

MODELING PROPULSION SYSTEM OF A SUBSONIC CIVIL JET AIRCRAFT FOR MULTIDISCIPLINARY CONCEPTUAL DESIGN OPTIMIZATION

Vivek Sanghi* and S.K. Sane†

*Engine Simulation Group, Gas Turbine Research Establishment, Bangalore-560 093

†Department of Aerospace Engineering, Indian Institute of Technology, Mumbai-400 076

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Abstract

The integrated modeling of multidisciplinary engine-airframe interactions over the prescribed mission is an essential pre-requisite for performing system analysis during an engine or an aircraft conceptual design optimization.

This paper first highlights the significance of propulsion system's steady-state performance to this conceptual system analysis, and then describes a generic rubber engine model for its digital simulation, which is consistent with conceptual design requirements and accuracy. It doesn't use components' characteristics, and operates on simplifying assumptions like choked turbines and constant components' efficiency at all operating conditions etc. to supplement the lack of sufficient information during a conceptual design phase. The model results compare fairly well with a typical manufacturer's data available in open literature, thereby confirming its adequacy.

The focus of present work is on propulsion systems for a subsonic civil jet aircraft, but it can be readily adapted to simulate the performance of propulsion systems for other aircraft applications as well.

1 Introduction

The primary objective of an aircraft conceptual design process is to parametrically investigate the sensitivity of important design variables on vehicle synthesis, so as to arrive at a

configuration that is not only feasible, but also best satisfies a prescribed mission in terms of performance and cost-effectiveness. Keeping in view the long development period and the high cost of ownership of an aircraft system, the emphasis is on an optimum and competitive design, and not just a suitable design.

An aircraft comprises of two main sub-systems, namely the propulsion (or the engine unit) and the airframe. Each sub-system is defined by a set of design variables, and assigning a numerical value to each of them creates an aircraft design option.

A conceptual system analysis (CSA) is central to the optimization studies. For a given design option, the numerical value of engine design point mass flow ($W_{ENG,DP}$) fixes the maximum installed thrust that is available from engine at all mission operating conditions. The CSA first performs a drag/thrust matching to determine if the installed thrust demand of chosen design at current mission segment in prescribed power setting is met. If yes, it computes the fuel consumed during the mission leg as product of engine installed thrust, thrust specific fuel consumption (TSFC), and mission segment flight time. If thrust demand of any mission segment exceeds the maximum available installed thrust, that aircraft design option is deemed infeasible. The final outcome for a feasible design is the mission-matched value of aircraft take-off gross weight (W_{TO}).

Summarizing, CSA evaluates the feasibility of flight, initial size, and the performance of an aircraft design option over the prescribed mission. It involves an exchange of information between various disciplines like the aircraft equations of motion, airframe weight and aerodynamic characteristics, and engine steady-state thermodynamic performance. CSA is therefore an integrated assessment of multidisciplinary engine-airframe-mission interactions, and needs to be supported by digital computers to facilitate a rapid simulation of complicated logic flow without any or minimal simplification.

The multidisciplinary design optimization (MDO) techniques are now used to identify that feasible aircraft design option, which is optimum over the prescribed mission. Utilizing response from CSA, a non-linear constrained optimization problem is solved to locate an n -dimensional vector of design variables, that is, $X = (x_1, x_2, \dots, x_n)$, while simultaneously minimizing (or maximizing) a certain figure of merit (FOM), which is a measure of design optimality. Typically, minimization of W_{TO} has been used as the FOM because a smaller aircraft costs less to build and operate.

An aircraft can be optimized either around a fixed engine cycle, or else some of the important engine cycle variables can also be included in the design vector being optimized. Alternatively, an engine cycle can be optimized around a known or a given class of airframe. References [1-11] give a good understanding of the issues and techniques for engine as well as aircraft concept optimization. Whatever may be the case, CSA is an essential pre-requisite, and it can be performed only if propulsion systems' steady-state performance (that is, installed thrust and TSFC), at desired operating conditions is made available.

