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Performance Cycle Analysis of a Two-Spool, Separate-Exhaust Turbofan With Interstage Turbine Burner

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This paper presents the performance cycle analysis of a dual-spool, separate-exhaust turbofan engine, with an Interstage Turbine Burner serving as a secondary combustor. The *ITB*, which is located at the transition duct between the high- and the low-pressure turbines, is a relatively new concept for increasing specific thrust and lowering pollutant emissions in modern jet engine propulsion. A detailed performance analysis of this engine has been conducted for steady-state engine performance prediction. A code is written and is capable of predicting engine performances (i.e. thrust and thrust specific fuel consumption) at varying flight conditions and throttle settings. Two design-point engines were studied to reveal trends in performance at both full and partial throttle operations. A mission analysis is also presented to assure the advantage of saving fuel by adding *ITB*.

Nomenclature

A = cross-sectional area

a = sound speed

F = uninstalled thrust

f = fuel/air ratio, or function

g_c = Newton's constant

h_{PR} = low heating value of fuel

M = Mach number

$\dot{m}$ = mass flow rate

P = static pressure

P_t = total pressure

R = universal gas constant

S = uninstalled thrust specific fuel consumption

T_t = total temperature

V = absolute velocity

Greek symbols

α = bypass ratio

γ = specific heat ratio, c_p/c_v

η = efficiency

π = total pressure ratio

π_r = ratio between total pressure and static pressure due to the ram effect, P_t/P_0

τ = total temperature ratio

τ_r = ratio of total temperature and static temperature due to the ram effect, T_t/T_0

τ_λ = ratio of burner exit total enthalpy to enthalpy at ambient condition

Subscripts

b = main burner

c = engine core, compressor, or properties at upstream of main burner

cH = high pressure compressor

cL = low pressure compressor

d = diffuser

f = fan

itb = *ITB*, or properties at downstream of *ITB*

m = mechanical or constant value

n	=	constant value
0	=	engine inlet
o	=	total
R	=	reference conditions
t	=	properties between main burner exit and downstream, or total/stagnation values of properties
tH	=	high pressure turbine
tL	=	low pressure turbine

Abbreviations

HPC	=	High-Pressure Compressor
HPT	=	High-Pressure Turbine
ITB	=	Interstage Turbine Burner
LPC	=	Low-Pressure Compressor
LPT	=	Low-Pressure Turbine
MFP	=	Mass Flow Parameter
SLS	=	Sea Level Static

Introduction

Turbofan engine, a modern variation of the basic gas turbine engine, has gained popularity in most new jet-powered aircrafts, including military and civilian types. Basically, it is a turbojet engine with an addition of a fan. The fan causes more air to bypass the engine core and exit at higher speeds, resulting in greater thrust, lower specific fuel consumption and reduced noise level. Usually, the fan and low-pressure compressor (*LPC*) are connected on the same shaft to a low-pressure turbine (*LPT*). This type of arrangement is called a *two-spool* engine.

Interstage Turbine Burner (*ITB*) is relatively a new concept in modern jet engine propulsion. Most commercial turbofan engines have a transition duct between the high-pressure turbine (*HPT*) and the *LPT*. The *ITB* considered in this study is the placement of flame-holders inside the transition duct. *ITB* is also known as a reheat cycle[1], where the expanded gas from each expansion process in a turbine is reheated before the next expansion process, as shown in Fig. 1. In *ITB*, fuel is burnt at a higher pressure than a conventional afterburner, leading to a better thermal efficiency. The major advantages associated with the use of *ITB* are an increase in thrust and potential reduction in NO_x emission[2]. Recent studies on the turbine burners can be found in the literature, see for example, Liew et al.[2], Sirignano and Liu[3, 4], and Vogeler[5], However, these studies are limited to parametric cycle analysis only, which is also known as *on-design* analysis.

The work presented here is a systematic performance cycle analysis of a dual-spool, separate-exhaust turbofan engine with an *ITB*. Performance cycle analysis is also known as *off-design* analysis. It is an extension work for the previous study[2], i.e. on-design cycle analysis, in which we showed how the performance of a family of engines was determined by design choices, design limitations, or environmental conditions[6].

In general, off-design analysis differs significantly from on-design analysis. In on-design analysis, the primary purpose is to examine the variations of specific engine performance at a flight condition with changes in design parameters, including design variables for engine components. Then, it is possible to narrow down the desirable range for each design parameter. Once the design choice is made, it gives a so-called *reference-point* (or *design-point*) engine for a particular application. Off-design analysis is then performed to estimate how this specific reference-point engine will behave at conditions other than those for which it was designed. Furthermore, the performance of several design-point engines can be compared to find the most promising engine that has the best balanced performance over the entire flight envelope.

Approach

The station numbering for the turbofan cycle analysis with *ITB* is in accordance with APR 755A[7] and is given in Fig. 2. The *ITB* (the transition duct) is located between station 4.4 and 4.5.

The resulting analysis gives a system of eighteen nonlinear algebraic equations and equations for $A_{4.5}$ and A_8 that are solved for the dependent variables. Table 1 gives the variables and constants in this analysis. As will be shown, specific values of the independent variables m and n are desirable. Thus the number of independent variables is reduced to 5 and the number of dependent variables increased to 20.

