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Technical Notes

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Sensitivity of Design Point Choice on Engine Cycle Selection

Vivek Sanghi*
Gas Turbine Research Establishment, Bangalore 560 093, India

Nomenclature

C_{D0} = zero-lift drag coefficient
C_L = lift coefficient
g = design constraints
H = altitude
K = induced-drag coefficient
M = Mach number
N = mechanical speed
n = load factor in sustained turn
S_{wing} = aircraft wing area
T = total temperature
t = time
W_{EMP} = airframe empty weight
W_{ENG} = mass flow rate/per engine
W_F = mission fuel consumed
W_{TO} = aircraft takeoff gross weight

Subscripts

ACC = acceleration
CL = climb
DP = design point
max = maximum value
SLS = sea-level static

Introduction

THE optimum engine cycle is identified with respect to a baseline reference called the engine design point. The optimum cycle is the one for which an aircraft weapon system is most responsive, in terms of the performance as well as the cost, to the requirements of a specified baseline-design mission application. The description of solution methodology (i.e., optimization with surface fit approximations), its computer simulation and validation, a typical design database, and a few case studies illustrating conceptual design cycle optimization and overall sizing of the resulting aircraft for typical combat missions are contained in Refs. 1 and 2. The engine steady-state performance over the entire flight envelope of the prescribed mission is an important input to the aforesaid conceptual design cycle optimization software. It is computed by thermodynamic relationships, after determining the engine equilibrium.

The engine steady-state performance estimation starts with design point analysis. The design point is the condition for which an engine is designed, i.e., a certain H, M, a set of engine design variables, and the area settings of variable geometry features. The data generated during the design point analysis are an input for off-design analysis, which then determines the performance at conditions other than those for which the engine was designed. The off-design performance follows from the conditions that are fixed by the choice of engine design point, flight conditions, power setting, and the area settings of variable geometry features like the exhaust nozzle and the variable stators in compressors.

The assumption of engine design point should be such that the chosen engine cycle provides the best performance over the entire flight envelope of specified mission and operates satisfactorily. Conventionally, the sea-level static condition in international standard atmosphere (ISA) with temperature deviation from standard day (DT_{amb}) equal to 0.0 has invariably been the engine design point. The reason being that it is the sea-level static at which the engine is manufactured and tested, and a common design point permits comparison of various engine designs.

The present and futuristic military missions require the aircraft to fly for varying duration over a wide range of altitude and Mach number points. Thus, it is of interest to identify if the choice of a flight point as the engine design point can lead to a better optimized cycle.

Mattingly[3] presented a typical case study in which the optimum cycle selection is shown with $H = 10.5$ km/$M = 1.60$ in ISA at $DT_{\mathrm{amb}} = 0.0$ as the engine design point. However, it does not discuss its relative merits with respect to cycle optimization studies at the conventional design point or at various other flight design points. This Note explores the sensitivity of engine design point choice on optimum engine cycle selection and attempts to identify its appropriate choice.

Problem Description

The optimization with surface fit approximations has been performed over a high-altitude air-combat mission.[3] It includes short takeoff and landing; loiter; mix of subsonic, transonic, and supersonic legs; high maneuverability; persistence; and supersonic dry cruise.

As per the current military trends, a twin-spool, mixed-flow afterburning turbofan was used as the propulsion concept. The minimization of W_{TO} was chosen as the figure of merit to optimize the engine/aircraft system because a smaller weapon system and its subsystems cost less to build and operate. The SI system of units has been used, except for weight which is in kilograms.

The formulation of optimization problem is as follows, where TLDG is thrust loading, and BCA and BCM are best cruise H and best cruise M, respectively. Minimize W_{TO}, subject to 1) box constraints, i.e., design space and 2) inequality constraints ($g_1 \cdots g_8$): (g_1) TLDG$_{\mathrm{SLS}}$ at $DT_{\mathrm{amb}} = 15K \leq 1.20$; ($g_2$) $W_{\mathrm{ENG,SLS}}$ at $DT_{\mathrm{amb}} = 15K \leq 90$ kg/s; (g_3) takeoff ground run (S_{TO}) ≤ 450 m; (g_4) n at $H = 9.0$ km, $M = 1.60 \geq 5.0$; (g_5) n at $H = 9.0$ km, $M = 0.90 \geq 5.0$; (g_6) t_{ACC}, $H = 9.0$ km, $M = 0.8$ to $1.5 \leq 50$ s; (g_7) landing ground run (S_{LND}) ≤ 450 m; and (g_8) t_{CL} to BCA/BCM from sea level ≤ 150 s.