The steady-state performance is computed by thermodynamic relationships, after determining the engine equilibrium. It starts with design point analysis. The design point is the condition for which an engine is designed, that is, a certain altitude (H), Mach number (M), atmosphere type (for example, the international

standard atmosphere, ISA), and the area settings of any variable geometry features. The data generated during the design point analysis is an input to off-design analysis, which then determines the performance at conditions other than design point.

Like the CSA, an explicit analytical model for engine performance estimation is also not possible due to the highly non-linear and mutually coupled behavior of its components, thereby requiring its digital simulation. Summarizing, availability of a digital computer model is an essential requirement to estimate steady-state performance of a propulsion unit. This propulsion model should cover engine operation from max to idling (or near idling) power settings at every H/M within the flight envelope, to ensure that performance at all the mission flight points is available.

The numerical MDO techniques search through a large sequence of non-optimal design options, prior to locating the candidate optimum design(s). Thus, propulsion model should provide a quick performance estimate for rapid evaluation of all the intermediate non-optimal design options, and should be numerically robust to prevent the failure of system optimization. Besides, conceptual design is a critical activity, since it determines the final quality and entire course of the design process. It requires that propulsion model must be fairly accurate to ensure the accuracy of CSA, so that designer is able to attach adequate confidence in the selection of initial design(s).

This paper presents a generic rubber engine model for digital simulation of propulsion system performance, which is consistent with conceptual MDO requirements as outlined earlier. The validation issues of this model have also been discussed to highlight its accuracy. A similar modeling approach for military engines is described in [12]. In fact, the propulsion model contained in [12] formed the starting point, which was suitably modified to simulate the performance of propulsion systems for a subsonic civil jet aircraft.

2 Propulsion Concepts

The commonly used propulsion concept for a subsonic civil jet aircraft is a high bypass, twin spool turbofan engine. The air entering the engine splits into the core- and bypass-streams at fan exit. While the bypass-stream passes through an annular bypass duct, the core-stream goes through the booster or low-pressure (LP) compressor, core or high-pressure (HP) compressor, combustor, and HP and LP turbines. The two streams are then exhausted from separate nozzles in the unmixed flow (or separate-exhaust) configuration. Alternatively, they can be mixed downstream of LP turbine, prior to expansion in a common exhaust nozzle.

Irrespective of mixed- or separate-exhaust configuration, the single stage fan is followed by a LP compressor, comprising of two to four stages. The fan and LP compressor are on the same shaft, driven together by the LP turbine. The multi-stage HP compressor is driven by the HP turbine. Fuel is injected in the combustor and burned to produce hot gas for driving turbines. There is no reheat combustor.

3 Engine Cycle Definition

A propulsion concept is defined in terms of cycle design variables, which are classified as the primary and the secondary design variables. Assigning a numerical value to each of them creates an engine cycle within the chosen propulsion concept.

The primary cycle variables are the ones that have high sensitivity on system response and should be included in the design vector being optimized. The primary cycle variables for a twin-spool, civil turbofan engine are $W_{ENG,DP}$, bypass ratio (BPR), fan pressure ratio (FPR), LP compressor pressure ratio (PRLC), overall pressure ratio (OPR), maximum turbine entry temperature (TET_{max}), and throttle ratio ($TR = TET_{max}/TET_{DP}$). The subscript DP denotes the design point. Knowing OPR, FPR, and PRLC, the pressure ratio of HP compressor (PRHC) gets defined automatically since OPR is the product of FPR, PRLC, and PRHC.

For the specific case of a mixed-stream turbofan, the core- and bypass-stream Mach numbers at the mixer inlet ($M_{c,DP}$, and $M_{h,DP}$) can also be used as primary variables. However, it is shown in [13] that for optimum mixing, the $M_{c,DP}$ and $M_{h,DP}$ should be nearly equal. Thus only $M_{h,DP}$ (or $M_{c,DP}$) may be used as a primary cycle variable, which lies between 0.40 to 0.50. Choosing such a condition will also eliminate FPR from the primary design variables set, as optimum mixing will always lead to an optimum FPR, for the prescribed values of remaining cycle variables.

