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Preliminary Estimation of Engine Gas-Flow-Path Size and Weight

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Subsequent to identifying the optimum engine cycle over a specified baseline design mission, one of the important goals of conceptual design is to translate this into an aerothermomechanically compatible gas-flow-path (GFP) layout and weight estimation. Engine GFP size and weight definition are important design parameters. They define the starting point for detailed component design and the physical limits for integration of the engine with the airframe. This paper describes the mathematical basis of engine GFP sizing and weight estimation, as well as their computer simulation, validation, and application to an optimum engine cycle, which will be typical of the next generation of combat aircraft.

Nomenclature

A = area
AR = blade aspect ratio
C = chord
C/S = blade solidity
D = diameter
H = height
HTR = hub-to-tip ratio
K = spool matching parameter
k = ratio of specific heats
N = number of blades, rotational speed
R = gas constant
T = shaft torque
U = blade speed
W^* = compressor loading parameter
Y^* = turbine loading parameter
Γ = stress in the shaft
η = component efficiency
η_m = mechanical efficiency

Subscripts

B = blade
H = hub
HP = high pressure
HPC = high-pressure compressor
HPT = high-pressure turbine
i = inner
LP = low pressure
LPC = low-pressure compressor
LPT = low-pressure turbine
M = mean
o = outer
T = tip

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Introduction

THE primary objective of conceptual design studies is to identify an optimum engine cycle that will result in an aircraft weapon system that is the most responsive, in terms of cost and performance, to the specified baseline design mission. The selected engine cycle preliminary design can be considered complete only after defining the engine envelope or gas-flow-path (GFP) layout (termed engine sizing) and its weight estimates.

The engine is a coupled system based on interactions between thermodynamic, aerodynamic, and mechanical design variables. Engine sizing begins with cycle analysis at the sizing point, which defines its thermodynamic state, i.e., total temperature T, total pressure P, and mass flow W at every engine station. The sizing point can be the engine design point or an off-design condition, where at least one of the maximum conditions of total temperature, total pressure, and rotational speeds occur. The engine mass flow at the sizing point is determined by the flight condition of that mission segment, which is most constraining in terms of the installed thrust demanded from the engine.

Engine thermodynamic parameters are linked with a large number of aerodynamic and mechanical variables, termed as sizing variables, through equations that define the engine sizing process. The information generated during sizing is input to an empirical database that contains correlations for the weight of various engine components in terms of sizing data. The summation of the weights of all of the components gives the total engine weight.

For a given engine cycle, a large number of engine envelopes and, hence, weight estimates are possible, depending on the numerical values the input sizing variables take. There is no unique solution. Computer simulation is a very useful design aid to create and investigate the relative merits of various GFP designs before a final decision is made. The final GFP layout has minimum weight and acceptable envelope dimensions, meets the constraint of aerothermomechanical compatibility, and uses an achievable technology level. Hence, when translated to actual hardware, it is operationally feasible, delivers the desired thermodynamic performance in terms of thrust and specific fuel consumption, and is suitable for integration with an airframe.

This paper illustrates the procedure for digital simulation of engine GFP sizing and weight estimation, which provides a preliminary set of results consistent with conceptual design accuracy. Validation and application of the resulting simulation code is also presented. The contents of the paper are based on work reported in Sanghi.[1] The mathematical basis of engine sizing is taken from Shlyakhtenko,[2] Sane,[3] and Pera et al.,[4] and weight estimation is based on the description in Pera et al.[4] Sane[3] provides an English translation of the engine sizing aspects discussed in Shlyakhtenko.[2]

The simulation code utilizes a design data bank, i.e., upper and lower limits on each of the sizing variables and various empirical correlations derived from existing design experience, which have limitations if used well outside this experience. The simulation code is therefore suitable for use with twin-spool, mixed-flow turbofan and twin-spool turbojet types of propulsion concepts only, which is sufficient because it is in accordance with current military trends. The existing design data bank and empirical correlations need continuous upgrading to incorporate ongoing advancements in engine design and construction technology. Also, they must be appropriately modified to account for advanced concepts like variable-cycle and thrust-vectored engines.

Mathematical Formulation

A set of governing equations defines the mathematical basis of engine sizing.[3] They are derived from fundamental considerations such as steady-state mass flow balance between two stations, work balance for the components of a common spool, and mechanical compatibility in terms of turbine blade and shaft stress. The governing equations, together with a few gasdynamic and geometric equations, are then used to size every constituent component.

Governing Equations

$$W^* = \Delta H_{\text{compressor}}/U_M^2 Z \tag{1}$$

Equation (1) is equally applicable to low-pressure (LP) and high-pressure (HP) compressors and determines the number of stages Z needed to achieve the required enthalpy change ΔH. U_M is defined as the mean of the entry and exit stations.

