JOURNAL OF PROPULSION AND POWER
Vol. 14, No. 1, January–February 1998

Digital Simulator for Steady-State Performance Prediction of Military Turbofan Engine

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The significance of propulsion system steady-state performance prediction has been identified. The digital simulation techniques have been described for steady-state performance prediction, with component maps and a controller. In addition to performance prediction, the application of a simulation code has been demonstrated to an engine cycle redesigning case study without major modifications in already existing hardware. As per the current military application trends, the twin-spool turbofan with a mixed exhaust has been chosen as the propulsion concept. The proposed formulation can easily be modified to simulate the steady-state performance of the twin-spool turbojet engine and also to generate the control schedules in the early design stage.

Nomenclature

A = area
ALT = altitude
CD = nozzle flow blockage factor
C_p = specific heat
F = thrust
g = gravitational constant
H = enthalpy
J = Joule's constant
M = flight Mach number
N = rotational speed
P = total pressure
PR = nozzle pressure ratio
p = static pressure
R = gas constant
T = total temperature
t = static temperature
V = flow velocity
W = mass flow rate
Δ = change in quantity
γ = ratio of specific heats
η = component efficiency
θ = nozzle petal angle

Subscripts
CC = combustion chamber
G = gross (for gross thrust)
HC = high-pressure compressor
HT = high-pressure turbine
LC = low-pressure compressor
LT = low-pressure turbine
NZ = exhaust nozzle
P = pressure (for pressure thrust)

Introduction

THE propulsion unit when integrated with an airframe defines the aircraft weapon system. The propulsion system steady-state thrust dictates the level of performance that can be attained at a given aircraft weight and for a given configuration, while the specific fuel consumption will have a significant bearing on its range and endurance characteristics. Thus, propulsion system steady-state performance plays a very significant role in determining the mission performance of the aircraft with which it is integrated.

The propulsion system performance is the function of its cycle parameters. The mixed-flow twin-spool turbofan is a more widely accepted propulsion concept for military applications, though twin-spool turbojet may be more suitable for mission applications with high supersonic Mach numbers, during which aircraft is required to spend a greater portion of flight time in a supersonic regime.

In the conceptual design phase, where emphasis is mainly on parametric studies for configuration definition and preliminary sizing, a generalized and robust engine model, like the one reported in Sanghi,[1] that works over a wide flight envelope and over a large parametric variation in engine cycle variables, is required. The critical inputs such as component maps and control schedules are not available during the conceptual design phase. Hence, generalized engine models use alternate methods that work without component maps and operate on simplified assumptions such as constant component efficiency, constant pressure loss, choked turbines, and a fixed geometry exhaust nozzle, etc., at all flight points, to compute engine steady-state performance.

The generalized engine models are therefore not expected to reproduce the performance of any specific engine cycle. The primary purpose is to quickly estimate the engine performance during parametric studies, with reasonable levels of accuracy, to study the impact of varying levels of engine cycle variables on an aircraft weapon system response. The accuracy, though highly desirable, should not be overemphasized, keeping in view the lack of adequate design information during the conceptual design phase.

Thus, performance estimates within a maximum of $\pm 10\%$, with respect to an available engine-specific baseline model, are sufficient during validation. As indicated in Sanghi,[1] the large error occurs only at a few discrete flight points, which may not form the part of the mission envelope. Any loss in accuracy would be equally reflected in all of the parametric design combinations. It is therefore not likely to affect the identification of the regions of the optimum solution and, hence, the initial baseline design selection.

After the baseline optimum design is identified upon completion of a conceptual design, the design phase progresses to the detailed studies in which more details are added to the baseline design to improve upon the preliminary design esti-

Received June 1, 1997; revision received Aug. 12, 1997; accepted for publication Aug. 18, 1997. Copyright © 1997 by the American Institute of Aeronautics and Astronautics, Inc. All rights reserved.

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mates. The weapon system mission response is fine-tuned to account for any change in the engine performance as a result of the refinements in the engine model. The baseline design is identified for a reference condition, termed as the engine design point. Traditionally, a design point has been the sea level static condition in the international standard atmosphere.

With increasing levels of aircraft performance and the resulting complexity of a propulsion system because of stringent demands placed on it, the development cost and time of modern gas turbines have increased considerably. The customer requires assurance on the ability of the aircraft to satisfy the specified mission requirements, in the early design stage itself, which in turn, depends upon the propulsion system performance. It is therefore very important to have an accurate prediction of propulsion system steady-state performance, subsequent to the conceptual design studies, to have an accurate assessment of the mission capabilities of chosen baseline optimum aircraft configuration. Also, the aerodynamic and mechanical design cannot begin until an engine thermodynamic design has been completed, i.e., its thermodynamic state at the flight point at which mechanical and aerodynamic design is done and is defined.

The digital simulation has to be engine-specific and must be able to accurately represent its steady-state behavior over the entire flight spectrum. This paper describes the development and application of an engine digital simulation model to predict the propulsion system steady-state performance, upon completion of conceptual design phase.

The simulation model assumes the availability of component maps for representing component behavior and control schedules that define inputs for control variables such as spool speeds, main combustor fuel flow (or turbine entry temperature), nozzle area, and reheat fuel flow (or afterburner exit temperature) at a given flight and power setting condition. It is the low-pressure (LP) spool speed that is linked to engine power setting. The high-pressure (HP) spool speed is aerodynamically coupled to LP speed and is obtained as a dependent variable during performance estimation. Each power setting therefore corresponds to a certain value of LP compressor speed and an associated nozzle area, which then uniquely defines engine operation.