The remaining variables like component's efficiency, total pressure loss, power extraction, customer and cooling bleeds etc. are referred as secondary variables. It is always advantageous to maximize components' efficiency, and minimize pressure loss, power extraction, and bleeds. Thus, their values are kept fixed as per the state-of-art, instead of being optimized in a conceptual design study.

4 Salient Modeling Features

Due to the lack of sufficient design information in an early conceptual design phase, the methods that operate without components' maps are used. Also, it would not be possible to generate component maps for every intermediate non-optimal point while locating the optimum using numerical MDO techniques.

In absence of rotational speeds, corrected mass flow at fan entry ($W_{1,cor}$) is used as a control variable to modulate engine power during off-design performance prediction. At a given H, as M increases, $W_{1,cor}$ is maintained constant at its design point value (which is equivalent to constant fan corrected speed) by allowing TET to increase. If TET exceeds its limiting value, the $W_{1,cor}$ is reduced till TET equals its limiting value. To move from max to part power setting, $W_{1,cor}$ is reduced.

The HP and LP turbines are assumed to be operating choked at all operating conditions. The components' real behavior is defined by isentropic efficiency terms, the design values of

which are user specified. In absence of component maps, it is assumed that off-design values of components' efficiency take their design values.

The various ducts (like intake, fan-LP compressor interconnecting duct, bypass duct, jet pipe etc.) are modeled as adiabatic components, that is, total temperature (T) across them is constant, and loss in total pressure (P) due to fluid friction is accounted by appropriate loss coefficients. An additional loss coefficient is used for the combustor to model the loss in total pressure due to fluid friction as well as heat addition. These pressure loss coefficients take their respective user defined design values at all off-design points.

The losses due to windage, bearing friction etc. in components that transmit mechanical power by means of a shaft are accounted by a mechanical efficiency term. It is assigned a fixed numerical value for the LP as well as HP spool, and is usually of the order of 0.99. A small proportion of HP turbine power is tapped to drive engine and aircraft accessories. The HP and LP turbines power should be appropriately adjusted to include these effects in work balance equation between the compressor and turbine on the same shaft.

A small percentage of airflow is tapped from the core- and bypass-streams for cooling the hot parts, aircraft cabin pressurization, and de-icing some of the wing and nacelle surfaces. For the mixing of cooling air, only the mass flow and enthalpy balances are done. The momentum balance is ignored, and hence the total pressure upon mixing remains unchanged. However, during the mixing of core- and bypass-streams in a mixed-stream turbofan, momentum balance is also performed to compute the total pressure of mixed-out stream. Besides, mixing of core- and bypass-streams is taken to be complete and ideal.

The exhaust nozzle(s) are assumed to be fixed geometry convergent nozzles. A velocity coefficient term is used to simulate the real behavior of an exhaust nozzle, which takes its design value at all off-design points. The specific heat at constant pressure (C_p) is an

important requirement during cycle calculations. It is computed as; $C_p=f(T,FAR)$, where FAR is the fuel-to-air ratio.

5 Design Point Uninstalled Performance

Sea-level static in international standard atmosphere is taken as the engine design point. The primary and secondary cycle variables at the design point are user specified. TET_{DP} is obtained as; $TET_{DP}=TET_{max}/TR$, where TET_{max} and TR are primary cycle variables.

The components' performance is sequentially evaluated, according the layout of engine configuration, and conditions of engine equilibrium are imposed. Work compatibility determines the turbines' work and hence the pressure drop across them. The mass flow compatibility determines the areas of turbines' nozzle guide vanes, and the exhaust nozzle area. For the specific case of mixed-stream turbofan, static pressure equality between the core- and bypass-streams at the mixer inlet plane also needs to be satisfied. The relevant gas-dynamic equations and mathematical expressions for evaluating components' performance and final thrust and TSFC are readily available in open literature [14,15], and not reproduced here.