Similarly

$$Y^* = \frac{2.0\Delta H_{\text{turbine}}}{U_{M,\text{exit}}^2 Z} \tag{2}$$

Equation (2) is equally applicable to the LP and HP turbines

$$\frac{D_{M,x_1}}{D_{M,x_2}}\sqrt{\frac{Z_{\text{LPT}}}{Z_{\text{LPC}}(\text{BPR}+1)}} = K_{\text{LP}} = \sqrt{\frac{W_{x2}}{\eta_{m,\text{LP}}\eta_{\text{LPT}}W_{x1}}}\sqrt{\frac{W^*_{\text{LPC}}}{Y^*_{\text{LPT}}}} \tag{3}$$

where BPR is the bypass ratio, x_1 is the LPT exit station, and x_2 is the LPC entry station. Use of mass flow and work compatibility equations is made between x_1 and x_2. Upon simplification, Eq. (3) is obtained

$$\frac{D_{M,x_1}}{D_{M,x_2}}\sqrt{\frac{Z_{\text{HPT}}}{Z_{\text{HPC}}}} = K_{\text{HP}} = \sqrt{\frac{W_{x_2}}{\eta_{m,\text{HP}}\eta_{\text{HPT}}W_{x_1}}}\sqrt{\frac{W^*_{\text{HPC}}}{Y^*_{\text{HPT}}}} \tag{4}$$

where x_1 is the HPT exit and x_2 is the HPC entry. The derivation of Eq. (4) is similar to that of Eq. (3)

$$K_\sigma = \sigma_\tau^T/\sigma_{\text{BP}} \tag{5}$$

K_σ is the factor of safety, σ_τ^T is the allowable stress for τ hours of blade life at T°C, and σ_{BP} is the blade pull stress:

$$\sigma_{\text{BP}} = \rho_B U_T^2[(1-\text{HTR}^2)/2]\Phi \tag{6}$$

The σ_{BP} is computed only in rotating components. Φ is blade form factor and ρ is the material density:

$$\varepsilon = \pi A(N/60)^2 = \sigma_{\text{BP}}/2\rho_B\Phi \tag{7}$$

ε is the stress parameter, computed at the exit of the HP and LP turbines, and is a measure of σ_{BP}

$$\Gamma = \frac{16TD_0}{\pi(D_0^4 - D_i^4)} \Rightarrow D_0 = \left[\frac{16T}{\pi\Gamma X}\right]^{1/3} \tag{8}$$

where

$$T = \frac{60\times\text{WRK}}{2\pi N}, \qquad X = 1 - \frac{D_i^4}{D_0^4}$$

Equation (8)[4] is used for sizing the shaft connecting the compressor and turbine. WRK is the work output of turbine, connected to shaft being sized.

Gasdynamic Equations

$$\begin{gathered} 1 - \frac{k-1}{k+1}\boldsymbol{\lambda}^2 = \frac{1}{1+[(k-1)/2]M^2} \\ q(\lambda) = \left(\frac{k+1}{2}\right)^{1/(k-1)}\lambda\left(1-\frac{k-1}{k+1}\lambda^2\right)^{1/(k-1)} \end{gathered} \tag{9}$$

If the Mach number M is known at an engine station, Eq. (9) enables estimation of velocity coefficient λ and, hence, flow coefficient $q(\lambda)$.

At any engine station

$$W = \frac{S\times P\times q(\lambda)\times A}{\sqrt{T}} \tag{10}$$

Knowing the engine thermodynamic state and $q(\lambda)$, the GFP area is computed. S is a gasdynamic constant defined as

$$S = \sqrt{(k/R)} \times \sqrt{[2/(k+1)]^{(k+1)/(k-1)}}, \quad \text{Vander Kirchoff function}$$

Geometric Equations

Geometric Eqs. (11–14) define the physical dimensions of the GFP and parameters such as blade chord, number of blades, blade height, and blade speed.

At any engine station

$$A = \frac{\pi}{4}D_T^2[1-(\text{HTR})^2] = \pi D_M H_B \tag{11}$$

$$\frac{D_T - D_H}{2} = H_B, \qquad \frac{D_H}{D_T} = \text{HTR}, \qquad \frac{D_T + D_H}{2} = D_M \tag{12}$$

$$C_B = H_B/\text{AR}, \qquad N_B = \pi D_T(C/S)(\text{AR}/H_B) \tag{13}$$

At the entry or exit of the compressor or turbine

$$U_T = (\pi/60)D_T N \tag{14}$$

Similarly, U_M and U_H are computed by substituting D_M or D_H for D_T.