Off-design Cycle Analysis

The following assumptions are employed:

1) The working fluid is air and products of combustion which behaves as perfect gases.

2) All component efficiencies are constant;

3) The area at each engine station is constant, except the areas at station 4.5 and 8;

4) The flow is choked at the *HPT* entrance nozzles (station 4), at *LPT* entrance nozzles (station 4.5), and at the throat of the exhaust nozzles (station 8 and 18).

5) At this preliminary design phase, turbine cooling is not included.

An off-design cycle analysis is used to calculate the uninstalled engine performance. The methodology is similar to those described in Mattingly[8,9]. Two important concepts are mentioned here to help explain the analytical method.

The first is called *referencing*, in which the conservation of mass, momentum, and energy are applied to the one-dimensional flow of a perfect gas at an engine steady-state operating point. This leads to a relationship between the total temperatures (τ) and pressure ratios (π) at a steady-state operating point, which can be written as $f(\tau,\pi)$ equal to a constant. The reference-point values (subscript 'R') from the on-design analysis can be utilized to give value to the constant and allow one to calculate the off-design parameters, as described below:

$$f(\tau,\pi)=f(\tau_R,\pi_R) \tag{1}$$

The second concept is the mass flow parameter (*MFP*), where the one-dimensional mass flow property per unit area can be written in the following functional form:

$$MFP = \frac{\dot{m}\sqrt{T_t}}{P_t A} = M\sqrt{\frac{\gamma g_c}{R}}\left(1+\frac{\gamma-1}{2}M^2\right)^{\frac{\gamma+1}{2(1-\gamma)}} \tag{2}$$

This relation is useful in calculating flow areas, or in finding any single flow quantity, provided the other four quantities are known at that station.

Component Modeling

In off-design analysis, there are two classes of predicting individual component performance. First, actual component characteristics can be obtained from component hardware performance data, which give a better estimate. However, in the absence of actual component hardware in a preliminary engine design phase, simple models of component performance in terms of operating conditions are used.

High-pressure Turbine

Writing mass flow rate equation at station 4 and 4.5 in terms of the flow properties and *MFP* gives

$$\dot{m}_4 = \frac{P_{t4}}{\sqrt{T_{t4}}} A_4 MFP(M_4) = \dot{m}_3(1+f_b) \tag{3}$$

and

$$\dot{m}_{4.5} = \frac{P_{t4.5}}{\sqrt{T_{t4.5}}} A_{4.5} MFP(M_{4.5}) = \dot{m}_3(1+f_b+f_{itb}) \tag{4}$$

Rearranging Eqs. (3) and (4), and equating $\dot{m}_3$ yields

$$\frac{P_{t4.4}}{P_{t4}}\frac{\sqrt{T_{t4}}}{\sqrt{T_{t4.5}}} A_{4.5} = A_4 \frac{MFP(M_4)}{MFP(M_{4.5})}\frac{(1+f_b+f_{itb})}{(1+f_b)}\frac{P_{t4.4}}{P_{t4.5}} \tag{5}$$

The right-hand side of the above equation is considered constant because of the following assumptions: the flow is choked at stations 4 and 4.5, the flow area at station 4 is constant, variation of fuel-air ratios (f) are ignored compared to unity and the total pressure ratio of *ITB* is constant. Using referencing, it yields

$$\frac{P_{t4.4}}{P_{t4}} \frac{\sqrt{T_{t4}}}{\sqrt{T_{t4.5}}} A_{4.5} = \left(\frac{P_{t4.4}}{P_{t4}} \frac{\sqrt{T_{t4}}}{\sqrt{T_{t4.5}}} A_{4.5} \right)_R \tag{6}$$

Rearranging and solving for $\pi_{tH} \, (= P_{t4.4} / P_{t4})$ yields

$$\pi_{tH} = \frac{\sqrt{\tau_{tH} \tau_{itb}}}{\left(\sqrt{\tau_{tH} \tau_{itb}}\right)_R} \frac{A_{4.5R}}{A_{4.5}} \pi_{tHR} \tag{7}$$

The equation relating π_{tH} and τ_{tH} comes from *HPT* efficiency equation:

$$\tau_{tH} = 1 - \eta_{tH} \{1 - \pi_{tH}^{(\gamma_t - 1)/\gamma_t}\} \tag{8}$$

$A_{4.5}/A_{4.5R}$ is related to the total temperature ratio of the *ITB* raised to the power of a value n:

$$A_{4.5} / A_{4.5R} = (\tau_{itb} / \tau_{itbR})^n \tag{9}$$

In the case when n equal to ½ in Eq. (9), then Eqs. (7) and (8) result in π_{tH} and τ_{tH} being constant at off-design condition.

Low-pressure Turbine

Writing the mass conservation at stations 4.5 and 8 using *MFP* and flow properties gives

$$\pi_{tL} = \pi_{tLR} \sqrt{\frac{\tau_{tL}}{\tau_{tLR}}} \frac{A_{8R}}{A_8} \frac{A_{4.5}}{A_{4.5R}} \frac{MFP(M_{8R})}{MFP(M_8)} \tag{10}$$

Similarly, *LPT* efficiency equation gives

$$\tau_{tL} = 1 - \eta_{tL} \left\{ 1 - \pi_{tL}^{(\gamma_{itb}-1)/\gamma_{itb}} \right\} \tag{11}$$

One relationship for A_8/A_{8R} is similar to $A_{4.5}/A_{4.5R}$ except that it is raised to the power of a value m

$$A_8 / A_{8R} = (\tau_{itb} / \tau_{itbR})^m \tag{12}$$

In the case when m equal to ½ in Eq. (12) and $M_8 = M_{8R}$, then Eqs. (10) and (11) result in π_{tL} and τ_{tL} being constant at off-design condition. With these functional relationships in Eq. (9) and (12), the engine's low pressure turbine performance will vary the same as the turbofan engine without the *ITB* when the *ITB* is turned off.