The estimation of design point performance involves an iteration loop for the mixed-stream turbofan. For a given cycle, FPR is continuously iterated till static pressures of the core- and bypass-streams become equal. If nearly equal $M_{c,DP}$ and $M_{h,DP}$ are used, the resulting FPR shall be the optimum. In case of a separate-exhaust turbofan, this iteration loop is bypassed. However, here also, it is possible to determine an optimum FPR. The FPR for a separate-exhaust turbofan shall be the optimum when the velocity ratio between the cold- and hot-streams is equal to the product of fan efficiency, LP compressor efficiency, and the LP spool mechanical efficiency. This condition can be achieved by setting up an iteration loop over FPR. Thus, although a primary cycle variable, FPR need not be included in the design vector as its optimum value can be identified during the design point cycle analysis itself.

6 Off-Design Uninstalled Performance

The engine cycle variables are decided at the design point, and can no longer be chosen independently at off-design points. To compute the steady-state performance of this specific design choice at an off-design point (that is, a certain H, M, and power setting in a specified atmosphere with temperature deviation DT_{amb} from a standard day), estimates of primary cycle variables, namely (i) mass flow (W_1), (ii) BPR, (iii) FPR, (iv) PRLC, (v) PRHC, and (vi) TET are required.

The engine is assumed to reach steady state if the following five errors are satisfied within a prescribed tolerance band ($\pm 0.001\%$). The LP spool work balance is imposed.

- (1) ER1: HP turbine flow error. It is computed as mismatch between upstream T_{42} and the one from HP turbine choked assumption, where station 42 is at nozzle guide vanes (NGV) of HP turbine, after mixing of cooling bleeds.
- (2) ER2: HP spool work imbalance. HP turbine work is computed based on the condition that its pressure ratio (PRHT) takes design value since LP turbine is assumed choked.
- (3) ER3: Static-pressures mismatch between core- and bypass-streams at mixer inlet if mixed-stream turbofan. Else, bypass nozzle entry mass flow error (that is, mismatch between nozzle area required to pass the upstream flow and design point nozzle area).
- (4) ER4 :Core nozzle entry mass flow error.
- (5) ER5 : Limiters violation (the conditions $TET \leq TET_{max}$ and $OPR \leq OPR_{DP}$ are imposed to ensure a feasible solution).

The errors are the dependent variables and constitute a system of equations that must be solved to identify that set of independent variables which brings the errors within the prescribed tolerance band. For this purpose, a total of five independent variables are needed. An initial value is guessed for each of them, and they are continuously updated using quadratic interpolation, till engine reaches the steady state. The system of independent or cycle variables (to be determined iteratively) and the errors to which they are linked is as follows:

$$\begin{aligned} ER1 &\Leftrightarrow TET \\ ER2 &\Leftrightarrow PRHC \\ ER3 &\Leftrightarrow FPR \\ ER4 &\Leftrightarrow BPR \\ ER5 &\Leftrightarrow W_{1,cor} (W_1 = W_{1,cor} * P_1 / \sqrt{T_1}) \end{aligned}$$

The preceding procedure, where each iteration variable is linked to one error, and satisfaction of only one error is attempted at a time is termed as the nested-loop approach to determine engine steady state. Since LP compressor and fan are on the same shaft, PRLC is computed by assuming enthalpy rise across LP compressor to be proportional to enthalpy rise across fan, and then referencing it to the design point condition to remove the proportionality constant [15]. Thus, it is not required to include PRLC in the iterative scheme, and hence only five (and not six) errors are balanced.

At a given off-design point, the engine performance is evaluated first in max dry mode, and then for a range of part power settings. As stated earlier, $W_{1,cor}$ is the handle to define a part power setting. Its value to simulate various part power settings is chosen as follows:

$$W_{1,cor} = (X_1 \dots X_n) * W_{1,cor,max \text{ dry}}$$

where $0.7 \leq X_n < X_{n-1} < \dots < X_2 < X_1 < 1.0$

A simple guess vector is defined to provide an intelligent starting estimate for each iteration parameter during cycle matching in max dry mode. Upon convergence, off-design performance is evaluated in the first part power mode, where converged values of iteration parameters in the max dry mode are used as their initial guess for cycle balancing. The converged values of iteration parameters in the first part power are subsequently utilized to initiate the cycle balancing in second part power mode. This sequence is continued till all the part power modes are evaluated. Such a procedure speeds up the convergence, and makes it numerically robust.

