Project: Jet Engine Cycle Design

In this assignment, you will explore parametric engine cycle analysis (e.g., conceptual design) for airbreathing aircraft engines. This is a **group/team** project (*members will be assigned by the instructor*). Your team will:

1) develop the equations necessary to model a general turbine engine cycle **and** implement your equations in a computer model to simulate engine cycles using any programming language or modeling tool (e.g., Matlab, Python, C^++, Excel, …) that you wish *as long as the group develops its own engine simulation software*;

2) perform engine cycle analysis and provide preliminary designs for a number of operating conditions **and** then use your results to make suggestions on a "best" engine design for some specific aircraft; and

3) prepare a written report.

You may perform your calculations and report your results in *either SI or English units*, but be consistent - stick to one system.

1. Engine Cycle Equation Development and Simulation

In this project, your team will be using an engine cycle design approach to choose individual engine configurations for different aircraft flight conditions. Here, a configuration means a grouping of engine components, including component design operating parameters. You can consider using any of the standard engine **components** examined in class that we used to analyze jet engines, specifically: inlets/diffusers, nozzles, forward-mounted fans, compressors, turbines, main burners, and afterburners. In addition, you should consider the following additions/modifications that can improve engine performance: **(1) bleeding compressor air** to cool your turbine blades, thereby increasing the maximum allowable turbine inlet temperature; **(2)** using a **combined/mixed nozzle** that allows the core and bypass flows to mix rather than employing separate core and fan/bypass nozzles; and **(3)** adding a **bypass split** that can allow just a portion of the bypass air (when using a fan) to go through the fan/bypass nozzle with the rest mixing with the core flow and going through a combined nozzle. In addition, you will need to include in your engine cycle model the **pump** that supplies fuel to the combustor(s).

The first thing you need to do is **formulate equations** that describe the thermodynamic variable changes across each engine component, as well as overall engine performance parameters such as specific thrust.

Engine Components and State Numbering

You must develop a single set of equations that would be **generally applicable** to any engine that might include **any** of the following components:

- diffuser/inlet (**d**)
- fan (**f**)
- compressor (**c**)
- fuel pump (**p**)
- main combustor/burner (**b**)
- turbine (**t**): runs compressor and fuel pump in our simulation
- fan turbine (**ft**): runs the fan in our simulation
- afterburner (**ab**)
- core nozzle (**n**)

- fan/bypass nozzle (**fn**)
- combined nozzle (**cn**)

To model the use of bleed air from the compressor mixing with the turbine flow, and to model the core and fan flows mixing in the combined nozzle, you should add some "virtual" components to the engine, similar to the approach you took in a homework problem:

- turbine mixer (**tm**)
- nozzle mixer (**nm**)

(these components are virtual in the sense that they would not be separate components of a real engine, but rather are simplified models of what happens in the turbine or nozzle).

For your analysis, you **must** use the following state numbering scheme.

–	ambient	a	–	afterburner exhaust	6
–	diffuser exhaust	1	–	core nozzle exhaust	e
–	fan exhaust	2	–	fan/bypass nozzle exhaust	ef
–	compressor exhaust	3	–	nozzle mixer exhaust	7
–	main combustor/burner exhaust	4	–	combined nozzle exhaust	ec
–	turbine exhaust	5.1	–	fuel storage	f.1
–	turbine mixer exhaust	5.1m	–	fuel pump exhaust	f.2
–	fan turbine exhaust	5.2			

Inputs

The equations you develop should depend on the **following flight conditions**:
- ambient temperature (T_a), ambient pressure (p_a); fuel storage pressure (p_f)
- flight Mach number (**M**);