Though the availability of engine controller has been assumed, it is not available in the early design stage. The digital simulation is a highly useful tool to develop the steady-state control schedules. At each speed line of the LP compressor, the simulation code is run for various conditions of bypass ratio (for specified gas flow path areas of the cold and hot streams at the mixer entry), to arrive at the optimum engine operation conditions. The optimum operating conditions are the ones that have an adequate surge margin, a smooth and mechanically feasible nozzle area variation, thrust levels that meet the mission performance, and values of specific fuel consumption such that the mission fuel consumed is compatible with aircraft fuel capacity. Also, it must be simultaneously ensured that maximum limits on physical speed, corrected speed, turbine entry temperatures, and pressures are not exceeded.

The gas flow path areas or mixer geometry at the entry plane must ensure that none of the cold- and hot-stream Mach numbers take values that result in reverse or choked flow, to permit trouble-free engine operation over the entire flight envelope. Typically, design point Mach numbers of the cold and hot streams at the mixer inlet are in the 0.30–0.60 range, from which the mixer geometry is determined. Another important design condition is that the ratio of total pressures of the cold to hot stream at the mixer entry should be 1.00–1.10 to ensure proper mixing of the flows.

The values that control variables take at the optimum operating conditions at various flight points are correlated as some function of a power setting and a total temperature at engine entry, or similar parameters, which defines the control schedules. The power setting is defined in terms of a numerical value of the power lever angle (PLA). For example, x_1, x_2, and x_3 will always correspond to the engine idling, max dry, and max reheat condition, irrespective of the flight condition. The engine is in part a power dry mode if the PLA is between x_1 and x_2, and in part a power reheat mode if it is between x_2 and x_3.

Having defined the steady-state control schedules, the definition of LP spool speed and nozzle area is sufficient to obtain engine performance estimates in the dry mode, i.e., when the afterburner or reheat is off. The afterburner exit temperature is the additional parameter required to compute the performance in the reheat mode. Thus, digital simulation is highly useful and is an essential requirement for the development of the control schedules.

As the design phase progresses further, the engine hardware corresponding to the chosen cycle is subjected to experimental investigation by test bed running to evaluate its aerothermomechanical integrity. One or more of these components may not be able to achieve its theoretically intended performance because of the use of an ambitious technology level, which probably cannot be practiced with available resources and expertise. The problem may require finding alternate engine cycles with no major hardware modifications to account for the loss in component performance, without significantly penalizing the propulsion system performance, to save on development time and cost. This paper also shows the application of a steady-state simulation code to address such a design problem.

Engine Description

The first step in digital simulation is the description of engine system. The engine under study is a twin-spool, mixed-flow bypass engine, the typical schematic of which is shown in Fig. 1. The station designations are consistent with Aerospace Recommended Practice.[2] The intermediate stations are described in the next section.

Component Modeling

Having defined the engine system, the mathematical description of the behavior of various components that constitute the engine is required. The performance of rotating engine components is based on component maps or component characteristics.[3] These maps are obtained from the analytical and/or numerical methods in the early detail design phase and are converted into a look-up table form to facilitate their use for the purpose of digital simulation. The component maps are unique to an engine and must be changed while simulating another engine. The mathematical model is described next for each component.

Intake

Intake acts like a diffuser. Its mathematical modeling is described by the following equations where station 0 is for free-stream conditions, station 1 is the intake entry, and station 2

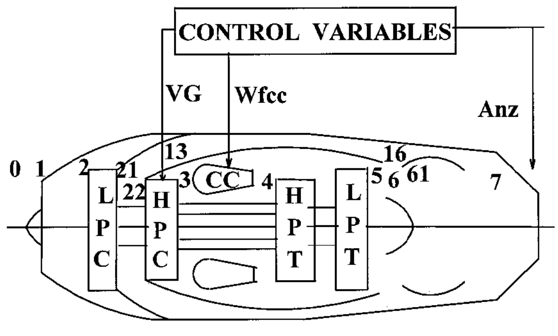

Fig. 1 Twin-spool mixed flow turbofan engine.

is the exit, i.e., entry to LP compressor. The provision exists to input user-defined intake loss:

$$\frac{T_1}{t_0} = \left(1 + \frac{\gamma - 1}{2} M^2\right), \qquad T_2 = T_1$$

$$P_1 = p_0 \left(\frac{T_1}{t_0}\right)^{\gamma/(\gamma-1)}, \qquad P_2 = P_1 \times \eta_{\text{intake}}$$

where

$$\begin{cases} \eta_{\text{intake}} = 1, & \text{if } M \leq 1 \\ \eta_{\text{intake}} = 1 - 0.075(M - 1)^{1.35}, & \text{if } M > 1 \end{cases}$$