7 Information Flow Logic

This section gives a comprehensive description of information flow logic based on nested loop approach for engine cycle balancing at an off-

design point in the max and part power modes, which can easily be implemented on a digital computer. The extraction of cooling and customer bleeds and accessory power is not shown here, which must be appropriately included in the analysis.

(A) Inputs:

- (1) Design point engine state, geometry, and choking values of HP and LP turbines.
- (2) H, M, atmosphere type, and DT_{amb} .
- (3) Total number of power setting, max plus part power (NPLA).

(B) Sequence of Steps:

- (1) Compute T_{amb} , $P_{amb} = f(H)$. Add DT_{amb} to T_{amb} to get net atmospheric temperature.
- (2) Initialize an integer variable KPLA to 0.
- (3) Increment KPLA by 1 ($KPLA=KPLA+1$).
- (4) Check if $KPLA=1$ (max power mode).
 - Yes: $W_{1,cor}=W_{1,cor,DP}$
 - No: $W_{1,cor}=X*W_{1,cor,ref}$ ($0.7 \leq X < 1.0$)
 $X=[1-0.030*(KPLA-1)]*W_{1,cor,ref}$
- (5) Guess BPR, FPR, PRHC, and TET.
- (6) Compute Intake performance.
- (7) Compute Fan performance.
- (8) Compute LP compressor performance.
- (9) Compute HP compressor performance.
- (10) Estimate fuel mass flow rate for prescribed TET, and total pressure and mass flow at combustor exit.
- (11) Perform mass flow and enthalpy balance between core-stream and bleed for cooling of HP turbine NGV to get T_{42} . Obtain another T_{42} using design point HP turbine choking value.
- (12) Is ER1 within prescribed tolerance band:
 - Yes: proceed.
 - No: Iterate TET. Go to step (10).
- (13) Compute work required from HP turbine from HP spool work balance equation, and work produced by HP turbine (based on $PRHT=PRHT_{DP}$ condition since LP turbine is assumed choked and has fixed geometry).
- (14) Is ER2 within prescribed tolerance band:
 - Yes: compute HP turbine rotor exit conditions (that is total temperature, total pressure, and mass flow), and proceed.
 - No: Iterate PRHC. Go to step (9).

- (15) Perform mass flow and enthalpy balance between core-stream and bleeds for cooling of HP turbine rotor and LP turbine NGV.
- (16) Impose LP spool work balance, and compute temperature drop across LP turbine, and its pressure ratio (that is, LP turbine rotor exit conditions).
- (17) Incorporate LP turbine exhaust cone losses.
- (18) Estimate bypass duct exit conditions, which form entry to the main-mixer in a mixed-stream turbofan, and to the bypass exhaust nozzle in a separate-exhaust turbofan.
- (19) Is it a mixed-stream turbofan cycle:
 - Yes: proceed
 - No: go to step 21.
- (20) For a mixed-stream turbofan only:
 - Compute core- and bypass-streams static pressures (using mixer geometry from design point calculations)
 - Is ER3 within prescribed tolerance band.
 - If yes, perform mass flow, enthalpy, and momentum balance between core- and bypass-streams to get mixed-out engine state. Else iterate FPR, and go to step (7).
- (21) Is it a separate-exhaust turbofan cycle:
 - Yes: proceed
 - No: go to step 23.
- (22) For a separate-exhaust turbofan only:
 - Compute the bypass nozzle area required to pass the upstream mass flow, and the nozzle exit velocity.
 - Is ER3 within prescribed tolerance band.
 - If yes, proceed. Else iterate FPR, and go to step (7).
- (23) Compute jet-pipe exit (core nozzle entry) conditions.
- (24) For mixed- as well separate-exhaust cycles:
 - Compute the core nozzle area required to pass the upstream mass flow, and the nozzle exit velocity.
 - Is ER4 within prescribed tolerance band.
 - If yes, proceed. Else iterate BPR, and go to step (7).
- (25) If $KPLA=1$, check if limiters are satisfied. If yes, compute thrust and TSFC, and assign $W_{1,cor,ref}=W_{1,cor}$. Else reduce $W_{1,cor}$, and go to step (7).
- (26) If $KPLA < NPLA$, go to step (3), else stop.