Engine Bypass Ratio

An expression for the engine bypass ratio is expressed by

$$\alpha = \frac{\dot{m}_f}{\dot{m}_c} \tag{13}$$

In terms of *MFP* and flow properties, the bypass ratio can be rewritten using referencing as

$$\alpha = \alpha_R \frac{\pi_{cLR} \pi_{cHR} / \pi_{fR}}{\pi_{cL} \pi_{cH} / \pi_f} \sqrt{\frac{T_{t4}/T_{t4R}}{\tau_r \tau_f / (\tau_{rR} \tau_{fR})}} \frac{MFP(M_{18})}{MFP(M_{18R})} \tag{14}$$

Fan and Low-pressure Compressor

The equation for the total temperature ratio of the fan, which can be derived directly from the power balance of the low-pressure spool, is written as

$$\tau_f = 1 + (\tau_{fR} - 1)\,\eta_{mL}\frac{\tau_{\lambda-itb}}{\tau_r}\left\{\frac{(1-\tau_{tL})(1+f_b+f_{itb})}{\tau_{cLR}-1+\alpha(\tau_{fR}-1)}\right\} \tag{15}$$

Fan total pressure ratio is given by

$$\pi_f = \{1+\eta_f\,(\tau_f - 1)\}^{\gamma_c/(\gamma_c-1)} \tag{16}$$

Since the *LPC* and the fan are on the same shaft, it is reasonable to approximate that the total enthalpy rise of *LPC* is proportional to that of the fan. The use of referencing thus gives

$$\frac{h_{t2.5}-h_{t2}}{h_{t13}-h_{t2}} = \frac{\tau_{cL}-1}{\tau_f-1} = \left(\frac{\tau_{cL}-1}{\tau_f-1}\right)_R \tag{17}$$

Equation above is rewritten to give the *LPC* total temperature ratio:

$$\tau_{cL} = 1 + (\tau_f - 1)\frac{(\tau_{cL}-1)_R}{(\tau_f-1)_R} \tag{18}$$

The *LPC* total pressure ratio is expressed as

$$\pi_{cL} = \{1+\eta_{cL}\,(\tau_{cL}-1)\}^{\gamma_c/(\gamma_c-1)} \tag{19}$$

High-pressure compressor

From the power balance of the high-pressure spool, solving for the total temperature ratio across *HPC* gives

$$\tau_{cH} = 1 + \eta_{mH}\,(1+f_b)\frac{\tau_{\lambda-b}(1-\tau_{tH})}{\tau_r\tau_{cL}} \tag{20}$$

HPC total pressure ratio is then given by

$$\pi_{cH} = \{1+\eta_{cH}(\tau_{cH}-1)\}^{\gamma_c/(\gamma_c-1)} \tag{21}$$

Exhaust Nozzles

The Mach number at both core (stations 8, 9) and fan exhaust nozzles (stations 18, 19) follows directly using

$$M_9 = \sqrt{\frac{2}{\gamma_{itb}-1}\left\{\left(\frac{P_{t9}}{P_9}\right)^{\frac{\gamma_{itb}-1}{\gamma_{itb}}}-1\right\}} \tag{22}$$

If $M_9 > 1$, then $M_8 = 1$, else $M_8 = M_9$ (23)

$$M_{19} = \sqrt{\frac{2}{\gamma_c-1}\left\{\left(\frac{P_{t19}}{P_{19}}\right)^{\frac{\gamma_c-1}{\gamma_c}}-1\right\}} \tag{24}$$

If $M_{19} > 1$, then $M_{18} = 1$, else $M_{18} = M_{19}$ (25)

Engine Mass Flow Rate

An expression for the overall engine mass flow rate follows by using *MFP* at station 4, giving

$$\dot{m}_0 = \dot{m}_{0R}\frac{1+\alpha}{1+\alpha_R}\frac{P_0\pi_r\pi_d\pi_{cL}\pi_{cH}}{(P_0\pi_r\pi_d\pi_{cL}\pi_{cH})_R}\sqrt{\frac{T_{t4R}}{T_{t4}}} \tag{26}$$

Fuel-air Ratios

Constant Specific Heat model[8] is used to compute the fuel-air ratios for main burner and *ITB*.

After the operating conditions for each engine component are determined, it is then possible to calculate the engine performance parameters.

