) ratio between total temperature and static temperature due to the ram effect, ( T/T )</td>
</tr>
<tr>
<td>( \tau_0 )</td>
<td>ratio between total enthalpy and enthalpy at ambient condition</td>
</tr>
</tbody>
</table>

**Subscripts**

- \( b \) main burner or properties between main burner exit and ITB
- \( c \) properties between upstream and main burner or engine core
- \( d \) diffuser
- \( f \) fuel
- \( fn \) fan-nozzle
- \( hpc \) high pressure compressor
- \( hpt \) high pressure turbine
- \( itb \) interstage turbine combustors
- \( lpc \) low pressure compressor
- \( lpt \) low pressure turbine
- \( o \) inlet
- \( n \) nozzle
- \( r \) ram
- \( t \) properties between ITB exit and downstream or total/stagnation values of properties (i.e. temperature, pressure or enthalpy)

**References**


### Title and Subtitle

A Parametric Cycle Analysis of a Separate-Flow Turbofan with Interstage Turbine Burner

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### Abstract (Maximum 200 words)

Today's modern aircraft is based on air-breathing jet propulsion systems, which use moving fluids as substances to transform energy carried by the fluids into power. Throughout aero-vehicle evolution, improvements have been made to the engine efficiency and pollutants reduction. This study focuses on a parametric cycle analysis of a dual-spool, separate-flow turbofan engine with an Interstage Turbine Burner (ITB). The ITB considered in this paper is a relatively new concept in modern jet engine propulsion. The JTB serves as a secondary combustor and is located between the high- and the low-pressure turbine, i.e., the transition duct. The objective of this study is to use design parameters, such as flight Mach number, compressor pressure ratio, fan pressure ratio, fan bypass ratio, linear relation between high- and low-pressure turbines, and high-pressure turbine inlet temperature to obtain engine performance parameters, such as specific thrust and thrust specific fuel consumption. Results of this study can provide guidance in identifying the performance characteristics of various engine components, which can then be used to develop, analyze, integrate, and optimize the system performance of turbofan engines with an ITB.