LP Compressor

The LP compressor is represented by a set of overall performance maps in terms of flow function, rotational speed, pressure ratio, temperature ratio, and efficiency. For the known pressure ratio and speed, the flow and temperature ratio is calculated. The work required by the compressor is computed by calculating the change in enthalpy (ΔH) across the component. The following equations represent LP compressor modeling. Station 2 is entry and station 21 is exit:

$$\frac{W_2\sqrt{T_2}}{P_2} = f\left(\frac{P_{21}}{P_2}, \frac{N_{\text{LC}}}{\sqrt{T_2}}\right)$$

$$\frac{T_{21}}{T_2}, \eta_{\text{LC}} = f\left(\frac{P_{21}}{P_2}, \frac{N_{\text{LC}}}{\sqrt{T_2}}\right)$$

$$\text{WORK}_{\text{LC}} = W_2 \cdot \Delta H_{\text{LC}}$$

HP Compressor

The overall performance maps corresponding to different variable guide vane positions define the HP compressor. Its modeling is similar to the LP compressor. Station 22 is the entry (exit of intercasing duct), 23 is the intermediate station from where bleed for LP turbine cooling is extracted, and 3 is the exit. The bleed for the cooling HP turbine is taken from the HP compressor exit.

The simulation code can operate on a nominal variable geometry (VG) positioning schedule as a function of corrected speed ($N_{\text{HC}}/\sqrt{T_{22}}$), or for a specified VG position. Allowance is made in a HP compressor work calculation to account for the air bleed at the intermediate stage for LP turbine cooling. The ΔT_{23} is the temperature rise-up to intermediate station 23:

$$\frac{W_{22}\sqrt{T_{22}}}{P_{22}} = f\left(\frac{P_3}{P_{22}}, \frac{N_{\text{HC}}}{\sqrt{T_{22}}}, \text{VG}_{\text{HC}}\right)$$

$$\frac{T_3}{T_{22}}, \eta_{\text{HC}} = f\left(\frac{P_3}{P_{22}}, \frac{N_{\text{HC}}}{\sqrt{T_{22}}}, \text{VG}_{\text{HC}}\right)$$

$$\frac{\Delta T_{23}}{(\Delta T_{23})_{\text{Design Point}}} = \frac{N_{\text{HC}}^2}{(N_{\text{HC}}^2)_{\text{Design Point}}}$$

$$\text{WORK}_{\text{HC}} = W_{22} \times (H_{23} - H_{22}) + W_{23} \times (H_3 - H_{23})$$

Bleeds

All internal bleeds, for engine cooling and aircraft application, are modeled as constant fractions of HP compressor inlet flow. The mixing of bleed flows with core flow is carried out using only the mass flow and enthalpy balance. The momentum balance is ignored. The bleed air that is returned for cooling the nozzle guide vanes (NGV) only contributes to the turbine work output.

Combustor

The combustor is considered as a simple duct where fuel is burnt in compressed air. Station 3 is entry and station 4 is exit. The following equations are used in combustor modeling, where η_{CC} is the combustor efficiency (expressed as a function of the combustor loading parameter), Vol_{CC} is the combustor volume, EHV is the effective fuel heating value, FAR is the fuel–air ratio, and $\Delta P/P$ is the pressure loss factor in the combustor:

$$\eta_{\text{CC}} = f\left(\frac{W_3 \cdot 3600}{\text{Vol}_{\text{CC}} \cdot P_3^2 \sqrt{T_3 \cdot 1.8}}\right)$$

$$\text{EHV} = f(T_4)$$

$$\text{FAR} = \frac{\Delta H_{\text{CC}}}{\text{EHV} \cdot \eta_{\text{CC}}}$$

where

$$\Delta H_{\text{CC}} = (C_P \cdot T)_4 - (C_P \cdot T)_3$$

$$\frac{\Delta P}{P} = f\left(\frac{W_{31}\sqrt{T_{31}}}{P_{31}}\right)$$

The preceding modeling holds well when the fuel flow required to attain the specified turbine entry temperature (T_4) is to be computed. Instead, if fuel flow is specified, the T_4 computation follows an iterative path.

HP Turbine

The HP turbine is represented by a set of overall performance maps in terms of flow function, speed, pressure ratio, and enthalpy drop. For the known pressure ratio and speed, flow, temperature ratio, and work are calculated. Station 41 is the exit to NGV, station 42 is the rotor entry after incorporating the mixing of cooling bleed for NGV, station 43 is the rotor exit upon work balance, and 44 is the rotor exit after mixing of bleed for its cooling. The bleed for rotor cooling does not participate in turbine work output. The work balance equation of the HP spool must account for power tapped (P_{OT}) from the HP compressor to drive the engine and aircraft accessories:

$$N_{\text{HT}} = N_{\text{HC}}$$

$$\frac{W_{42}\sqrt{T_{42}}}{P_{42}} = f\left(\frac{P_{42}}{P_{43}}, \frac{N_{\text{HT}}}{\sqrt{T_{42}}}\right)$$

$$\frac{\Delta H_{\text{HT}}}{T_{42}}, \eta_{\text{HT}} = f\left(\frac{P_{42}}{P_{43}}, \frac{N_{\text{HT}}}{\sqrt{T_{42}}}\right)$$

$$\text{WORK}_{\text{HT}} = W_{42} \cdot \Delta H_{\text{HT}} = \text{WORK}_{\text{HC}} + P_{\text{OT}}$$