and **design parameters**:
- compressor (stagnation) pressure ratio (**Pr$_c$**);
- fan (stagnation) pressure ratio (**Pr$_f$**);
- fuel-air ratios for the main burner (*f*) and afterburner (*f$_{ab}$*) with $f = \dot{m}_f / \dot{m}_a$ and $f_{ab} = \dot{m}_{f\,ab} / \dot{m}_a$, where $\dot{m}_a$ is the "core" air mass flowrate <u>entering</u> the compressor;
- bypass ratio (β), the mass ratio of secondary air (air bypassing the compressor) flowrate to the air flowrate entering the compressor ($\beta = \dot{m}_s / \dot{m}_a$);
- bleed ratio (**b**), the mass ratio of the flowrate of air bled from compressor to cool the turbine to the flow rate of air entering the compressor ($b = \dot{m}_b / \dot{m}_a$);
- split ratio (σ), the fraction of bypass air that goes through the fan/bypass nozzle rather than the combined nozzle – so σ=1 means you are using a fan/bypass nozzle and core nozzle, σ=0 means you are using only a combined nozzle, and intermediate values mean you are using both a combined nozzle and fan/bypass nozzle;

and the following **component/flow properties**:
- average normalized specific heats (**c$_{p,i}$/R**), and gas molecular weights ($\overline{W}_i$) for each component;

- *adiabatic* efficiencies (η_i) for the nozzles, diffuser, and fuel pump; and ram pressure recovery (r_d) for the diffuser (see below for definitions);
- *polytropic* efficiencies ($\eta_{p,i}$) for the fan, compressor and turbines (see explanation below);
- combustion efficiencies (η_b, η_{ab}) and stagnation pressure ratios (Pr_b, Pr_{ab}) for the main burner and afterburner;
- fuel heating value (Δh_R) and fuel density (ρ_f) - the same fuel is used for all combustors in a jet engine;
- the specific drag loss (Δd) associated with the bypass fan (see below for explanation);
- the pressure loss function for the virtual nozzle mixer (Pr_{nm}, see below).

In addition, you will be given the following **constraints**/**limits**:

- temperature limits for the main burner (T_{max}) and afterburner ($T_{max,ab}$)
- maximum allowed bleed fraction (b_{max}), and maximum bypass ratio (β_{max})
- minimum and maximum fan pressure ratios, and maximum overall compression ratio

Outputs

You should develop equations for the following "outlet" values, in terms of the above parameters or some of the other outputs listed below:

- air speed (**u**) (from the flight Mach number)
- exit stagnation temperatures (T_{oi}) and pressures (p_{oi}) for each component <u>except the nozzles and fuel pump</u>
  - for the fan, compressor and turbines, write your outputs (where appropriate) in terms of the polytropic efficiencies, see below (i.e., *do not report equations for these components that use adiabatic efficiencies*)
- exhaust temperature (T_i), velocities (u_i) and Mach numbers (M_i) from each of the nozzles: core, fan, and combined (even if not used)
- exit stagnation pressure for the fuel pump ($p_{f.2}$)
- power per unit mass **core** air flow rate ($w_i = \dot{W}_i/\dot{m}_a$) for the fan, compressor, main turbine, fan turbine, and fuel pump
- effective specific thrust ($\tau/\dot{m}_a$) accounting for additional drag (see below)
- thrust specific fuel consumption (**TSFC**), based on the *effective* specific thrust
- propulsive, thermal and overall engine efficiencies (η_P, η_{th}, η_o), again based on *effective* thrust
- maximum allowed fuel/air ratios (f_{max}, $f_{max,ab}$), based on each burner's maximum temperature limits (T_{max}, $T_{max,ab}$)

Note, you should write the **simplest forms for the equations** that you can. The output conditions for a component (real or virtual) should only be a function of: 1) inputs to that component; 2) the component/flow properties, e.g., efficiencies, c_p/R, etc.; and 3) other outputs from that component if you already have an equation for them. For example, do not try and write

an expression for a nozzle exit velocity based on the aircraft speed, ambient temperature, etc., instead write it in terms of temperatures entering and exiting the nozzle and gas properties (e.g., $\overline{W}$, c_p/R). As another example, if you need to use the compressor work to determine the turbine exit temperature, **do not** insert a complicated equation for the compressor work in your turbine equation; just use the compressor work as an "input" variable. Similarly, write an expression for TSFC in terms only of specific thrust and fuel-air ratio(s).

Additional Models/Assumptions

1. Assume all components are **adiabatic** (meaning heat transfer to/from the surroundings is zero).

2. Assume that the nozzle exit pressures are always equal to the ambient pressure (**perfectly expanded nozzles**, even under supersonic conditions).