8 Installation Penalty

An empirical correlation; {Installation Penalty = $f(\text{flight } M, \text{BPR})$ } is used to compute installed performance [16].

9 Validation

Based on the foregoing description, an engine model to estimate design and off-design installed performance of a twin-spool turbofan, in the mixed- as well as separate-exhaust modes was developed in FORTRAN, and successfully implemented on a personal computer (PC).

The first validation check is to test the accuracy of model construction, which is done by running the design point as an off-design condition. The initial guess for each iteration parameter was taken to be its design value. The off-design analysis converged without any iteration, which is expected since the initial guess values refer to a balanced engine state. The off-design results match with their respective design values within $\pm 0.10\%$.

The next validation check is to test the model accuracy. It is possible only if results from a performance deck (or alternatively manufacturer's data) are available for at least one civil jet engine in service. However, such an information being manufacturer's proprietary, it is classified, and is available in a very limited form only in open domain [17]. An attempt has therefore been made to make the best possible use of this limited information to validate the accuracy of engine model.

The case study of Airbus A320 aircraft with V2527-A5 engine (mixed-stream turbofan) was considered, for which the following data is available in [17].

Engine (cycle values at sea level static in ISA):

$W_{\text{ENG,DP}}=385$ kg/s, $\text{BPR}=4.75$, $\text{OPR}=27.45$, $\text{FPR} = 1.70$ (approximately), and Installed thrust at take-off, sea level, $\text{ISA}=117.8$ kN.

Aircraft:

Empty weight (W_{EMP})= $42,220$ kg., Max fuel weight (W_{F})= $19,159$ kg., Standard internal fuel capacity= $23,859$ liters, $W_{\text{TO}}=73,500$ kg., Wing loading (W_{LDG})= 599.5 kg/m², Range (with

150 passenger, domestic reserves, and 370 km. diversion)= 4874 km., Cruise $H/M=11.0$ km./0.78, Take-off run (sea level, $\text{ISA}+15^{\circ}\text{C}$)= 1950 meters, and Landing run (at max landing weight of $64,500$ kg)= 1490 meters.

Utilizing this limited information, an attempt was made to simulate the mission performance of Airbus A-320 with V2527-A5 engines. The typical civil aircraft mission profile was constructed for Airbus A320, in accordance with description in [18]. The clean aircraft lift and drag characteristics, lift increment due to flaps, and drag increment due to landing gear and flaps, that are representative of an Airbus A-320 type of aircraft were used.

Assuming passenger and baggage weight to be 80 kg., the payload weight (W_{P}) for 150 passengers (plus 2 crew and 4 cabin attendants) works out to be $12,480$ kg. ($156*80=12,480$ kg.). Since W_{TO} is the summation of W_{EMP} , W_{P} , and W_{F} , it results in a W_{F} of $18,800$ kg. (that is, $73,500-42,200-12,480=18,800$), which is consistent with standard internal fuel capacity of Airbus A-320.

As design point FPR , PRLC , TET_{max} , and TR are not known, it is required to make a judicious estimate of their values. V2527-A5 engine has a four-stage LP compressor, and hence it is appropriate to assume a PRLC of 1.80. The FPR is of the order of 1.70, take-off thrust at sea level in ISA is 117.8 kN, and engine is flat rated to $\text{ISA}+15^{\circ}\text{C}$. Thus TET_{max} and TR were chosen as 1600K and 1.03. It results in $\text{TET}_{\text{DP}}=1555\text{K}$, $\text{FPR}=1.68$, and design point installed thrust of 115.4 kN, which seem to be quite in order. The values for secondary cycle variables that are typical of a civil turbofan were assumed.