LP Turbine

The LP turbine modeling is similar to the HP turbine. Station 44 is the entry and station 45 is the exit to NGV. Station 46 is the entry to the rotor after mixing of cooling bleed for NGV, station 5 is the rotor exit after work balance, and station 51 is the rotor exit subsequent to mixing of bleed for rotor cooling:

$$N_{\text{LT}} = N_{\text{LC}}$$

$$\frac{W_{46}\sqrt{T_{46}}}{P_{46}} = f\left(\frac{P_{46}}{P_5}, \frac{N_{\text{LT}}}{\sqrt{T_{46}}}\right)$$

$$\frac{\Delta H_{\text{LT}}}{T_{46}}, \eta_{\text{LT}} = f\left(\frac{P_{46}}{P_5}, \frac{N_{\text{LT}}}{\sqrt{T_{46}}}\right)$$

$$\text{WORK}_{\text{LT}} = W_{46} \cdot \Delta H_{\text{LT}} = \text{WORK}_{\text{LC}}$$

Bypass Duct

The entry parameters to the bypass duct get defined from the engine state at the exit of the LP compressor. It is assumed to be an adiabatic duct with pressure loss caused by friction being present. Only a fraction of the bypass flow mixes with the core flow in the main mixer. The remaining air is gradually introduced in the core flow for afterburner and nozzle cooling. The following equations describe bypass duct modeling where ΔP_{DUCT} is the pressure loss in the bypass duct, TLF is the tip loss factor (from the presence of the LP compressor at the bypass duct entry), BPR is the bypass ratio, station 21 is the LP compressor exit, station 13 is the entry to the bypass duct, and station 16 is its exit in mixer.

$$P_{13} = P_{21} \times \mathrm{TLF} \qquad T_{13} = T_{21}$$

$$P_{16} = P_{13} \times (1 - \Delta P_{\mathrm{DUCT}}) \qquad T_{16} = T_{13}$$

$$W_{13} = W_{21} \times \mathrm{BPR}/(1.0 + \mathrm{BPR})$$

$$W_{16} = W_{13} \times \text{fraction mixing with hot flow}$$

Main Mixer

The main mixer is the component in which the specified fraction of the bypass flow mixes with the core flow, after ensuring that static pressure of bypass flow is at least equal to that of hot flow at the entry plane. The mass flow, enthalpy, and momentum balance are enforced to determine the mass flow, total temperature, and total pressure at the exit of the main mixer. The assumption that static pressure rise is uniform in the jet pipe has been made, i.e., mixer exit (station 61) static pressure is the same as that of hot flow at its entry (station 6).

Afterburner

The afterburner is another combustor where fuel is burned to raise the enthalpy of the gases. Station 62 is the entry and station 63 is the exit. The procedure for calculating the afterburner exit temperature for a given fuel flow or vice versa is the same as that for the main combustor. The cold loss in the duct is taken as a constant percentage and exists irrespective of whether the afterburner is on or off. The hot loss is calculated based on Raleigh's equation, where total pressure loss is a function of the inlet and outlet Mach number and the ratio of specific heats. The outlet Mach number is calculated first, knowing the total temperature rise in afterburner and then the hot loss:

$$\frac{P_{\mathrm{out}}}{P_{\mathrm{in}}} = \left(\frac{1 + \gamma M_{\mathrm{in}}^2}{1 + \gamma M_{\mathrm{out}}^2}\right)^2 \left\{\frac{1 + [(\gamma - 1)/2] M_{\mathrm{out}}^2}{1 + [(\gamma - 1)/2] M_{\mathrm{in}}^2}\right\}^{\gamma/(\gamma-1)}$$

Convergent Exhaust Nozzle

The following equations define the modeling of a convergent nozzle. Station 7 is the entry to the nozzle. The net thrust is obtained by subtracting the inlet momentum from the gross thrust:

$$PR_{\mathrm{critical}} = \left(\frac{2}{\gamma + 1}\right)^{\gamma/(\gamma-1)}, \qquad CD_{\mathrm{NZ}} = f(PR, \theta)$$

for the choked nozzle ($PR_{\mathrm{NZ}} \geq PR_{\mathrm{critical}}$)

$$\frac{W_7 \cdot \sqrt{T_7}}{A_{\mathrm{NZ}} \cdot CD_{\mathrm{NZ}} \cdot P_7} = \sqrt{\frac{g \cdot \gamma}{R}} \cdot \left(\frac{2}{\gamma + 1}\right)^{(\gamma+1)/[2(\gamma-1)]}$$

$$\frac{F_G}{W_7 \cdot \sqrt{T_7}} = \sqrt{\frac{R}{\gamma \cdot g}} \cdot \left[\sqrt{2(\gamma - 1)} - \left(\frac{\gamma + 1}{2}\right)^{(\gamma+1)/[2(\gamma-1)]} \cdot \frac{p_0}{P_7}\right]$$

for the unchoked nozzle ($PR_{\mathrm{NZ}} < PR_{\mathrm{critical}}$)

$$\frac{W_7 \cdot \sqrt{T_7}}{A_{\mathrm{NZ}} \cdot CD_{\mathrm{NZ}} \cdot P_7} = \left(\frac{p_0}{P_7}\right)^{1/\gamma} \sqrt{\frac{g \cdot \gamma}{R(\gamma - 1)} \left[1 - \left(\frac{p_0}{P_7}\right)^{(\gamma-1)/\gamma}\right]}$$

$$\frac{F_G}{W_7 \cdot T_7} = \sqrt{\frac{2 C_P J}{g} \left[1 - \left(\frac{p_0}{P_7}\right)^{(\gamma-1)/\gamma}\right]}$$

Convergent–Divergent Nozzle

For an HP ratio engine, nozzle pressure ratio is large enough to warrant the use of a convergent–divergent nozzle for better performance. Its modeling is based on the description of Palsane et al.[4] The convergent–divergent nozzle operation is categorized into four cases based on nozzle pressure ratio and ratio of exit area to throat area (AR). The availability of functional relationship that defines AR in terms of throat area is assumed. The standard gasdynamic equations are used for isentropic flow computations.