Sizing of Axial Flow Compressors

The thermodynamic state at the entry and exit is known from engine cycle analysis. The remaining inputs and their lower and upper limits are stated in Table 1.[3,4]

Equal work or equal temperature rise per stage is assumed. The number of stages required is found by iteration until first-

Table 1 Inputs for compressor sizing[a]

	LPC/fan		HPC	
	Lower limit	Upper limit	Lower limit	Upper limit
Π_{max}	1.50	1.80	1.30	1.60
M_{entry}	0.50	0.60	0.40	0.50
M_{exit}	0.45	0.55	0.20	0.30
HTR_{entry}	0.35	0.50	0.60	0.80
AR_{entry}	3.00	5.00	2.00	5.00
AR_{exit}	2.00	3.00	1.00	2.00
$(C/S)_{entry}$	1.00	1.50	1.00	1.50
$(C/S)_{exit}$	1.00	1.50	1.00	1.50
Φ	0.60	0.80	0.60	0.80

[a]GFP law: D_T or D_M or D_H = constant.

Table 3 Inputs for turbine sizing[a]

	HPT		LPT	
	Lower limit	Upper limit	Lower limit	Upper limit
Y^*	3.00	4.50	3.00	4.50
Z	1	2	1	2
M_{entry}	0.30	0.40	0.40	0.50
M_{exit}	0.45	0.50	0.55	0.60
AR_{entry}	1.00	2.00	2.00	3.00
AR_{exit}	1.00	2.00	4.00	6.00
$(C/S)_{entry}$	1.00	1.50	1.00	1.50
$(C/S)_{exit}$	1.00	1.50	1.00	1.50
Φ	0.60	0.80	0.60	0.80

[a]GFP law: D_T or D_M or D_H = constant.

Table 2 Blade tip speed estimation

Π_{max}	$U/\sqrt{\theta}$, m/s
1.0	185.0
1.1	230.0
1.2	275.0
1.3	320.0
1.4	360.0
1.5	400.0
1.6	435.0
1.7	475.0
1.8	505.0
1.9	535.0

stage pressure ratio Π is equal to or less than first-stage maximum pressure ratio Π_{max}. U_T is estimated from data in Table 2.[4] It is only an approximation and can be adjusted by specifying a speed multiplier factor. θ is the ratio of total temperature at compressor entry to standard temperature.

Knowing the Mach number at compressor entry M_{entry}, Eqs. (9) and (10) are used to determine the corresponding GFP area A_{entry}. With HTR and AR being inputs at the entry station, D_T, and thus D_M, D_H, H_B, and C_B are computed. From D_T and U_T, the spool rotational speed is obtained. To perform sizing at the compressor exit station, a GFP law is assumed. Consider a constant-mean-diameter type of GFP law, from which D_M at the exit station is known. M and AR are additional inputs. Thus, exit station area and, hence, D_T, D_H, H_B, HTR, and C_B are computed.

To estimate stage length[4] stator chord length is taken equal to rotor chord length. Seventeen percent of rotor chord length is required for clearance between the rotor and the stator. The same clearance is also used between the stator and the next rotor. The summation over all the stages gives total compressor length. To determine individual stage sizing characteristics, a linear variation in flow-path dimensions, AR, and C/S is assumed from the entry to the exit station. Knowing C/S, the number of blades in a stage is estimated.

Sizing of Axial Flow Turbines

Unlike the compressor the turbine is a combination of stator–rotor, not rotor–stator. Thus, sizing begins from the exit station. The engine thermodynamic state is known from cycle analysis. The remaining inputs are stated in Table 3.[3,4]

Equal work or equal temperature rise per stage is assumed. The rotational speed is transferred from the compressor that the turbine drives. Using the number of stages and the loading parameter, the mean diameter at the exit station is estimated. With M and AR as inputs at the exit station, A and, hence, D_T, D_H, H_B, HTR, and C_B are computed.

Now a GFP law is assumed. Based on the choice of GFP law, either the tip, mean, or hub diameter is known at the entry station. Knowing M and AR, the entry station area is computed and, hence, all the corresponding sizing parameters. The total turbine length, individual stage sizing, and number of blades in a stage are estimated in a manner similar to the compressor.

Concentric Shafts

The shaft is the power connection between the compressor and the turbine. The allowable stress limit ≈350 MPa, and a diameter ratio of the inner shaft ≈0.85 and a radial clearance of 0.50 cm are assumed between the o.d. of inner shaft and i.d. of outer shaft.[4] The length of the LP and HP spool shafts is the sum of the lengths of the HP compressor, main combustor, and HP turbine.

The inner shaft is sized first, beginning with estimation of its torque capability. Substituting the allowable stress limit and diameter ratio, the o.d. of the inner shaft is computed. The i.d. is then obtained from the specified diameter ratio. An iterative procedure is used to size the outer shaft.[4] The i.d. of the outer shaft is known. After computing its torque capability, an initial estimate of diameter ratio is made, which gives one value of o.d. From Eq. (8) another value of o.d. is obtained. When the two values match within ±0.5%, iteration terminates and the correct value of o.d. is obtained.