3. For the fan, compressor and turbines, the adiabatic efficiency will be modeled using the concept of the **polytropic efficiency** η_p. It turns out that well-designed, modern turbomachines have similar polytropic efficiencies, independent of the overall pressure ratio. The polytropic efficiency is the efficiency of an adiabatic device with an infinitesimal (very small) pressure change, and like the adiabatic efficiency, $\eta_p<1$. Since a finite pressure change can be thought of as many infinitesimal pressure changes, we can produce a relationship between the adiabatic efficiency of a real device and the polytropic efficiency.

   For a compressor or fan with a stagnation pressure ratio Pr_i ($=p_{o,outlet}/p_{o,inlet}$) and a constant polytropic efficiency η_{pi}, the **adiabatic efficiency is** given by

   $$\eta_i = \frac{Pr_i^{R/c_p}-1}{Pr_i^{(R/c_p)/\eta_{p,i}}-1}$$

   Similarly, **for a turbine** with a stagnation temperature ratio Tr_i ($=T_{o,outlet}/T_{o,inlet}$), the **adiabatic efficiency** can be shown to be given by

   $$\eta_i = \frac{Tr_i-1}{Tr_i^{1/\eta_{p,i}}-1}.$$

   For the turbine, it is convenient to write the efficiency as a function of the temperature ratio since that is what we first calculate (then we find the pressure ratio from the efficiency).

4. For the fuel pump, assume that the fuel is stored at a pressure that is $\Delta p_{f,1}$ above the local ambient pressure and that the pump must produce an exit pressure that is higher by an amount Δp_{inj} than the <u>main</u> combustor/burner inlet pressure.

5. Neglect any mechanical power transmission loss between turbines and the devices they drive using shafts (e.g., assume shaft/transmission efficiencies of 100%).

6. For an engine flying at supersonic speeds, there will usually be additional stagnation pressure losses outside the engine, due to the formation of shock waves. Account for this by including a **ram recovery factor** (r_d) to find the stagnation pressure at the diffuser exit, i.e.,

   $$p_{o1} = r_d \times p'_{o1}$$

   where p'_{o1} is the diffuser exhaust stagnation pressure that would be calculated from the adiabatic efficiency η_d (and which accounts for boundary layers in the inlet). A standard model for r_d for a well-designed inlet is based on MIL-E-5008B:

$$r_d = \begin{cases} 1 & \text{for } M \leq 1 \\ 1 - 0.075(M-1)^{1.35} & \text{for } 1 < M < 5 \end{cases}$$

7. When you bleed air from the compressor (at its outlet) to cool the main turbine, assume the maximum allowable turbine inlet temperature T_{max} (one of the engine design constraints) will increase according to the following expression:

$$\mathbf{T_{max}} = T_{max,0} + C_b \cdot (b/b_{max})^n$$

where $\mathbf{T_{max,0}}$ is the maximum allowable temperature without bleed, $\mathbf{C_b}$ and $\mathbf{n}$ are constants, and $\mathbf{b_{max}}$ is the maximum allowed bleed fraction. (Essentially, we are saying the blade cooling is proportional to the velocity of the air passing through the blades.) To simplify the actual physics of the reintroduction of the air through the holes in the turbine blades, we will assume the bleed air returns to the main air flow *just after the turbine exhaust* and is mixed with the turbine exhaust in our virtual component, **the turbine mixer**. To account for irreversibilities (stagnation pressure losses) in this process, assume the bleed air enters the virtual mixer with the same stagnation temperature it had when it exited the compressor, but with a (lower) stagnation pressure equal to the turbine exhaust stagnation pressure.

8. You can only use forward mounted fans, and the **maximum allowed bypass ratio** is β_{max}.

   However, adding the turbofan increases the size of the engine (relative to the size of the core engine). This should increase the drag on the aircraft. So assume that the **specific increase in the drag** ($\Delta d \equiv Drag\,Increase/\dot{m}_a$, e.g., N/(kg/s)) is given by a modified form of a drag coefficient,

$$\mathbf{\Delta d} = C_\beta \cdot M^2 \cdot (p_a/p_{STP}) \cdot \beta^{1.5}$$

where $\mathbf{C_\beta}$ is a constant, M is the flight Mach number, p_a is the ambient pressure of the air at the cruise conditions, and p_{STP} is the standard atmospheric pressure at sea-level. Thus, the **effective thrust** of the engine is $\tau/\dot{m}_a\big|_{effective} = \tau/\dot{m}_a - \Delta d$.