The first critical pressure ratio (subsonic operation as venturi) is defined as

$$(PR)_{1\,\mathrm{critical}} = f(AR)$$

The second critical pressure ratio (normal shock at exit), for supersonic operation, is calculated as follows. M_x is the Mach number before normal shock, and p_x and p_y are static pressures before and after the normal shock, respectively:

$$M_x = f(AR)$$

$$p_y/p_x = f(M_x)$$

$$(PR)_{2\,\mathrm{critical}} = (p_y/p_x) \cdot (PR)_{3\,\mathrm{critical}}$$

The third critical pressure ratio (isentropic supersonic expansion in divergent portion), as in the previous equation, is defined as

$$(PR)_{3\,\mathrm{critical}} = f(AR)$$

The convergent–divergent nozzle operation is now modeled as follows, where $A^* = A_{\mathrm{throat}}$.

1) Venturi operation: isentropic expansion in a convergent portion and an isentropic compression in a divergent portion, resulting in subsonic flow. This condition arises when $PR \geq (PR)_{1\,\mathrm{critical}}$:

$$\frac{A}{A^*} = f(PR)$$

$$A^* = \frac{AE}{A/A^*}$$

$$M^* = f(PR)$$

AE is the exit area.

2) Normal shock in divergent portion:

$$(PR)_{1\,\mathrm{critical}} > PR > (PR)_{2\,\mathrm{critical}}$$

The exit Mach no. is computed as

$$M^* = f(PR \cdot AR)$$

3) Normal shock at exit:

$$PR = (PR)_{2\,\mathrm{critical}}$$

The exit Mach no. is calculated as follows, where VR is the velocity ratio at the exit to that at the throat:

$$VR = f(M_x) \quad \text{and} \quad M_x^* = f(AR)$$

from which $M^* = M_x^*/VR$.

4) Overexpansion:

$$(PR)_{2\,\mathrm{critical}} > PR > (PR)_{3\,\mathrm{critical}}$$

5) Underexpansion:

$$PR < (PR)_{3\,\text{critical}}$$

No distinction is made between the over- and underexpansion cases. The static pressure at the nozzle exit is calculated and used for pressure thrust computation, as given next where P_j is the total pressure at the nozzle entry. The static pressure at the nozzle exit p_{exit} is equal to the ambient pressure for the overexpansion case:

$$p_{\text{exit}} = P_j \cdot (PR)_{3\,\text{critical}}$$

$$M^* = f(AR)$$

$$F_P = AE \cdot (p_{\text{exit}} - p_0)$$

The nozzle flow and gross thrust are calculated as follows:

$$W = P_j \cdot A^* \cdot CD_{\text{NZ}} \sqrt{\frac{g \cdot \gamma}{R \cdot T}} \left(\frac{2}{\gamma + 1}\right)^{(\gamma+1)/[2(\gamma-1)]}$$

$$V_{\text{exit}} = C_v \cdot M^* \sqrt{\frac{2\gamma g \cdot R \cdot T}{\gamma + 1}}$$

where

$$C_v = f\left(AR, \frac{1}{PR}\right), \qquad F_G = \frac{W \cdot V_{\text{exit}}}{g} + F_P$$

General Modules

The general-purpose modules define the variation in specific heat with temperature and fuel–air ratio, effective fuel heating value, computation of flow parameters, flow velocity, static temperature and pressure, etc. These modules are an important supplement to the simulation code and are repeatedly used during the computations:

$$H, C_P = f(T, \text{FAR})$$

$$\text{EHV} = f(T_4)$$

$$\frac{W\sqrt{T}}{A \cdot P} = f\left(\frac{P}{p}, \gamma\right)$$

$$\frac{T}{t}, \frac{P}{p}, \frac{V}{\sqrt{T}} = f\left(\frac{W\sqrt{T}}{A \cdot P}, \gamma\right)$$

Atmosphere

The atmospheric module makes available standard atmospheric temperature and pressure as a function of altitude. The temperature deviation, if any, from standard day conditions is incorporated via the term DT_{amb}.

$$p_0 = f(ALT)$$

$$t_0 = f(ALT, DT_{\text{amb}})$$

Solution Methodology

At a given flight point and at specified power setting, the engine is assumed to reach equilibrium or steady state, if the following seven errors are satisfied within a prespecified tolerance band.

1) ERR1: flow error between the LP and HP compressors.
2) ERR2: flow error at HP turbine entry, i.e., mismatch between upstream flow function value and that obtained from the map.
3) ERR3: work error between the compressor and turbine on the HP spool, i.e., work demanded by HP compressor and that supplied by HP turbine, the value of which is obtained from the map.
4) ERR4: flow error at LP turbine entry.
5) ERR5: work error between the compressor and turbine on the LP spool.
6) ERR6: static pressure error between the static pressures of the cold and hot streams at the main mixer entry plane.
7) ERR7: nozzle area error, i.e., nozzle area to pass the upstream flow and that specified from control schedules.