A total of five design set variables are chosen for parametric variation to identify the optimum engine cycles. The description of design

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*Scientist, Engine Simulation Division. Member AIAA.

set variables and their design space is given next, where BPR is engine bypass ratio, OPR is engine overall pressure ratio, TR is throttle ratio, T_{AB} is the maximum afterburner exit temperature, and WLDG is aircraft wing loading. TR is defined as $TR = TET_{max}/TET_{DP}$, where TET is turbine entry temperature. In accordance with a near-term achievable technology level, TET_{max} was kept fixed at $1900K$: $0.20 \leq BPR \leq 1.0$, $20.0 \leq OPR \leq 30.0$, $1.00 \leq TR \leq 1.40$, $1800K \leq T_{AB} \leq 2000K$, and $325\ kg/m^2 \leq WLDG \leq 400\ kg/m^2$. The remaining engine cycle variables that do not appear in the design space like components' efficiency, pressure loss coefficients, and cooling bleeds take a fixed value, as per the state of the art.

Because most of the mission is flown at an altitude of 9.0 km, $H = 9.0$ km in ISA at $DT_{amb} = 0.0$ was chosen as the design altitude to perform cycle optimization with a flight point as the engine design point. At this altitude, a range of design Mach numbers was chosen in the vicinity of supercruise Mach number (which is 1.50 for an air-combat mission), i.e., 1.4, 1.5, 1.55, and 1.6, to investigate their sensitivity on optimum cycle selection. These H/M combinations are either critical or near-critical flight segments, and the engine must perform satisfactorily in this flight regime. The cycle optimization was also performed at $H = 9$ km$/M = 0.90$ because it is the prolonged subsonic cruise condition, at $H = 9.0$ km$/M = 0.60$ as a typical loiter condition, and at $H = 6$ km$/M = 1.30$, which also is a flight point in the envelope of air-combat mission.

To define the airframe, the variation of C_L, C_{D0}, and K with Mach number that is typical of a modern combat aircraft was assumed.[1] It was further assumed that this variation is not specific to any aircraft design and is applicable to all the engine-airframe combinations within the prescribed design space. The W_{EMP} is estimated as a function of W_{TO} from the statistical correlations,[3] and the fuel consumed over the specified mission defines the internal fuel capacity. It eliminates the airframe design variables like aspect ratio, wing sweep, thickness ratio, and taper ratio from the design set and permits a simplified representation of the airframe. It is acceptable because the problem is addressed to the initial conceptual sizing.

The engine-component maps are not available during the conceptual design studies, and alternative performance estimation methods that work without component maps have been used.[1] In absence of rotational speeds, corrected mass flow (W_{cor}) at the fan (or low pressure compressor) inlet is used as a control variable to estimate off-design dry mode performance at max power setting. At a given H, with increase in M, W_{cor} at the fan inlet ($W_{1,cor}$) is maintained constant at its design point value by allowing TET to increase. If TET exceeds TET_{max}, $W_{1,cor}$ is reduced until TET equals TET_{max}. To move from max to part power setting in dry operation mode, $W_{1,cor}$ is reduced.

Results and Discussion

Table 1 contains the engine cycle optimization results and the associated system sizing and mission response data at various design points. For every case, the numerical values of all of the engine design variables are stated with respect to the chosen design point, except for the TLDG and W_{ENG} that are referenced back to the sea-level statistic, $DT_{amb} = 15$ K. All of the optimum solutions correspond to a twin-engine aircraft.

At the design altitude of 9.0 km, the results indicate a better system optimization when a subsonic or a high-subsonic Mach number is used as the design Mach number. But the benefits are illusory because it may not be possible to translate the resulting optimum cycle into an aerothermomechanically compatible design because of the high value of TR and a low value of design total temperature at engine entry ($T_{1,DP}$). At low $T_{1,DP}$, a high TR results to provide sufficient allowance for TET to increase to maintain $W_{1,cor}$ constant at its design value; otherwise it will penalize the engine thrust and specific fuel consumption at those flight points where T_1 is greater than $T_{1,DP}$.

At $H = 9.0$ km, $M = 0.60$, and $DT_{amb} = 0.0$ K, the $T_{1,DP}$ is 246 K, whereas it is 267 K at $H = 9.0$ km and $M = 0.90$. The fan or low spool mechanical speed at an off-design flight point is computed by Eq. (1):

$$N = N_{DP} * \sqrt{T_1/T_{1,DP}} * X \tag{1}$$

where $X (\leq 1.0)$ is the reduction in corrected speed if TET exceeds its maximum value. Because engine steady-state performance estimation method as used in the present work does not utilize component maps, any reduction in fan-corrected speed is reflected in the form of reduction in fan-corrected mass flow. The X is 1.0 or close to 1.0 at a large number of flight points if TR is sufficiently high because it delays the occurrence of TET_{max} to higher T_1.