Main Combustion Chamber

The current trend is to use an annular combustion chamber. Its three principal components are the inlet dump diffuser, the dome, and the liner, in addition to subcomponents such as the fuel injector, ignitor, burner case, and air swirler. Combustor sizing[3] begins with estimation of the primary air mass flow W_p, i.e., the air introduced through the dome and through the first row of liner holes. The remaining air is termed secondary air, and its introduction into the combustor follows that of primary air. The secondary air is classified as intermediate air to complete the combustion process, dilution air to produce an acceptable exit temperature profile, and cooling air to protect the liner and dome from high temperature levels. Thus, the combustor has three distinct zones, primary, secondary, and dilution.

To estimate W_p, a stoichiometric air-to-fuel ratio AFR is assumed in the primary combustion zone. Thus

$$W_p = \mathrm{AFR}_{\mathrm{stoichiometric}} \times QF_{CC} \tag{15}$$

where QF_{CC} is the fuel flow in combustor. The remaining air mass flow $W_3 - W_p$ is the secondary air W_s. W_3 is the total air mass flow at the combustor entry. QF_{CC} and W_3 are known from engine cycle analysis.

The passage is the area between the liner and burner case in which secondary air first enters. The Mach numbers in the primary combustion zone M_{comb} and in the passage M_{pass} are assumed to be 0.015 and 0.075, respectively, which permit computation of the combustion area A_{comb} and passage area A_{pass}. The sum of the combustion and passage areas is the reference area A_{ref}. A_{ref} is the one-dimensional maximum cross-sectional area of the combustor at the combustion zone.

At the combustor entry, D_M is taken to be the same as that at the HP compressor exit. Using this D_M value, combustor height with respect to reference area, i.e., H_{ref}, is computed, from which combustor outer and inner diameters are obtained. Similarly, the outer and inner diameters of the diffuser are also computed, using the D_M at the HP compressor exit as the diffuser mean diameter. To estimate the diffuser area A_{diff}, a 25% aerodynamic diffusion is assumed before the air is allowed into the combustor. Thus

$$\frac{A_{diff} - A_3}{A_{ref} - A_3} = 0.25 \Rightarrow A_{diff} = 0.25 \times A_{ref} + 0.75 \times A_3 \qquad (16)$$

To compute the height of the flame tube H_{ft}, an equal passage area is assumed between the combustor inner case and the flame tube i.d., and between the combustor outer case and the flame tube o.d. Thus

$$D_{ft,o} = \sqrt{D_o^2 - \frac{2 \times A_{pass}}{\pi}} \qquad (17a)$$

$$D_{ft,i} = \sqrt{D_i^2 + \frac{2 \times A_{pass}}{\pi}} \qquad (17b)$$

$$H_{ft} = \frac{D_{ft,o} - D_{ft,i}}{2} \qquad (17c)$$

The combustor length L_{CC} is the summation of diffuser length L_{diff} and the lengths of the primary zone L_p, secondary zone L_s, and dilution zone L_d, each of which is empirically defined in terms of H_{ft} as follows:

$$L_{diff} \approx 1.125 \times H_{ft} \qquad (18a)$$

$$L_p \approx 0.50 \times H_{ft} \qquad (18b)$$

$$L_s \approx 0.70 \times H_{ft} \qquad (18c)$$

$$L_d \approx 1.50 \times H_{ft} \qquad (18d)$$

Reheat Combustion Chamber

As a limiting case, the o.d. of the afterburner is taken to be the same as D_T at the LP compressor (or fan) entry, i.e., the engine frontal diameter. The i.d. is equal to 0.0 because the afterburner does not have an inner wall. The length-to-diameter ratio L/D is computed from the following correlation[5]:

$$L/D = 2 - (\mathrm{BPR}/2)_{\text{sea-level static}} \qquad (19)$$

Mixer, Ducts, and Nozzle

For the mixer and interconnecting ducts, the inner and outer diameters are the same as those of the upstream component. The o.d. of the bypass duct is taken to be equal to D_T at the fan exit. Knowing the total temperature, pressure, and mass flow at the entry to the bypass duct and $q(\boldsymbol{\lambda})$ at the fan exit, the bypass duct area is computed. Because the o.d. is known, the i.d. also can be computed.

A fully variable convergent–divergent nozzle is assumed. As a limiting case, its diameter is taken to be equal to engine frontal diameter. L/D (or L/H) is assumed to be 1.50 for the mixer and interconnecting ducts and 1.10 for the nozzle. The length of the bypass duct is the summation of the lengths of the LP and HP compressors, fan-compressor duct, main combustor, and the HP and LP turbines.