Uninstalled specific thrust can be shown as

$$\frac{F}{\dot{m}_0} = \frac{a_0}{g_c(1+\alpha)}\left\{\begin{aligned}&\left[1+f_o(1+\alpha)\right]\frac{V_9}{a_0}+\alpha\frac{V_{19}}{a_0}-(1+\alpha)M_0\\&+\left[1+f_o(1+\alpha)\right]\frac{R_{itb}}{R_c}\frac{T_9/T_0}{V_9/a_0}\frac{(1-P_0/P_9)}{\gamma_c}\\&+\alpha\frac{T_{19}/T_0}{V_{19}/a_0}\frac{(1-P_0/P_{19})}{\gamma_c}\end{aligned}\right\} \tag{27}$$

where f_o is the total fuel-air ratio per engine inlet airflow, and a_0 is the sound speed of the incoming air.

Hence, *uninstalled thrust* produced by the engine is

$$F = \dot{m}_0 \left(\frac{F}{\dot{m}_0} \right) \tag{28}$$

Uninstalled thrust specific fuel consumption (*S*) is simply obtained by

$$S = \frac{f_o}{F/\dot{m}_0} \tag{29}$$

Engine Controls

A model for engine control system presented in Mattingly[8,9] is included into off-design analysis. It is necessary because it avoids compressor stalls or surges and also ensures that maximum limits on internal pressures, and turbine entry temperatures are not exceeded. In addition, *n* equal to ½ and *m* equal to ½ are used for area variations at stations 4.5 and 8, respectively.

Engine Configurations

Two sets of reference-point engine data at sea level static (*SLS*) condition are selected, i.e. case A and B, as provided in Table 2. For each case, engine operating with *ITB* on is termed as *ITB* engine. While *ITB* is turned off, it is considered as a baseline engine. In addition, the component performance parameters, listed in Table 3, are kept the same for both cases.

For full throttle operation, the maximum inlet *HPT* total temperature (T_{t4} or main burner exit total temperature) and the *LPT* inlet total temperature ($T_{t4.5}$ or *ITB* exit total temperature) are set to the values as listed in Table 2. For partial throttle operation, the minimum thrust is set to 20 percent of the maximum thrust.

A program was written in combination among *Microsoft*® *Excel* spreadsheet neuron cells, *VisualBasic*, and macro code to provide user-friendly interface so that the compilation and preprocessing are not needed.

Predicted Performance Results

Full Throttle Performance

Figures 3a-c present the uninstalled performance of the turbofan engines operating at full throttle settings for case A. These figures show the variations of thrust and thrust specific fuel consumption (S) with flight Mach number (M_0) and altitude, respectively. Two different altitudes are *SLS* condition and 10km. The solid lines represent *ITB* engine performance while the dashed lines represent baseline engine performance. While specific thrust is presented in on-design cycle analysis, thrust is commonly presented in off-design cycle analysis. As shown in Eq. (29), thrust accounts for the variation in both specific thrust and mass flow rate.

In Fig. 3a, *ITB* engines at two different altitudes exhibit an increase in thrust over the baseline engine as M_0 increases. Because of more fuel injected into *ITB* in addition to the main burner, *ITB* engines do have slightly higher fuel consumption than the baseline engine. Nevertheless, adding *ITB* is still beneficial because the improvement in thermal efficiency (Fig 3c) reflects that the gain in thrust offsets the slight increase in S. In addition, *ITB* engines perform even better at supersonic flight because there is no increase in S at all as M_0 is greater than 1.1.

In Figs. 3a and 3c, both thrust and thermal efficiency curves at 10km exhibits a slope change at a M_0 of 1.2. The engine control system takes place at that operating point in order to limit the main burner exit temperature from exceeding the maximum inlet turbine temperature limit.

Figures 4a-c present the uninstalled performance of the turbofan engines operating at full throttle settings for case B. It is found that both engines have similar performance trends over the flight spectrum as in case A. While gaining higher thrust, *ITB* engine at 10km starts consuming less fuel at M_0 greater than 0.7.

Partial Throttle Performance

Figures 5a-b (case A) and 6a-b (case B) show the '*S* versus thrust' and '*Thermal efficiency* versus thrust' curves at partial throttle settings for three different values of M_0 at an altitude of 10km. In Figs. 5a and 6a, the curves for *ITB* engines preserve the classical hook shape that is known as "*throttle hook*" in the propulsion community. The "neck" of each "hook" is the operating condition where the *ITB* is being shut off, resulting in a change in slope from a linear curve to a spline. This change is accompanied by an abrupt increase in *S* and a drop in thrust. As the throttle (i.e. T_{t4}) is further reduced, *S* will decrease slightly and start to increase at lower throttle settings.

According to Figs. 5a and 6a, it is clearly noticed that adding *ITB* further extends the engine operational range by producing higher thrust at lower *S* than that of a baseline engine. Further extending to full throttle setting, *ITB* engines may or may not yield a higher *S* than that of a baseline engine. For example, the fuel consumption for *ITB* engines at full load in case B is always lower while it is equal to or higher than that of baseline engine in case A. Therefore, to take full advantage of *ITB*, i.e., higher thrust at lower *S*, it is good enough to run the *ITB* engine at partial throttle settings. In addition, it is at partial-throttle setting where the highest thermal efficiency is attained, as shown in Figs. 5b and 6b. This will provide fuel saving to many aircraft engines, which normally run at partial throttle settings during cruise operations at high altitude. However, one drawback is that when *ITB* is turned off, *ITB* engine will consume more fuel to produce the same amount of thrust as a baseline engine.