The errors are the dependent variables and constitute the system of equations that must be solved to identify a certain set of independent variables that brings the errors within the prespecified tolerance band. A total of seven independent variables are needed. An initial value is guessed for each of the independent variables and they are continuously updated (or iterated) until the engine attains equilibrium. The system of independent (or iteration) variables and the error to which they are linked is as follows:

HP compressor speed ⇔ ERR1
Turbine entry temperature ⇔ ERR2
HP turbine pressure ratio ⇔ ERR3
HP compressor pressure ratio ⇔ ERR4
LP turbine pressure ratio ⇔ ERR5
Fan/LP compressor pressure ratio ⇔ ERR6
Bypass ratio ⇔ ERR7

The preceding procedure, when each iteration variable is linked to one specific error and satisfaction of only one error is attempted at a time, results in the nested-loop approach to determine engine equilibrium. An alternate approach is the multidimensional Newton–Raphson (MDNR) method, in which the system of equations defining the errors is solved to simultaneously update all of the iteration variables simultaneously.

The nested-loop approach is computationally more time consuming as compared to the MDNR scheme, but it still has been used in the present work because it is numerically more stable. It gives complete insight into the errors convergence pattern while tracing the path to reach equilibrium. It is possible to identify those errors that may not converge in the specified tolerance band. A slight relaxation of the tolerance band of such errors is only possible to ensure the convergence, thereby making the nested-loop approach more robust for engine performance prediction.

The MDNR method requires a good starting guess of engine state variables to provide fast convergence and to be numerically stable. It is possible only after a sufficient insight is gained into the engine behavior and its cycle matching aspects. In the early stages of a new engine development program, when little is known about the engine behavior, the nested-loop approach is more suited to understand the engine system. Subsequently, it can be replaced with the MDNR method.

Also, the MDNR technique may be more suitable in the iterative approach to engine transient performance prediction.[5] The intercomponent volumes to generate pressure dynamics are not used and instead it is assumed that there is no mass flow mismatch at any of the engine stations, even during the transients. Only work or power imbalance is used to compute the rotor accelerations. The use of rotor dynamics alone, together with a suitable numerical integration scheme to update engine response with time, enables marching with larger time steps during the transient computations. The use of the MDNR scheme will provide a faster convergence to arrive at the conditions of flow compatibility. All of these features combined, will increase the probability of a real-time computation of engine dynamic response, without any simplification in model equations. The real-time simulation is a valuable tool in engine control system development.

Digital Simulation

The digital simulation of the nested-loop approach has been done using Fortran, on a personal computer. For the given flight point, computations start with intake, which defines the total temperature and total pressure at the entry to the fan or LP compressor. The PLA position dictates the LP compressor speed, which together with total temperature at its entry, defines the corrected speed. A value of the LP compressor pressure ratio is guessed, which enables the computation of LP compressor performance.

A value of bypass ratio is guessed, which then splits the LP compressor flow into the bypass and core flow. The leakage bleed, if any, must be accounted for. The HP compressor speed and its pressure ratio are guessed, which together with entry total temperature and total pressure conditions, enables the computation of HP compressor performance. The ERR1 is generated and HP compressor speed is iterated upon to balance the ERR1.

Based on the guessed value of T_4, fuel flow in the main combustor is computed. The air mass flow for HP turbine NGV cooling is mixed, resulting in a total temperature at the HP turbine throat upon enthalpy balance. The speed of the HP turbine is the same as that of the HP compressor, which together with the entry total temperature and a guessed value of the HP turbine pressure ratio enables the computation of the HP turbine performance using the maps. ERR2 and ERR3 are generated and corresponding variables are iterated to balance them. The temperature at the exit of the HP turbine rotor is computed after doing an enthalpy balance with the flow that mixes for its cooling. The conditions at the HP turbine exit define the entry conditions to the LP turbine. Similar to HP turbine, the LP turbine performance is computed and ERR4 and ERR5 are balanced.

The total pressures of the cold and hot flow in the mixing plane are obtained after accounting for the total pressure loss in the bypass duct and the exhaust cone. Using the specified gas flow path areas of the cold and hot flow at the mixing plane, the static pressures are computed and the LP compressor pressure ratio is iterated upon until the ERR6 is balanced.

The mass flow, total pressure, and total temperature are computed at the mixer exit, from which conditions at the entry to the nozzle are computed. The ERR7 is generated and balanced by iterating upon the bypass ratio. The enthalpy rise in the reheat combustion chamber must be accounted for (using the reheat fuel flow or reheat temperature), prior to the nozzle performance calculation, if the PLA setting corresponds to the reheat mode.

For the case of a convergent nozzle, options of the user-specified nozzle area, the nozzle area from control schedules, and the variable area nozzle have been incorporated to make the simulation code more versatile. Instead of a convergent nozzle, a convergent–divergent nozzle can be used to assess the performance benefits arising out of its use.