Constraints System

Engine sizing is an interaction between a large number of thermodynamic, aerodynamic, and mechanical variables. The combination of the numerical values of these variables should be such that sizing various components individually and placing them together to define the gas turbine system must result in aerothermomechanical compatibility, i.e., an operationally feasible system integration. For this purpose, definitions of a set of design constraints follow[1,3]: 1) mechanical constraints—$0.32 \le$ HTR at fan and HPC exit ≤ 0.93, $0.50 \le$ HTR at LPT and HPT exit ≤ 0.93; 2) constraints on allowable stress at turbine blade root—$\varepsilon_{\text{LP and HP spool}} \le 2.75 \times 10^4$ m^2/s^2; 3) aerodynamic constraints—U_T at fan entry ≤ 500 m/s, H_B at HPC exit ≥ 2.0 cm; and 4) constraints on engine envelope—$D_{T,\text{HPT exit}}/D_{T,\text{LPT exit}} \le 1.0$, $D_{T,\text{LPT exit}}/D_{T,\text{fan entry}} \le 1.0$, main combustor $D_o/D_{T,\text{fan entry}} \le 1.0$.

The correct measure of σ_{BP} at the turbine root is

$$\sigma_{BP} \le \sigma_\tau^T/K_\sigma, \qquad K_\sigma \ge 2.0$$

But in the absence of sufficient data on material properties, $\sigma_{BP} \le 325$ MPa.[4] This, together with Eq. (7), $\rho_B = 8500$ kg/m^3, and $\Phi = 0.70$, then defines the constraining limit on σ_{BP} in terms of ε. The speed is expressed in rev/s to compute ε.

In addition to design constraints, Table 4 shows a set of design conditions derived from the experience of past and existing designs.[3,4] They basically serve as diagnostic aids in determining whether a valid design is made. It is desirable to satisfy design conditions; they need not be enforced strictly.

Table 4 Design conditions

Design conditions	Lower limit	Upper limit
Mass flow by frontal area at fan entry, kg/m^2-s	170	195
W^*_{fan}	0.35	0.55
W^*_{HPC}	0.30	0.45
K_{LP}	0.45	0.60
K_{HP}	0.40	0.60
$\boldsymbol{\lambda}$ at fan entry	0.60	0.70
$\boldsymbol{\lambda}$ at HP compressor exit	0.25	0.35
λ at HP turbine exit	0.40	0.50
λ at LP turbine exit	0.60	0.70
$D_{M,\text{LPT}}/D_{M,\text{fan}}$	0.80	1.20
$(D_M/H)_{\text{HPT exit}}$	6.0	16.0
A_{comb}/A_{ref}	0.65	0.75

Engine Weight Estimation

The method of Pera et al.[4] has been used. The preliminary mechanical design, as defined by engine sizing, is the input. The underlying objective is to compute the metal volume of every component. Multiplying the component metal volume with the appropriate material density gives the component weight. Summation over all of the constituent components results in the preliminary estimate of engine weight. For components such as the mixer, frames, and a few miscellaneous items in the combustor, the correlations[4] that directly provide their weight estimate have been used.

The frames are used to support a bearing or to provide structural support. Four types of frames can be identified[4]: 1) single bearing frame without power off-take P_{OT}, 2) single bearing frame with P_{OT}, 3) turbine frame, and 4) intermediate two-bearing or burner frame.

The mechanical design of frames is not available at the preliminary design stage. Instead, correlations[4] are used. In the present analysis, frames of types 1 and 4 are assumed at the entry and exit of the LP compressor. Also, a type 4 frame at the main burner entry and a type 3 frame at the LP turbine exit are assumed. Refer to Refs. 1 and 4 for remaining descriptions of engine weight estimation.

Having computed the metal volume, an appropriate density value is required to estimate the component weight, based on the material type used in its construction. Table 5 provides

Table 5 Materials and their densities

Component	Material	Density, kg/m^3
Fan/LP compressor disc, blades, and casing	Titanium 318	4420
Intermediate casing	INCO 718	8180
HP compressor casing	INCO 718	8180
HP compressor blades and disk of front stages (1, 2, and/or 3)	Titanium 685	4500
HP compressor blades and disk of remaining stages	INCO 718	8180
Diffuser casing from HP compressor to combustor	INCO 718	8180
Combustor casings	INCO 718	8180
Combustor liner	Nimonic C-263	8630
Blades of HP and LP turbines	Nickel-based superalloy	8500
Disc and casing of HP and LP turbines	INCO 718	8180
Bypass duct casing	Titanium 318	4420
Duct casing from LP turbine to mixer	INCO 718	8180
Jet pipe and exhaust nozzle	Nimonic C-263	8630
Afterburner casing	INCO 718	8180
Afterburner liner	Nimonic C-263	8630
LP spool shaft	Steel	8200
HP spool shaft	Steel	8200
Connecting hardware in components	Steel	8200

brief descriptions of the typical materials and their densities, as used in the construction of various components of a combat aircraft gas turbine engine.[1] It is not specific to any engine and is basically representative of current practice. The database therein is quite likely to change with advancements in materials technology.