9. When you use the combined nozzle, it is not realistic to assume the nozzle mixer is completely reversible. To try to account for the mixer losses, assume that when you use a combined nozzle, that the nozzle mixer exhaust pressure is given by

$$p_{o7} = p_{o7,rev} \times \mathbf{Pr_{nm}},$$

where $p_{o7,rev}$ is the stagnation pressure that would be calculated if the mixing was reversible and $\mathbf{Pr_{nm}}$ provides the effective pressure loss (or more accurately, Pr_{nm} is one minus the fractional stagnation pressure loss). We will model this term as

$$\mathbf{Pr_{nm}} = e^{-C_{nm}/(1+m_r^{0.5})} \text{ and } \mathbf{m_r} = \dot{m}_{max}/\dot{m}_{min}$$

where $\mathbf{C_{nm}}$ is a constant, $\dot{m}_{max}$ is the higher of the two mass flowrates entering the nozzle mixer and $\dot{m}_{min}$ is the lower of the two.

Model Implementation

You should implement the equations you have developed into a single model (code) that can simulate an engine with any (reasonable) combination of the components defined above. You should not try to create separate codes or models to handle different types of engines (e.g., don't write one code just for ramjets and another for turbofans). Rather by adjusting the design parameters, you can simulate any of our turbine engine cycles. For example, choosing the

bypass ratio β=0, Pr$_f$=1 and a compressor pressure ratio Pr$_c$>1 would mean you are simulating a turbojet; similarly, using β=0, Pr$_f$=1 and Pr$_c$=1 would mean you are simulating a ramjet.

2. Engine Cycle Simulation/Analysis and Cycle Design

Your team has been tasked to perform a cycle design for jet engines to power a supersonic business jet; you will use the cycle analysis simulations described in the previous section. The manufacturer has asked you to focus on four important flight conditions (see Table 1). Based on the allowed size of the engines and the thrust needed by the aircraft, a specific thrust requirement for the engine has been issued for each flight condition (also in Table 1).

Table 1. Flight conditions and required thrusts

Flight Condition	Altitude	T$_a$	P$_a$	M	Required Specific Thrust
	km//kft	K//R	kPa//psia		kN·s/kg // lb$_f$·s/lb$_m$
Max Takeoff Thrust (MTO)	0	288//518	101.3//14.7	0.20	1.23//125
Begin Transonic Cruise (BTR)	10.0//32.8	223//402	26.4//3.83	0.95	1.05//107
End Transonic Cruise (ETR)	10.0//32.8	223//402	26.4//3.83	0.95	1.46//149
Supersonic Cruise (SCR)	16.3//53.5	216//389	9.81//1.42	1.6	0.874//89.1

Your **first goal** is to identify four engine configurations, one for each flight condition in Table 1 that provides *the required specific thrust level* (and no more), while providing high fuel efficiency, i.e., *a low SFC*. In this case, a configuration means a unique set of the design parameters defined in the previous section.

Your **second goal** is to determine the <u>maximum thrust</u> (and corresponding SFC) at all four flight conditions for each of your four engine configurations. To determine the maximum thrust of your engine cycle you need to maintain a fixed set of components (e.g., you can't add an afterburner if not using one in that configuration initially, nor can you switch from separate nozzles to a combined to determine the maximum thrust); AND you cannot change the compressor and fan pressure ratios (Pr$_c$, Pr$_f$), the bypass ratio (β) or the nozzle mixer split ratio (σ). You <u>can</u> change the fuel/air ratios and bleed ratio.

Your **third goal** is to suggest a *single engine configuration* for the vehicle (not necessarily one of those chosen in the first goal), taking into account both of its flight conditions and requirements, and any other considerations you feel might make the engine more "desirable" to the aircraft manufacturer. Please note; there is **no "single right answer"** to these designs. Come up with the best designs based on your analysis and engineering insight.