Mission Analysis

A systematic mission studies of the fuel consumption is performed to reveal the advantage of saving fuel by adding *ITB*. However, at the preliminary design phase the engine manufacturer's published data is often unavailable; therefore, the off-design engine model like this one can be used to give a preliminary estimate of fuel consumption in each mission phase[9]. A 5% installation loss is accounted to give the mission analysis fuel consumption.

For the following mission study, only case A is considered. For simplicity, only critical mission phases and segments are selected. Each selected mission leg is judged to be critical because it has a high fuel consumption and is an extreme operating condition[9]. In each mission leg, the *ITB* engine is operating at partial load to attain the highest thermal efficiency as previously discussed.

Table 4 contains a summary of the mission performance of *ITB* engine (case A) as compared to baseline engine in term of fuel consumption. Each aircraft has an initial take-off weight of 24,000 lbf. It is found that *ITB* engine uses less fuel in all phases. Particularly, the fuel consumption in the Warm-up (1-2) phase is significantly less. This calculation also shows that *ITB* engine consumes about 3% less fuel for all those selected critical mission legs, which assure the fuel efficiency of an *ITB* engine over the baseline engine. To get an even better fuel consumption, one is free to return to the on-design cycle analysis[2] and choose other reference-point engines for further investigation.

Conclusions

A performance cycle analysis of a separate-flow and two-spool turbofan with *ITB* has been presented. The mathematical modeling of each engine component (e.g. compressors, burners, turbines and exhaust nozzles), in terms of its operating condition has been systematically described. Results of this study can be summarized as follows:

1) *ITB* engine at full throttle setting has enhanced performance over baseline engine.

2) *ITB* runs very efficiently at partial throttle setting, i.e., higher thrust at lower *S* than baseline engine. Furthermore, highest thermal efficiency is attained at this point.

3) At *ITB*-off condition, *ITB* engine consumes more fuel to produce the same amount of thrust as a baseline engine.

4) Mission study assures the *ITB* engine's advantage of saving fuel over the baseline engine.

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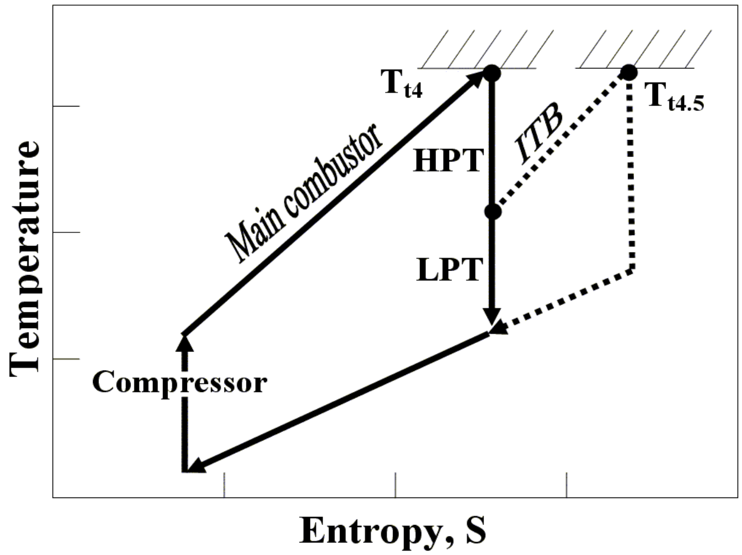