In case the control schedules are not available, the LP compressor speed is user defined and the nozzle area is assumed to be fully variable. For each value of bypass ratio in a certain predefined range, engine performance is computed for every speed line of the LP compressor. ERR7 is not generated and nozzle area matching is bypassed. The value of the bypass ratio and corresponding nozzle area, for which engine operation is most suitable at the specified flight condition, is identified as the operating point at a chosen corrected speed line. The operating points are identified for the other speed lines in a similar fashion. The option of a variable area nozzle and a user-defined LP compressor speed thereby permits the generation of steady-state control schedules.

Validation

The validation of the simulation model is best done by comparing the simulation results with actual results from engine testing. Because engine testing results are not available, the comparison is made with steady-state performance, obtained from an alternate formulation based on the MDNR scheme.[6] The results compare within ±2%. Because the two different approaches to steady-state simulation result in numerical data sets that compare fairly well, it can be stated with confidence that the mathematical modeling and computer simulation of a nested-loop approach to predict engine steady-state performance is correct.

Typical Results

The engine uninstalled thrust and specific fuel consumption (SFC) in max dry mode at various altitude and flight Mach number conditions are shown in Figs. 2 and 3. It is only to illustrate the typical results that are obtained from the use of a simulation model. The controller was used to define the LP compressor speed and nozzle area at various flight points. The similar set of results can be obtained in a max reheat mode and part power dry and reheat mode.

In another set of results in a max dry mode, a controller was not used. The LP compressor speed is assigned its design value at all of the flight points. The results are presented in Figs. 4 and 5, for a reference bypass ratio of 0.17. Because the bypass ratio takes a fixed value, ERR7 is not generated and the nozzle area is obtained in the downwash. Similarly, engine performance can be evaluated at various power settings in the dry as well as reheat mode. The performance estimation in this form facilitates the development of steady-state control schedules.

Redefining Engine Cycle

During an engine development program, there may be shortfalls in the sense that actual engine hardware during testing is

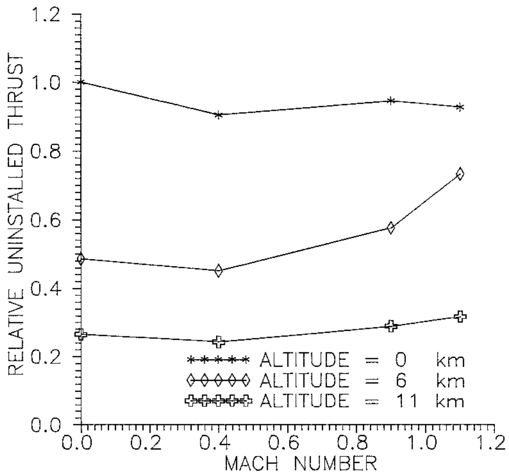

Fig. 2 Engine thrust in max dry mode with controller.

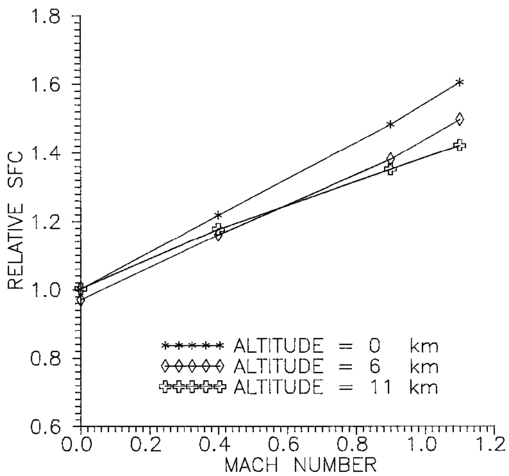

Fig. 3 Engine SFC in max dry mode with controller.

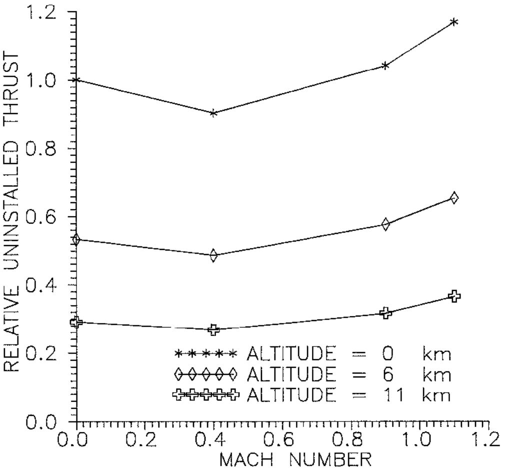

Fig. 4 Engine thrust in max dry mode with user-defined controls.

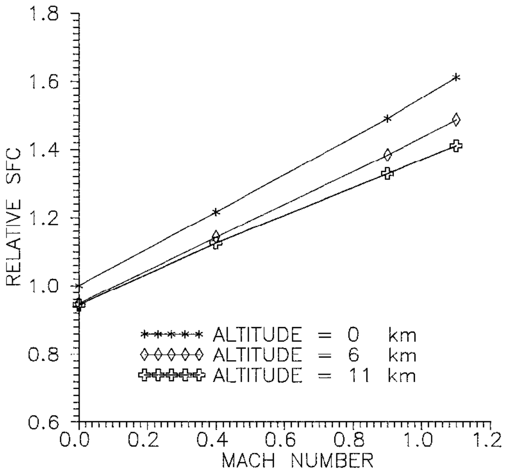

Fig. 5 Engine SFC in max dry mode with user-defined controls.

not able to achieve the theoretically intended performance. As an example, a case study where the HP compressor was observed to be deficient in mass flow capacity is considered. The steady-state digital simulation code is used to investigate the possibility of identifying alternate engine cycles that meet performance goals, without requiring major modifications in engine hardware.