Computer Simulation

Computer simulation of engine sizing and weight estimation has been developed in Fortran.[1] The sizing point is taken as sea-level static in international standard atmosphere. A theoretical correlation[4] is used to estimate disk volume and, therefore, disk weight as a function of relative disk loading (σ_{BP}*D_H/2.0), i.e., blade load per unit thickness. To have throttle ratio greater than 1.0, overspeeding of the LP and HP spool shafts beyond their design speeds must be allowed. This overspeeding increases σ_{BP} and, hence, relative disk loading, disk volume, and weight. Thus, to estimate the disk volume of rotating components, relative disk loading[4] is multiplied by throttle ratio to compensate for allowable overspeeding.

The simulation code has two parts:

1) Engine cycle analysis is performed at the sizing point using a generalized simulation model[1] that generates the description of thermodynamic parameters.

2) Output of engine cycle analysis and numerical values of various sizing variables are the inputs for the next step. The materials and density database of Table 5 is built in, with an option to specify the number of front stages in the HP compressor that are made of titanium alloy. The engine sizing data, numerical values of constraint system variables, and length and weight of every component, together with engine total length, weight, and GFP schematic, are the output.

The automatic satisfaction of design constraints and design conditions is not incorporated. Thus, simulation code cannot, on its own, make decisions regarding the aerothermomechanical compatibility of GFP layout. It only provides the description of GFP layout, based on the input description. The man-in-loop approach is used to gain good visibility of engine sizing. Based on the guidelines of the built-in constraint system, the input description of sizing variables is continuously trimmed until aerothermomechanical compatibility is achieved. Because engine cycle thermodynamic data at the sizing point form the starting point, this computer simulation is responsive to variations in engine cycle parameters. It is also responsive to the technology level, which is reflected in the numerical values of various sizing variables, materials used, and their densities.

Table 6 Compressor inputs for validation[a]

Inputs	LPC/fan	HPC
Π_{max}	1.50	1.35
M_{entry}	0.68	0.50
M_{exit}	0.50	0.27
HTR_{entry}	0.40	0.672
AR_{entry}	3.0	2.0
AR_{exit}	2.5	1.5
$(C/S)_{entry}$	1.10	1.2
$(C/S)_{exit}$	1.10	1.2
Φ	0.80	0.80
Speed scalar	1.24	1.31

[a]GFP law: D_M = constant, D_T = constant; HPC front stages with titanium 685 = 3.

Table 7 Turbine inputs for validation[a]

Inputs	HPT	LPT
Y*	3.40	3.30
Z	1	1
M_{entry}	0.30	0.45
M_{exit}	0.45	0.525
AR_{entry}	1.5	2.0
AR_{exit}	1.5	3.0
$(C/S)_{entry}$	1.5	1.5
$(C/S)_{exit}$	1.5	1.5
Φ	0.80	0.90

[a]GFP law: D_M = constant, D_M = constant.

Validation

An attempt is now made to reproduce the GFP, length, and weight of an existing engine.[1] The baseline engine is a twin-spool, mixed-flow turbofan engine with an afterburner, one of the current generation of combat aircraft engines in use. The numerical values of the input sizing parameters are given in Table 6 for compressors and in Table 7 for turbines. The percent deviation of results with respect to the baseline engine is stated in Table 8.

The L/D correlation, as stated in the preceding text, results in overprediction of afterburner length. Thus, only 80% of the estimated length is taken to be afterburner length. This is acceptable because the reference diameter has been taken to be the engine frontal diameter. But the afterburner diameter is

Table 8 Validation results

Sizing variable	Deviation
$D_{T,\mathrm{fan}}$ at entry and exit	0.06/−7.35
$D_{H,\mathrm{fan}}$ at entry and exit	−0.33/−10.7
$\mathrm{HTR}_{\mathrm{fan\,exit}}$	−3.67
$H_{\mathrm{FAN,entry}}$ and $H_{\mathrm{fan,exit}}$	0.32/0.01
$A_{\mathrm{FAN,entry}}$ and $A_{\mathrm{fan,exit}}$	0.28/−8.70
Z_{FAN} and $\mathrm{RPM}_{\mathrm{fan}}$	0.00/−0.19
$U_{T,\mathrm{fan\,entry}}$	−0.12
Mass flow by frontal area at fan entry	0.24
$D_{T,\mathrm{HPC}}$ at entry and exit	−0.24/1.52
$D_{H,\mathrm{HPC}}$ at entry and exit	−0.84/1.44
$\mathrm{HTR}_{\mathrm{HPC,exit}}$	−0.06
$H_{\mathrm{HPC,entry}}$ and $H_{\mathrm{HPC,exit}}$	−0.80/2.39
$A_{\mathrm{HPC,entry}}$ and $A_{\mathrm{HPC,exit}}$	−1.44/5.15
Z_{HPC} and $\mathrm{RPM}_{\mathrm{HPC}}$	0.0/−0.05
W^*_{fan} and W^*_{HPC}	10.37/0.78
$D_{T,\mathrm{HPT,exit}}$ and $D_{H,\mathrm{HPT,exit}}$	0.01/0.08
$\mathrm{HTR}_{\mathrm{HPT,exit}}$ and $\mathrm{HTR}_{\mathrm{LPT,exit}}$	0.08/2.44
$H_{\mathrm{HPT,exit}}$ and $H_{\mathrm{LPT,exit}}$	−0.35/−9.23
K_{HP} and K_{LP}	6.17/1.51
$\varepsilon_{\mathrm{HPT}}$ and $\varepsilon_{\mathrm{LPT}}$	−1.11/−12.52
$A_{\mathrm{HPT,exit}}$ and $A_{\mathrm{LPT,exit}}$	−0.32/−12.1
$D_{T,\mathrm{LPT,exit}}$ and $D_{H,\mathrm{LPT,exit}}$	−4.12/−1.86
Engine length	3.30
Engine weight	4.00