Figure 1. ***T-s*** **diagram of a gas turbine engine with** ***ITB***

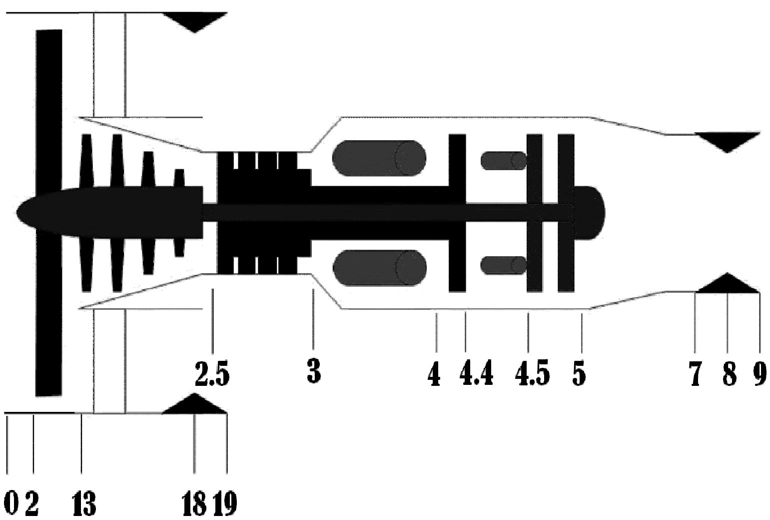
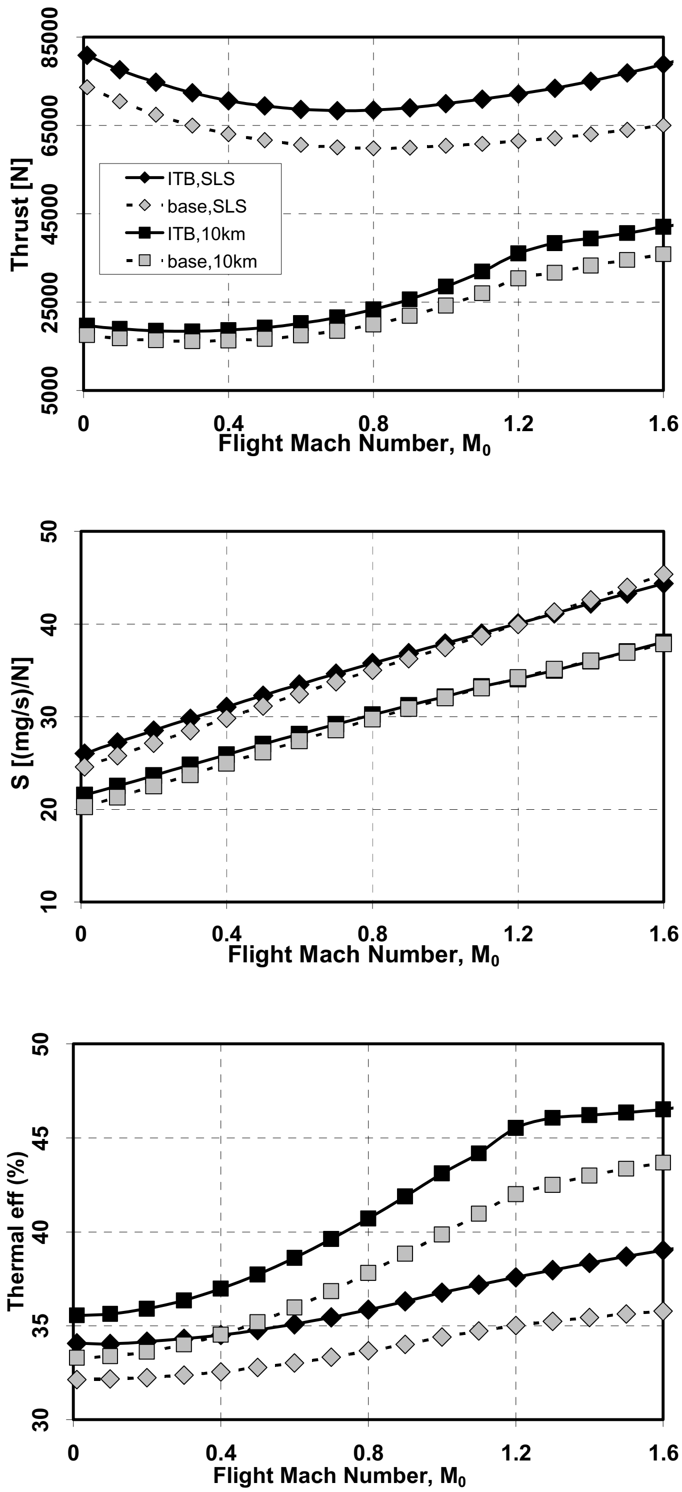


Figure 2. Station numbering of a turbofan engine with ***ITB***

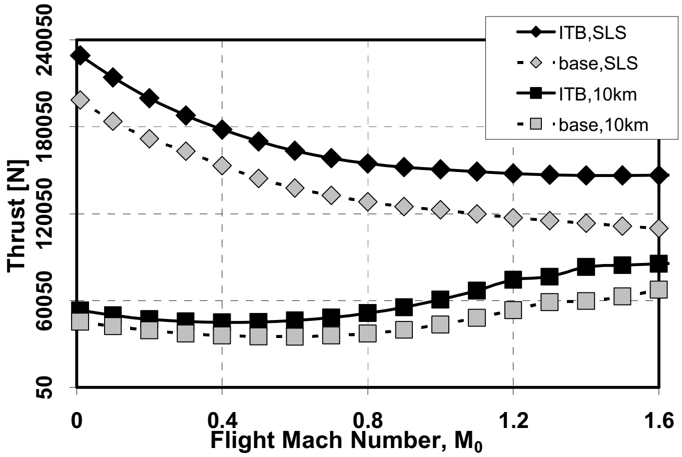


(a)

(b)

(c)

Figure 3. Full-throttle performance comparison of turbofan engines (case A) versus M_0, π_{fR} = 2.43, π_{cR} = 20, T_{t4R} = 1450K, $T_{t4.5R}$ = 1350K, $\dot{m}_{0R}$ = 118 kg/s,**and α_R = 0.73.**



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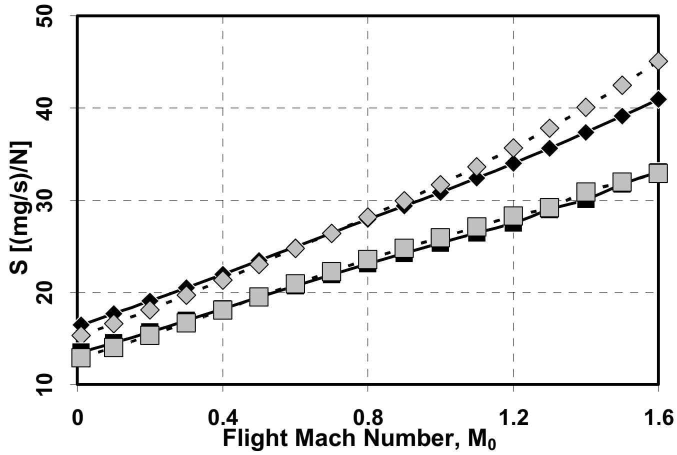