Component Characteristics and Properties

Typical characteristics for each engine component (efficiencies, specific heat ratios, model coefficients, etc.) that you will need are listed in Table 2. Careful to note which efficiencies are adiabatic efficiencies (*) and which are polytropic efficiencies (**).

Table 2. Characteristics of the turbine engine components.

Component	η_i	Pr_i	MolecWt	c_p/R	Other
Diffuser (d)	0.94[*]		28.9	3.5	
Fan (f)	0.92[**]	$1.15 \leq Pr_f \leq 1.7$ if used	28.9	3.5	β_{max}=15, C_β=0.263 kN·s/kg (26.8 lb$_f$·s/lb$_m$) (*if β=0, you can't have a fan*)
Compressor (c)	0.91[**]	$Pr_c < 55/Pr_f$	28.9	3.62	
Main Burner (b)	0.99	0.95 (if used	28.9	Note 1	Heating Value$_{fuel}$= 43.52 MJ/kg or 18,710 BTU/lbm
Turbine (t)	0.94[**]		28.9	Note 2	$T_{max,0}$ = 1400 K (2520 R) b_{max}=0.15, n=0.6, C_b=700 K (1260 R)
Turbine Mixer (tm)			28.9	Note 3	
Fan Turbine (ft)	0.94[**]		28.9	Note 4	
Afterburner (ab)	0.96	0.97 (if used)	28.9	Note 5	Same heating value as main, $T_{max,ab}$ = 2300K (4140 R)
Core Nozzle (n)	0.96[*]		28.9	Note 6	
Fan/Bypass Nozzle (fn)	0.97[*] if used		28.9	Note 7	
Nozzle Mixer (nm)			28.9	Note 7	C_{nm}=2.00
Combined Nozzle (cn)	0.96[*]		28.9	Note 8	
Fuel Pump (p)	0.48[*]				ρ_f=780 kg/m^3 (48.7 lb$_m$/ft^3) $\Delta p_{f.1}$= 20.7 kPa (3.00 lb$_f$/in^2) Δp_{inj}=572 kPa (83.0 lb$_f$/in^2)

[*]Adiabatic/isentropic efficiency

[**]Polytropic efficiency

[1] $c_{p,b}/R = 3.70 + 0.660 (T_{o3}/1000K)^2 - 0.200 (T_{o3}/1000K)^3$

[2] $c_{p,t}/R = 3.38 + 0.700 (T_{o4}/1000K)^2 - 0.200 (T_{o4}/1000K)^3$

[3] hot flow (from turbine): $c_{p,tm\text{-}hot}/R = 3.43 + 0.780 (T_{o5.1}/1000K)^2 - 0.270 (T_{o5.1}/1000K)^3$
  cold/bleed flow (from compressor): $c_{p,tm\text{-}cold}/R = 3.70 + 0.780 (T_{o3}/1000K)^2 - 0.360 (T_{o3}/1000K)^3$

[4] $c_{p,ft}/R = 3.40 + 0.630 (T_{o5.1m}/1000K)^2 - 0.200 (T_{o5.1m}/1000K)^3$

[5] $c_{p,ab}/R = 3.50 + 0.720 (T_{o5.2}/1000K)^2 - 0.210 (T_{o5.2}/1000K)^3$

[6] $c_{p,n}/R = 3.45 + 0.550 (T_{o6}/1000K)^2 - 0.150 (T_{o6}/1000K)^3$

[7] core flow: $c_{p,nm\text{-}core}/R = 3.44 + 0.790 (\overline{T}_{o,core}/1000K)^2 - 0.270 (\overline{T}_{o,core}/1000K)^3$
  bypass flow: $c_{p,nm\text{-}by}/R = 3.43 + 0.790 (\overline{T}_{o,by}/1000K)^2 - 0.280 (\overline{T}_{o,by}/1000K)^3$
    with $\overline{T}_{o,core} = (T_{o6}+T'_{o7})/2$
        $\overline{T}_{o,by} = (T_{o2}+T'_{o7})/2$
        T'_{o7} = *estimated* nozzle mixer output temperature if both flows had same c_p/R

[8] $c_{p,cn}/R = 3.45 + 0.550 (T_{o7}/1000K)^2 - 0.150 (T_{o7}/1000K)^3$

3. Written Report

Your team will prepare a *professional looking* report to describe your designs and analysis. The written report (to be turned in online) should have a separate **Cover Page** and the following five sections: 1) **Executive Summary,** 2) **Introduction,** 3) **Approach**, 4) **Results and Discussion** and 5) **Appendices**. (*You must incorporate the sections in this order and with the above titles, but not necessarily the numbering*.) Except for the Executive Summary, you are free to further divide each section into subsections with appropriate headings.