Consider a few typical flight points where the H (km)$/M$ combinations are 0.0/0.2, 0.0/0.8, 9.0/1.2, 9.0/1.5, 9.0/1.6, and 9.0/2.0. Because of high TR, the engine is able to operate at the design or near-design value of $W_{1,cor}$ (or fan-corrected speed). Together with low $T_{1,DP}$, it results in a large overspeeding of fan mechanical speed (as high as 10–12% of design value) continuously for a considerable amount of mission time. It makes mechanical design difficult, increases engine weight, and reduces the life of rotating components.

Table 1 Optimum solution sets

Variables	A	B	C	D	E	F	G	H
H (km)$/M$	0.0/0.0	9.0/0.6	9.0/0.9	9.0/1.4	9.0/1.5	9.0/1.55	9.0/1.6	6.0/1.3
$T_{1,DP}$, K	288.0	246.0	267.0	320.0	333.0	340.0	348.0	333.0
BPR	0.6166	0.6656	0.6218	0.6347	0.6518	0.6384	0.6357	0.6027
OPR	27.81	24.60	23.60	25.00	25.18	27.30	27.05	26.03
TET_{max}, K	1900	1900	1900	1900	1900	1900	1900	1900
TET, K	1602	1384	1520	1778	1842	1882	1900	1848
TR	1.1862	1.3736	1.2500	1.0690	1.0316	1.0097	1.0000	1.0282
T_{AB}, K	1830	1800	1800	1800	1800	1800	1948	1800
$TLDG_{SLS}$	1.1974	1.2024	1.2015	1.1914	1.1874	1.2029	1.2091	1.1926
WLDG, kg/m^2	335.9	329.8	325.7	329.7	332.0	331.7	343.7	331.3
$W_{ENG,SLS}$, kg/s	90.0	88.0	86.5	90.0	90.5	90.0	90.5	90.5
W_{TO}, kg	13060	12626	12755	13049	13131	13166	13445	13275
W_{EMP}, kg	7234	7025	7086	7229	7268	7285	7419	7337
$W_{F,msn}$, kg	4420	4196	4263	4415	4457	4475	4621	4532
S_{wing}, m^2	38.88	38.29	39.16	39.58	39.55	39.69	39.12	40.07
n at 9 km/1.6	7.16	7.20	6.91	7.10	7.12	7.17	7.53	6.99
n at 9 km/0.9	4.98	5.00	5.01	4.99	4.97	5.02	5.02	4.97
t_{ACC}, s	38.71	39.30	40.52	39.80	39.61	38.82	36.15	40.20
t_{CL}, s	156.6	158.8	153.3	157.3	158.1	156.3	159.8	155.5
S_{TO}, m	388.6	382.5	379.0	384.2	386.9	384.1	393.2	387.1
S_{LND}, m	405.8	401.0	398.0	401.3	403.1	402.9	411.9	402.8

The typical permissible overspeed limit is about 5–7% of design mechanical speed. Thus fan-corrected speed needs to be reduced if its mechanical speed exceeds the limiting value. It will alter the engine steady-state performance, and the optimum system definition as obtained earlier will also change.

To investigate this issue, another optimization study was performed in which the fan mechanical speed was limited to 1.07 times its design speed. It results in W_{TO} and $W_{ENG,SLS}$ that are of the same order of magnitude as at the conventional design point.

As design Mach number increases at $H = 9.0$ km., optimum TR begins to decrease. This is because the increase in design Mach number causes the $T_{1,DP}$ and, hence, TET_{DP} also to increase, which reduces TR. There is a flight point, which is $M = 1.55$ in the present case, at which TR equals 1.0. If a higher Mach number, e.g., 1.60, is chosen as the design point, the optimum value of W_{TO} begins to increase because the least value that TR can take is 1.0. Thus as design Mach number increases, TET_{max} occurs at a higher Mach number (or $T_{1,DP}$), and the engine operates at relatively reduced TET at a large number of flight points, where Mach number (or T_1) is lower than that of the design point.

Though not investigated, the trends as observed at design altitude of 9.0 km should also hold true at other design altitudes, because any design combination of H/M can be translated into an equivalent $T_{1,DP}$, which then dictates the quality of chosen flight point as the engine design point. As a typical example, cycle optimization results at $H = 6.0$ km/$M = 1.3$ in ISA at $DT_{amb} = 0$ K ($T_{1,DP} = 333$ K) are not significantly different in comparison to that at $H = 9.0$ km/$M = 1.5$ in ISA at $DT_{amb} = 0$ K, which also corresponds to $T_{1,DP} = 333$ K.