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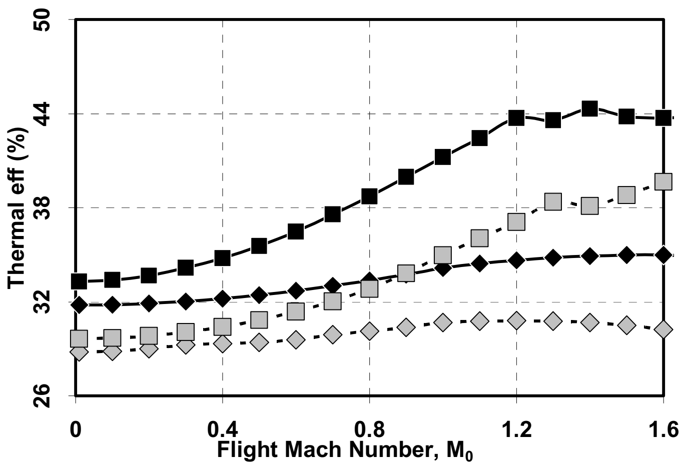

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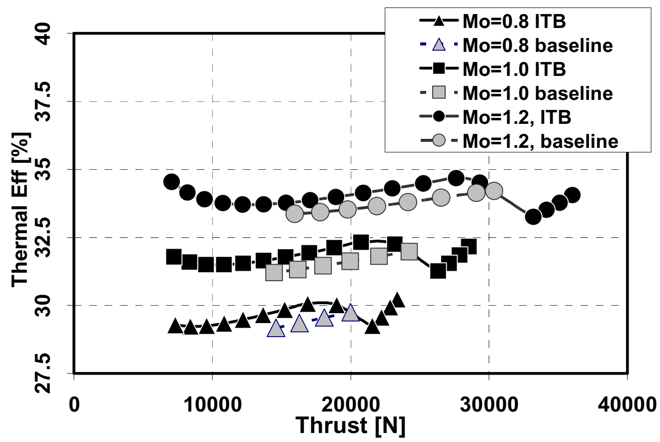
Figure 4. Full-throttle performance comparison of turbofan engines (case B) versus M_0, $\pi_{fR} = 2.2$, $\pi_{cR} = 25$, $T_{t4R} = 1550$K, $T_{t4.5R} = 1450$K, $\dot{m}_{0R} = 540$ kg/s, and $\alpha_R = 4.0$.



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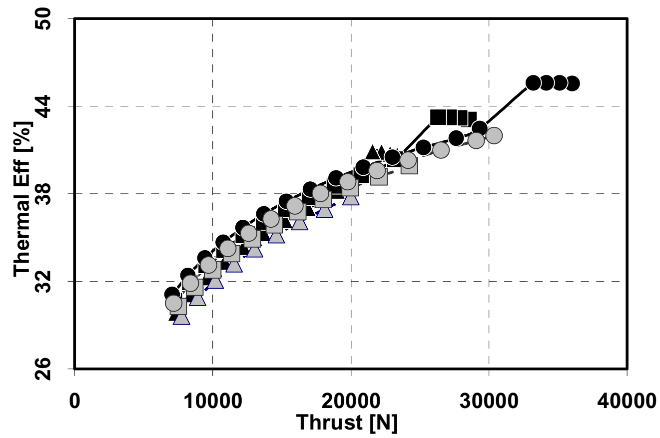

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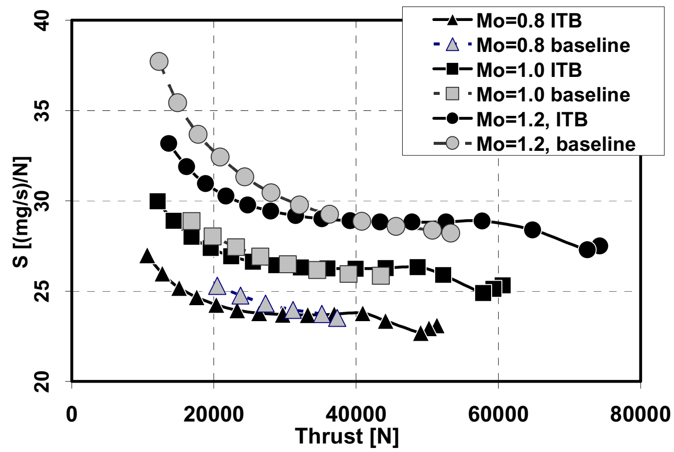
Figure 5. Partial-throttle performance of turbofan engine (case A) at altitude of 10km, π_{fR} = 2.43, π_{cR} = 20, T_{t4R} = 1450K, $T_{t4.5R}$ = 1350K, $\dot{m}_{0R}$ = 118 kg/s, and α_R = 0.73.