A total of 48 flight points were defined ($0.0 \leq H \leq 11.5\text{km.} / 0.0 \leq M \leq 0.80$), covering the entire flight envelope of an Airbus A320. The engine performance was evaluated in $\text{ISA}+15^{\circ}\text{C}$, in max and five part power setting, at each of these flight points, and was stored as a look-up table. A linear search and interpolation was used to retrieve performance at any desired mission operating condition. To

give an estimate of computation time, engine model takes about 300 seconds on a Pentium-III for generating performance over all H/M/power setting combinations, which is fairly fast and sufficient for conceptual design studies.

The mission analysis to achieve a range of 4874 km. results in a $W_{TO}=73,825$ kg. and $W_F=19,125$ kg., that is a deviation of 0.45% in W_{TO} and 1.70% in W_F . The take-off run, landing run, and time to climb to cruise altitude are 1910 meters, 1485 meters, and 1720 seconds respectively. The second segment climb gradient (SSCG) is 0.035, and specific excess power (SEP) at top of climb is 1.30 m/s, which satisfy a civil jet aircraft requirements. Keeping in view the approximate nature of analysis, it can be said that mission analysis results are well in agreement with actual aircraft data. This case study provides a good confidence in the correctness of installed thrust and TSFC computations by engine model.

10 A Typical Optimization Case Study

Using the engine model described in this paper and mission analysis as above, a case study was performed (based on optimization with surface fits) to identify if it is possible to arrive at V2527-A5 type of cycle for an Airbus A320. Minimization of W_{TO} was the figure of merit.

The design variables being optimized, design space, and constraints are as follow:

Design variables and design space:

$$4.25 \leq BPR_{DP} \leq 5.50$$

$$1.40 \leq PRLC_{DP} \leq 2.00$$

$$22.0 \leq OPR_{DP} \leq 35.0$$

$$1500K \leq TET_{max} \leq 1650K$$

$$1.00 \leq TR \leq 1.02$$

$$750kg/s \leq W_{ENG,DP} \leq 860.0kg/s$$

$$575kg/m^2 \leq WLDG \leq 650kg/m^2$$

Inequality constraints:

$$\text{take-off run} \leq 1950 \text{ meters}$$

$$\text{second segment climb gradient} \geq 0.024$$

$$\text{SEP at top of climb} \geq 1.50 \text{ m/s (300 ft/min)}$$

$$\text{landing run} \leq 1490 \text{ meters}$$

While locating the optimum solution, the $W_{ENG,DP}$ was kept fixed at 770 kg/s (that is, 385 kg/s per engine) as per the actual design value of V2527-A5. The optimum was obtained at different levels of TET_{max} , and it was at 1600K, that the resulting optimum matched most closely with V2527-A5 cycle. Moreover, the optimum OPR always takes the upper limiting value of its design space, which is 35. But V2527-A5 has a ten-stage HP compressor and it came in operation in early 1990. An OPR of 35 implies a PRHC of about 11, which probably was difficult to achieve at technology level prevailing during that period. Hence, the upper limit of OPR design space was restricted to 28.0, which is close to the design value of 27.40 for V2517-A5.

Within this framework, the optimum values of design variables at the design point are; $BPR=4.77$, $PRLC=1.77$, $OPR=28.0$, $TET_{max}=1600K$, $TR=1.025$, $W_{ENG,DP}=385$ kg/s per engine, and $WLDG=600.25$ kg/m². The system response at the optimum is; $W_{TO}=73770$ kg., $W_F=19070$ kg., take-off run=1945 meters, SSCG=0.035, SEP at top of climb=1.54 m/s, landing run=1490 meters, design point installed thrust and TSFC=116.2 kN and 12.49 mg/N-s, and design point installed thrust loading=0.32. These values are in good agreement with actual V2527-A5 and Airbus A320 data, and hence provide an additional confidence in the adequacy of engine model for use in conceptual design optimization.