The Executive Summary should not exceed one single-sided page (and should appear on its own, separate page). Combined, the next three sections (Introduction through Results and Discussion) should be **no longer than 16 single-sided pages** using a font no smaller than 11 pt. This page limit *includes* all tables and figure*s* in those sections. Note: there is no advantage to handing in a long report (your report can be shorter than the maximum page limit); covering the material described below *clearly* and *concisely* will likely improve your report's readability, and therefore your group's grade.

The **Cover Page** should at least include (in any order): the names of the team's members, a title for your report, and the date.

The **Executive Summary** is a *brief* but helpful statement of the design problem and what you were trying to accomplish (so that a person unfamiliar with your task would have a chance of understanding what you had done) and a **summary of your important findings and conclusions** (this would include - in part – brief but **quantitative** descriptions of your four engine designs with some meaningful overall performance results, and your recommendations for the final/overall engine configuration). The Executive Summary has to *sell* your ideas to a high-level executive; they need to know what you accomplished and what you recommend without having to read the detailed report.

The main body of your report (**Introduction**, **Approach** and **Results and Discussion** sections) should provide all the necessary information for the reader; they should not depend on the reader having read the Executive Summary.

The **Introduction** section should *completely* describe the problem you are trying to solve and clearly explain the overall and detailed design goals. Assume your reader has not seen anything about what you are doing, so make sure your Introduction lets them understand what you have been asked to do. Also, use your own words to do this; *do not copy the text from this document*.

In the **Approach** section, discuss your team's approach to solving the problem. The description of your approach should include, *at a minimum*, a description of the equations you developed and a reference to the equation list in the Appendices, the computer tool(s) used to solve the equations, and how you attacked the problem - what was your "game plan" to determine what sets of design parameters produce the best engines, and how to determine the maximum specific thrust for each configuration It should **not** include any results or findings/conclusions.

The **Results and Discussion** section should contain a clear presentation of your team's choices, numerical results and a meaningful discussion of those results (in the context of presenting your preliminary designs), for example why your design choices "make sense". You need to present results (as explained below) for the four individual "optimized" engine cycle designs and for your overall ("best") engine design. For each of the four individual designs (one for each flight condition), you must provide:

- your design choices for the "inputs", e.g., β [report 0 if not using a turbofan], $\mathbf{Pr_f}$ [if fan used], $\mathbf{Pr_c}$ [report 1 if not using a compressor], $\mathbf{b}$, f, and f_{ab} [if afterburner used] and whether you are using separate or combined nozzles (and σ if using a combined nozzle).

- each of the output results described in Table 3 for that design's flight condition

For all four individual engine designs, you also need to provide:

- the maximum possible **ST** and corresponding **SFC** values for all four flight conditions (so a total of 16 different pairs of results, 4 for each individual engine).

For the overall engine design, you must provide

- your fixed design choices for: β [report 0 if not a turbofan], $\mathbf{Pr_f}$ [if used], $\mathbf{Pr_c}$ [report 1 if not using a compressor], and whether you are using separate nozzles or a combined nozzle (and σ if combined nozzle used) - these values can't change between the vehicle's various flight conditions

- at each of the vehicle's four flight conditions, your choices for the *variable design parameters*: b, f, and f_{ab} [if used] AND the *performance results*: ST and the corresponding SFC

Remember to use **effective thrust** in your results and the units listed in Table 3 for each parameter. Please present the results in a clear and user-friendly form, for example you could use a table or list the values on a block diagram of the engine. In your discussion explaining your design choices, you could also consider presenting plots of engine performance versus variations in some of the design parameters (this is not a requirement, but may help you provide a clearer discussion of your results). **Do not present any of the required data by placing it in an appendix.**

Table 3. Engine (output) parameters required for each flight condition.