The alternate engine cycles should aim at HP compressor operation, which can attain the originally intended pressure ratio at a reduced mass flow. It is equivalent to a transformation in compressor maps, where for given pressure ratio at a given corrected speed line, the resulting flow function value is reduced by a certain percentage between 5 and 10%. The engine cycle is then matched with reduced HP compressor flow. The control schedules as originally designed were retained during cycle matching.

The impact of the following design changes, either one at a time or in combination on cycle matching, is investigated.

1) The HP compressor flow function as obtained from the map is reduced by a prespecified percentage.

2) In case of efficiency becoming poor, temperature ratio as obtained from the map is reduced by a prespecified percentage.

3) There is a $\pm 2\%$ variation in design values of HP and LP turbine flow function, because it is permissible by changing the throat area, without causing any change in turbine performance characteristics.

The following design changes result in a promising design:

1) HP compressor flow function and temperature ratio are reduced by 5 and 1%, respectively.

2) HP and LP turbine flow functions are reduced by 2% each.

The net effect is to speed up the HP compressor by 2.20% with respect to its original design value. Even at increased speed, the HP compressor flow function is lower by 2.15% from its design value, causing the bypass ratio to increase by 5.25%. The turbine entry temperature and LP and HP compressor pressure ratios are close to their design values. The thrust reduces by 1.50%, with the SFC being near its design value. The current design choice is therefore a promising option to account for degradation in HP compressor mass flow, without major hardware modifications.

Any further reduction in HP compressor flow function increases the turbine entry temperature, thereby reducing the achievable degree of flat rating that penalizes engine off-design performance. The HP compressor speed also increases, which implies that engine also has to tolerate higher levels of stressing, causing a reduction in a components' life.

As a next step, the gas flow path area of cold flow at the mixer entry was also increased by 5% in addition to earlier design changes. The increase in the cold flow area is favorable, because without any significant change in HP compressor speed, its flow function, turbine entry temperature, and SFC, it takes the LP and HP compressor pressure ratios and thrust values closer to their original design values. The variations in the hot flow area do not have a significant influence.

In another study, pressure ratios of the LP and HP compressors were kept fixed at their design values, and as a limiting case, an increase of 5% was permitted in the HP compressor speed. The HP compressor flow function was reduced by 10%, and it was ensured that turbine entry temperature takes its design value by adjusting the cold flow Mach number at the mixer entry. The cycle matching results in a decrease of 4.70% in the HP turbine flow function and 5.60% in the LP turbine flow function. The gas flow path area of cold flow at the mixer entry increases 30%. Because the LP compressor pressure ratio is forced at its design value, the nozzle area was made variable. The matched cycle point results in a decrease of 3.50% in the nozzle area. Any change in the nozzle area implies redefining the control schedules, which coupled with large changes in the turbines' flow function and mixer geometry at the cold flow entry station requires a major design effort.

Thus, if the degradation in a HP compressor mass flow is of the order of 5%, with respect to its original design value, then only redesigning the engine cycle with modifications in the existing hardware is possible; otherwise, it calls for a new definition of engine cycle and associated hardware.

Conclusions

The significance and, hence, the need for a simulation code for propulsion system steady-state performance prediction has been identified. The mathematical modeling and information flow logic, for the digital simulation of a twin-spool mixed-flow turbofan engine, to estimate its steady-state performance using component maps and control schedules, has been described. The simulation code is easily adaptable to the twin-spool turbojet simulation and for the condition when control schedules are not available and need to be developed.

The simulation code is an extremely useful design tool to assess the capabilities of the propulsion system and, hence, the aircraft to which it is integrated over the entire aircraft flight envelope. The simulation is also very valuable in performing preliminary feasibility studies to determine if the alternate cycles are permissible with minimum modifications in the existing hardware, to account for the loss in a certain components' performance and their suitability for use with the aircraft.

References

[1]Sanghi, V., "Computer Aided Conceptual Design of Propulsion System for Combat Aircraft," Ph.D. Dissertation, Indian Inst. of Technology, Bombay, India, 1996.

[2]"Gas Turbine Engine Steady State Performance Presentation for Digital Computer Programs," Society of Automotive Engineers, Aerospace Recommended Practice, 681, 1989.

[3]Cohen, H., Rogers, G. F. C., and Saravanamutto, H. I. H., "Gas Turbine Theory," 3rd ed., Longman Scientific & Technical, London, 1987.

[4]Palsane, S., Lakshmanan, B. K., and Sundararajan, V., "Convergent Divergent Nozzle Modeling for Kaveri Engine," Gas Turbine Research Establishment, ITM/ES-8709, Bangalore, India, April 1994.

[5]Fawke, A. J., and Saravanamutto, H. I. H., "Digital Computer Methods for Prediction of Gas Turbine Dynamic Response," Society of Automotive Engineers, Paper 710550, June 1971.

[6]Palsane, S., Lakshmanan, B. K., and Sundararajan, V., "Design Document on Kaveri Engine Simulation," Gas Turbine Research Establishment, ITM/ES-8704, Bangalore, India, Nov. 1991.