usually less than the engine frontal diameter. The individual breakup of the length and weight of every component is not available. Thus, only overall engine length and weight have been compared.

The order of magnitude of the deviation is quite acceptable for conceptual design phase studies. The possible sources of deviation are as follows:

1) The baseline engine thermodynamic state is not exactly reproduced. This is because a generalized engine model has been used that was developed to parametrically evaluate the performance of a large number of engine design options, consistent with conceptual design requirements. It is not specifically tailored to simulate the performance of the baseline or of any particular engine cycle.

2) The constant-diameter type of GFP law is not adequate to reproduce the sizing parameters at the fan exit station.

Application of Simulation Code

The engine cycle that follows is the optimum over a high-altitude air combat mission.[1] OPR is overall pressure ratio, $\mathrm{TET}_{\max}$ is the maximum turbine entry temperature, TR is the throttle ratio, T_{AB} is the afterburner exit temperature, FPR is the fan pressure ratio, and $W_{\mathrm{ENG,DP}}$ is the design point mass flow: BPR = 0.61, OPR = 26.00, $\mathrm{TET}_{\max}$ = 1900 K, TR = 1.1380, T_{AB} = 1935 K, FPR = 3.448, and $W_{\mathrm{ENG,DP}}$ = 72.5 kg/s.

The aim is only to demonstrate the typical application of computer simulation. Attempts have not been made to identify the GFP that meets the criteria of the final design choice. But this can easily be done by repeated use of computer simulation for various parametric combinations of input data sets, then identifying the solution that best meets the final design choice criteria. Alternately, computer simulation may be directly integrated into a constrained optimization problem, with engine weight minimization as the functional objective.

The two sets of sizing variables were used, as stated in Table 9 for compressors and in Table 10 for turbines.[1]

Set 1 corresponds to the existing technology level and is derived from the experience of the baseline engine that is used for validation. The loading parameters of HP and LP turbines, HTR at HP compressor entry, and speed scalars of the LP and HP spools were readjusted to satisfy the aerothermomechanical compatibility. Set 2 incorporates an advanced level of technology, achievable in the near future. The input description of Tables 9 and 10 is consistent with design trends.[3,4] Set 2 results in a weight and length reduction of 9.5 and 5.7%, respectively.

The schematic of the resulting engine GFP is shown in Fig. 1 for set 1 and in Fig. 2 for set 2. There is such a sharp curvature in the duct connecting the fan and the HP compressor that it may induce high pressure loss in the flow. The tip diameter of the HP compressor needs to be increased to reduce the curvature. In Fig. 2, the following parameters were altered in the input data of set 2 for the HP compressor only; their modified values and resulting input data set are referred to as set 3: M_{entry} = 0.45, M_{exit} = 0.19, speed scalar = 1.125, and $\mathrm{HTR}_{\mathrm{entry}}$ = 0.70.

A slight diffusion is permitted in the fan-compressor duct, because of which the Mach number from the fan exit to the HP compressor entry changes from 0.49 to 0.45. A lower Mach

Table 9 Description of inputs for compressors[a]

	LPC/fan		HPC	
Inputs	Set 1	Set 2	Set 1	Set 2
$\Pi_{\max}$	1.50	1.60	1.35	1.50
M_{entry}	0.70	0.70	0.49	0.49
M_{exit}	0.49	0.49	0.23	0.23
$\mathrm{HTR}_{\mathrm{entry}}$	0.40	0.40	0.60	0.60
$\mathrm{AR}_{\mathrm{entry}}$	3.0	3.0	2.0	2.0
$\mathrm{AR}_{\mathrm{exit}}$	2.50	2.50	1.5	1.5
$(C/S)_{\mathrm{entry}}$	1.10	1.10	1.2	1.2
$(C/S)_{\mathrm{exit}}$	1.10	1.10	1.2	1.2
Φ	0.80	0.80	0.80	0.80
Speed scalar	1.10	1.025	1.125	1.075

[a]GFP law: D_M = constant, D_T = constant; number of HPC front stages with titanium 685 = 3.