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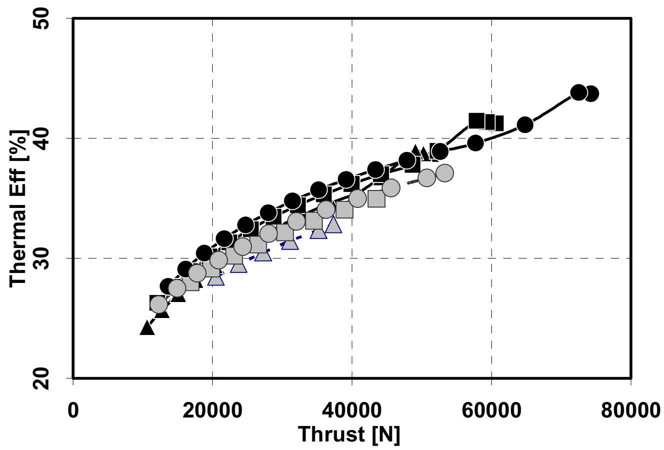

(b)

Figure 6. Partial-throttle performance of turbofan engine (case B) at altitude of 10km, π_{fR} = 2.2, π_{cR} = 25, T_{t4R} = 1550K, $T_{t4.5R}$ = 1450K, $\dot{m}_{0R}$ = 540 kg/s, and α_R = 4.0.

Table 1. Engine Performance Variables

Component	Independent Variable	Constant or Known	Dependent Variable
engine	M_0, T_0, P_0		$\dot{m}_0, \alpha$
diffuser		$\pi_d = f(M_0)$	
fan		η_f	π_f, τ_f
low-pressure compressor		η_{cL}	π_{cL}, τ_{cL}
high-pressure compressor		η_{cH}	π_{cH}, τ_{cH}
burner	T_{t4}	π_b	f
high-pressure turbine		η_{tH}, M_4	π_{tH}, τ_{tH}
inter-stage burner	$T_{t4.5}$	π_{itb}	f_{itb}
low-pressure turbine	n	$\eta_{tL}, M_{4.5}$, $A_{4.5} = f(\tau_{itb}, n)$	π_{tL}, τ_{tL}
fan exhaust nozzle		π_{fn}	M_{18}, M_{19}
core exhaust nozzle	m	π_n, $A_8 = f(\tau_{itb}, m)$	M_8, M_9
Total number	7		18

Table 2. Design-point engine reference data

Reference Conditions	Case A	Case B
Mach number (M_{0R})	0	0
Altitude (h_R)	*SLS*	*SLS*
Main burner exit total temperature (T_{t4R}, K)	1450	1550
ITB exit temperature ($T_{t4.5R}$, K)	1350	1450
Compressor pressure ratio (π_{cR})	20	25
Fan pressure ratio (π_{fR})	2.43	2.2
Fan bypass ratio (α_R)	0.73	4.0
Mass flow rate ($\dot{m}_{0R}$, kg/s)	118	540

Table 3. Engine component parameters

Component Parameters	Input value
Total pressure ratios	
Inlet ($\pi_{d,max}$)	0.99
Main burner (π_b)	0.95
ITB (π_{ITB})	0.95
Nozzle (π_n)	0.99
Fan nozzle (π_{fn})	0.98
Efficiencies	
Main burner (η_b)	0.99
ITB (η_{itb})	0.99
HP spool ($\eta_{m\text{-}HP}$)	0.92
LP spool ($\eta_{m\text{-}LP}$)	0.93
Polytropic Efficiencies	
Fan (e_f)	0.93
LP Compressor (e_{cL})	0.8738
HP Compressor (e_{cH})	0.9085
HP Turbine (e_{tH})	0.8999
LP Turbine (e_{tL})	0.9204
Fuel low heating value (h_{PR})	43,124 kJ/kg

Table 4. Summary of results for mission analysis (24,000 lbf of take-off weight)

	Mission phases and segments	M_0	Alt (kft)	Baseline Fuel used (lbf)	ITB Fuel used (lbf)	Fuel saved, (lbf)	Fuel saved (%)
1-2	A - Warm up	0.0	2	414	343	71	17.2
2-3	E - Climb/acceleration	0.875	23	483	474	9	1.9
3-4	Subsonic cruise climb	0.9	42	509	500	9	1.7
5-6	Combat air patrol	0.697	30	714	701	13	1.9
6-7	F - Acceleration	1.09	30	247	244	3	1.3
6-7	G - Supersonic penetration	1.5	30	1774	1713	61	3.4
7-8	I - 1.6M/5g turn	1.6	30	414	401	14	3.3
7-8	J - 0.9M/5g turn	0.9	30	297	291	5	1.8
7-8	K - Acceleration	1.2	30	228	225	3	1.2
8-9	Escape dash	1.5	30	518	503	16	3.1
10-11	Subsonic cruise climb	0.9	48	462	457	5	1.0
12-13	Loiter	0.378	10	631	624	7	1.1
			Total	**6691**	**6475**	**216**	**3.2**

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13. ABSTRACT *(Maximum 200 words)*

This paper presents the performance cycle analysis of a dual-spool, separate-exhaust turbofan engine, with an Interstage Turbine Burner serving as a secondary combustor. The ITB, which is located at the transition duct between the high- and the low-pressure turbines, is a relatively new concept for increasing specific thrust and lowering pollutant emissions in modern jet engine propulsion. A detailed performance analysis of this engine has been conducted for steady-state engine performance prediction. A code is written and is capable of predicting engine performances (i.e., thrust and thrust specific fuel consumption) at varying flight conditions and throttle settings. Two design-point engines were studied to reveal trends in performance at both full and partial throttle operations. A mission analysis is also presented to assure the advantage of saving fuel by adding ITB.

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