11 Method of Component Scaling

An alternative method for performance estimation is the method of component scaling, like the TURBOMATCH code that has been developed and used extensively at the Cranfield University (UK) [19]. Here, representative maps are stored for each component for a range of design values. Now based on the actual design values of the cycle being evaluated, those representative maps that are nearest to the cycle design values (that is, resemble most closely to intended application) are selected and scaled, such that the design values of scaled maps are

same as that of cycle. The scaling factors then remain constant at all off-design points.

Although components' description is used, it can't be stated conclusively that method of component scaling is superior to the one presented in this paper for a conceptual design study. The reason being that there is no certainty whether the scaled component maps are truly representative of the actual ones. The possibility of deviations in performance prediction is also high, if scaling is large. Besides, a number of component maps need to be stored to cover a wide design space, which may be a difficult task. An engine model based on such a method is likely to be computationally time intensive, and prone to numerical instability when used over a wide range of parametric cycle design space and H/M/power setting combinations. However, this technique may be used during preliminary design studies, once the optimum cycle has been finalized upon termination of conceptual design phase.

12 Conclusions

The conceptual design is the earliest design phase that creates an initial concept to optimally meet a given set of requirements. It is characterized by parametric studies, which need to be performed quickly, at relatively low cost, and with good accuracy, notwithstanding lack of sufficient design information. The decisions taken during this design phase shall determine the remaining course of design process, and above all, the final quality of product.

The availability of propulsion systems' steady-state performance is an important prerequisite during an engine or an aircraft conceptual design. A number of works are available in open literature on multidisciplinary conceptual design optimization of engine as well as aircraft systems, but they lack in terms of a comprehensive description of propulsion system model that is consistent with conceptual design requirements. This paper endeavors to bridge this gap. It presents a simple mathematical model for use in a conceptual design study, which can be easily implemented

on a digital computer for estimating steady-state performance of an aircraft propulsion system.

The proposed engine model is basically an extension of the methods already reported in open literature. However, these methods have been assimilated, and appropriately upgraded to bring them to a level where they can be utilized for a conceptual design with good confidence. In the present form, the scope of this engine model is restricted to the mixed- and separate-exhaust turbofans for a civil jet aircraft.

A nested loop approach has been used for cycle balancing, and iteration variables are updated using the quadratic interpolation scheme. A simple guess vector has been built in to provide an intelligent starting value to each iteration parameter, to speed up and ensure the convergence. The model can be easily tailored to obtain performance output (at a number of power settings over each of many H/M points covering the entire flight envelope) for all the parametric cycle combinations in a single execution. Further, for every cycle, the performance output can be presented in a look-up table form, for ease of integration in mission analysis software.

In absence of manufacturer supplied performance data for a typical civil turbofan, the limited information available in open domain has been used to validate engine model indirectly. The mission performance of AirbusA-320 with V2527-A5 engine was simulated. The model results like W_{TO} and W_F compare fairly well with actual aircraft data, which indicates that computation of installed thrust and TSFC are reasonably accurate. An engine cycle optimization (around the Airbus A320 configuration) was also performed which resulted in a cycle that is similar to that of V2527-A5. It provides additional confidence in the adequacy of engine model.

Summarizing, the digital simulation based on this modeling technique is computationally fast, numerically robust, and fairly accurate. It operates satisfactorily over a wide range of cycle parametric combinations as well as over a wide range of H/M/power settings, and can be readily integrated in mission analysis software.

Alternative methods based on scaled component maps can also be used. But keeping in view the conceptual design requirements, they may not be highly reliable in terms of numerical accuracy and robustness.

This paper doesn't address the issues like estimation of engine noise and emission levels, which are increasingly being attached a considerable significance in a civil aircraft conceptual design. It is also required to include a description towards preliminary estimation of the size and weight of an engine cycle. Nevertheless, these deficiencies don't hamper the model capability for use in an early multidisciplinary conceptual design study, where the primary concern is steady-state performance values.

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