Table 10 Description of inputs for turbines[a]

	HPT		LPT	
Inputs	Set 1	Set 2	Set 1	Set 2
Y^*	4.0	4.3	3.9	4.2
Z	1	1	1	1
M_{entry}	0.30	0.30	0.425	0.425
M_{exit}	0.425	0.425	0.550	0.550
$\mathrm{AR}_{\mathrm{entry}}$	1.5	1.5	1.5	1.5
$\mathrm{AR}_{\mathrm{exit}}$	1.5	1.5	1.5	1.5
$(C/S)_{\mathrm{entry}}$	1.5	1.5	1.5	1.5
$(C/S)_{\mathrm{exit}}$	1.5	1.5	1.5	1.5
Φ	0.80	0.80	0.90	0.90

[a]GFP law: D_M = constant, D_M = constant.

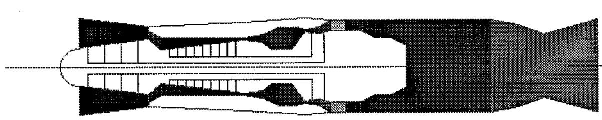

Fig. 1 Engine GFP for set 1. Frontal diameter, 0.7066 m; engine length, 3.7961 m; and engine weight, 782.7 kg.

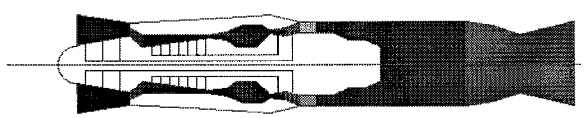

Fig. 2 Engine GFP for set 2. Frontal diameter, 0.7066 m; engine length, 3.5831 m; and engine weight, 706.1 kg.

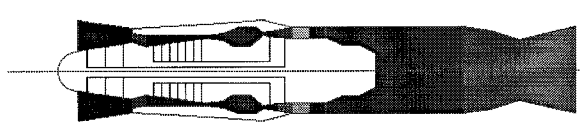

Fig. 3 Revised engine GFP. Frontal diameter, 0.7066 m; engine length, 3.4686 m; and engine weight, 734.6 kg.

Table 11 Engine sizing output for various input sets

Output sizing variable	Set 1	Set 2	Set 3
Engine length, m	3.7961	3.5831	3.4686
Engine weight, kg	783.0	707.0	735.0
Frontal diameter, m	0.707	0.707	0.707
Z_{LPC}/Z_{HPC}	4/9	3/7	3/7
LP spool speed, rpm	11007	12029	12029
HP spool speed, rpm	17145	18472	16717
Fan/HPC loading parameter	0.435/0.385	0.486/0.426	0.486/0.366
K_{LP}/K_{HP}	0.476/0.51	0.485/0.52	0.485/0.47
$\varepsilon_{LP}/\varepsilon_{HP}$, m^2/s^2	1.44/1.96	1.72/2.27	1.72/1.86
HTR at HP compressor exit	0.9082	0.9082	0.9182
H_B at HP compressor exit, cm	1.95	1.95	2.01

number at the HP compressor entry increases the flow area. This, together with higher HTR, increases D_T at the HP compressor entry. The net effect is also to reduce the H_B at the HP compressor exit. Thus, the Mach number at the exit station is lowered. The resulting engine GFP is shown in Fig. 3. The typical sizing output for all three input sets is shown in Table 11.

The description in the preceding text adequately reveals the capabilities of computer simulation of engine sizing and weight estimation. The aircraft empty weight during conceptual design is estimated using statistical correlations[6] and includes the airframe as well as engine weight. Knowing the engine weight, breakup of aircraft empty weight into engine and airframe weight is obtained, which acts as the preliminary design goal from the weight point of view. At a later stage, when more refined estimates are made available, the impact of variations in engine and/or airframe weight on weapon system response can be investigated using mission analysis techniques.[1]

Conclusions

Preliminary engine sizing and weight estimation methods are proprietary to the industry that performs conceptual design studies and are not available for general use. The present work bridges this gap. It provides a comprehensive description of mathematical formulation for computer simulation of preliminary engine sizing and weight estimation, supplemented with the required database. The resulting computer simulation has been validated with respect to a present-generation military engine, and results are presented to demonstrate its application that show it is a highly useful conceptual design aid. It provides good visibility into complex modeling equations based on the technology level used and the engine cycle definition. The scope of the present work is limited to twin-spool, mixed-flow turbofan and twin-spool turbojet types of propulsion concepts. But it is easily adaptable to continuing advancements in engine design and materials technology, as well as to alternate propulsion concepts, by suitable modifications in the built-in statistical design database and empirical correlations.

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