- effective specific thrust (**kN·s/kg or lbf·s/lbm**) based on core-air mass flow, including added drag
- TSFC (**kg/kN·hr or lbm/lbf·hr**), based on effective specific thrust
- exit velocities (**m/s or ft/s**) from core, fan or combined nozzles [whichever used]
- exit Mach numbers from core, fan or combined nozzles [whichever used]
- fuel/air ratios in main burner and afterburner [if used]
- maximum allowed fuel/air ratios in main burner and afterburner [if used] - BE SURE to include bleed cooling effects on main burner f_{max}
- propulsive, thermal and overall engine efficiencies (**%**); base your results on effective thrust
- required power per unit core-air flowrate (**kJ/kg or hp·s/lbm**) for the fuel pump, compressor and fan [if used]
- output power per unit core-air flowrate (**kJ/kg or hp·s/lbm**) for the main and fan turbines [if used]
- exit temperatures (**K or R**) and pressures (**kPa or psia**) for each component (use stagnation values for all components except nozzles, where static values are required), and except the fuel pump, for which you only need to report the exit stagnation pressure

The **Appendices** should <u>at least</u> include two sections. The first, **Appendix A - Engine Model Diagram**, should incorporate a flow diagram of the generalized engine showing the fluid and power relationships/connections. The diagram should have <u>all</u> the components described in this project, and the state numbering scheme should be indicated on the diagram. The second, **Appendix B - Equations**, should list the "final" equations for all the output variables listed in the Equation Development section; please write, **simple, short versions of the equations.**

4. Verification Data

The following results, which are only a subset of the values you are required to calculate, are provided as a means of testing and debugging your computer tool. The design values were chosen at random for this purpose; do not use them as guideposts for your design. The input data are the exact values indicated in the tables; the output data are reported to the number of significant figures indicated in the tables.

Inputs

T_a (K)	P_a(kPa)	M	Pr_c	Pr_f	β	b	f	f_{ab}
220	11.0	1.10	15	1.2	1.5	0.060	0.025	0.0050

Outputs (for separate nozzles)

T and p values

T_{o1} (K)	p_{o1} (kPa)	T_{o2} (K)	p_{o2} (kPa)	T_{o3} (K)	p_{o3} (kPa)	T_{o4} (K)	p_{o4} (kPa)	$T_{o5.1}$ (K)	$p_{o5.1}$ (kPa)
273.2	22.46	289.2	26.96	657.9	404.4	1628	384.1	1312	140.6

$T_{o5.1m}$ (K)	$p_{o5.1m}$ (kPa)	$T_{o5.2}$ (K)	$p_{o5.2}$ (kPa)	T_{o6} (K)	p_{o6} (kPa)	T_e (K)	T_{ef} (K)	$p_{f.1}$ (kPa)	$p_{f.2}$ (kPa)
1275.	147.9	1242	131.9	1403	127.9	798.5	225.8	31.70	976.4

Specific work and max FAR values

w_c (kJ/kg)	w_p (kJ/kg)	w_t (kJ/kg)	w_{ft} (kJ/kg)	f_{max}	$f_{max,ab}$
384.0	0.076	384.1	40.07	0.0297	0.0337

Nozzle and Selected Performance Values

u_e (m/s)	M_e	u_{ef} (m/s)	M_{ef}	$\tau/\dot{m}_a$ (kN s/kg)	TSFC (kg/kN·hr)	η_{th} (%)	η_o (%)
1197	2.17	357.2	1.18	0.887	121.8	53.6	22.2

Outputs (for combined nozzles using σ=0.72)

Additional T, p and Nozzle Values, and Selected Performance Values

T_{o7} (K)	p_{o7} (kPa)	T_{ec} (K)	u_{ec} (m/s)	M_{ec}	$\tau/\dot{m}_a$ (kN s/kg)	TSFC (kg/kN·hr)	η_p (%)	η_o (%)
1105	78.19	670.7	959.2	1.88	0.895	120.7	48.7	22.4