To summarize, at a prescribed design altitude there is a lower limit on $T_{1,DP}$, below which (despite a lower value of optimum W_{TO}) the resulting cycle is not practically feasible. There also exists an upper limit on $T_{1,DP}$ beyond which cycle optimization results in an increased optimum W_{TO}. Between these limits of $T_{1,DP}$, cycle optimization, results do not differ significantly within themselves and in comparison with that at the conventional design point. Thus instead of searching for an optimum design point, it is sufficient to perform conceptual design system optimization at the conventional engine design point.

Conclusions

The value that $T_{1,DP}$ takes at an H/M combination dictates its suitability as the engine design point. If minimization of W_{TO} is the criteria for engine cycle optimization, then a flight condition with a low $T_{1,DP}$ is more suited as the design point because it results in a lower value of optimum W_{TO}. However, it also requires the engine to overspeed continuously for long durations, thereby causing an increased engine weight and reduction in the life of the rotating components. If this overspeeding is restricted to the current design limits of about 7% of design speed, savings in W_{TO} diminish.

As $T_{1,DP}$ increases, there arises a flight point at which TR equals 1.0. Because TR cannot take a value lower than 1.0, choosing further higher values of $T_{1,DP}$ only delays the occurrence of TET_{max}. There will be a large number of flight points at which T_1 is lower than $T_{1,DP}$. At all such points, the engine will operate at relatively lower values of TET, causing a penalty in the optimum W_{TO}.

Between these lower and upper ranges of $T_{1,DP}$, there is no significant difference in cycle optimization results, and it is practically independent of design point choice. Thus the use of conventional engine design point itself is adequate for conceptual design cycle optimization.

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Evaluation of Fuel Distribution in a Gas-Turbine Premixer: Influence of Swirl

A. R. Eaton,* S. F. Frey,* D. M. Cusano,*
M. W. Plesniak,† and P. E. Sojka‡
Purdue University, West Lafayette, Indiana 47907-1003

I. Introduction

MORE stringent regulations are requiring gas turbine manufacturers to reduce NO_x emissions. One method is to control the flame zone temperature through lean combustion. Thorough mixing of air and fuel is essential for minimal emissions. Hence, studies must be performed to understand how design parameters influence mixing characteristics.

Several methods are available to increase fuel–air mixing prior to combustion. The introduction of swirl is a common one because the fuel–air interaction time in the premixer is increased, and mixing enhanced, for a swirling flow. The enhancement in mixing is due in part to an increase in turbulence production and transport that results from straining of the flow.

Swirl also benefits the very lean operating conditions of these premixers by creating a recirculation zone in the combustion chamber that enhances flame stability. This flow reversal occurs for swirl number values greater than 0.5 (Ref. 1) and improves the combustion process by entraining and recirculating a portion of the hot combustion products, which in turn mix with and ignite the incoming fuel–air mixture. (In all cases, the swirl number is defined as $S = U/W = \tan\theta$, where U and W are the axial and tangential velocity components and θ is the premixer swirl vane angle.) This helps to not only stabilize the flame, but also to improve combustion efficiency, provide better flame blowoff limits, and reduce formation of gaseous and particulate pollutants.[2]

Several studies on normal jets issuing into swirling crossflows (a configuration representative of radial spray bars in an axisymmetric swirling flow) have been performed to better understand premixer flow characteristics. Ferrell et al.[3] and Ong et al.[4] reported results from extensive flow visualization studies on normal jets in swirling crossflows. Swirl numbers of 0, 1, and 2.75 (corresponding to swirl angles θ of 0, 45, and 70 deg) were examined under jet-to-crossflow momentum ratios of 2, 4, and 6. They showed that jet penetration decreased with an increase in swirl number because the bending of the jet was more pronounced at higher swirl numbers.

The experiments of Ahmed and So[5] confirmed that swirl has a dramatic effect on jet penetration. They used a one-component laser Doppler velocimeter to examine a jet in swirling crossflow (swirl number 2.25) with momentum ratios of 0.46 and 0.96. Jet penetration was found to increase with increased momentum ratio. The mean and turbulent velocity disturbances were also found to dissipate quickly. They concluded that a substantial fraction of the jet mean kinetic energy is converted into turbulent energy very rapidly, thus creating a high level of turbulence in the vicinity of the jet. This was found to improve mixing.

Chao and Ho[6,7] used a computational model to numerically study the mixing of a normal jet in a swirling crossflow. Comparisons of

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*Graduate Research Assistant, Thermal Sciences and Propulsion Center, School of Mechanical Engineering.

†Associate Professor, Thermal Sciences and Propulsion Center, School of Mechanical Engineering; plesniak@ecn.purdue.edu. Senior Member AIAA.

‡Associate Professor, Thermal Sciences and Propulsion Center, School of Mechanical Engineering.