Project: Jet Engine Cycle Design

In this assignment, you will explore parametric engine cycle analysis for design of airbreathing aircraft engines. This is a **group/team** project (*members 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. Helpful conversions:

R=8.3145 kJ/kmol·K=1545 ft·lbf/lbmol·R=1.987 BTU/lbmol·R; **1 lb$_f$**=32.2 lb$_m$·ft/s^2=1 slug·ft/s^2

1. Engine Cycle Equation Development and Simulation

In this project, your team will be using an engine cycle design approach to choose different 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 nozzles; and **(3)** adding an **interturbine** combustor (an extra combustor located between turbines, between the "compressor turbine" and "fan turbine" for our project). 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**)
- bypass fan (**f**)
- compressor (**c**)
- fuel pump (**p**)
- main combustor/burner (**b**)
- afterburner (**ab**)

- turbine (**t**): runs compressor and fuel pump in our simulation
- fan turbine (**ft**): runs the fan in our simulation
- interturbine burner (**ib**)
- core nozzle (**n**)
- fan 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**)

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

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

– ambient	a	– fan turbine exhaust	5.2
– diffuser exhaust	1	– afterburner exhaust	6
– bypass fan exhaust	2	– core nozzle exhaust	e
– compressor exhaust	3	– fan nozzle exhaust	ef
– main combustor/burner exhaust	4	– nozzle mixer exhaust	7
– turbine exhaust	5.1	– combined nozzle exhaust	ec
– turbine mixer exhaust	5.1m	– fuel storage	f.1
– interturbine burner exhaust	5.14	– fuel pump exhaust	f.2

Inputs

The equations you develop should depend on the **following flight conditions**:

- ambient temperature ($\mathbf{T_a}$), ambient pressure ($\mathbf{p_a}$); fuel storage pressure ($\mathbf{p_f}$)
- flight Mach number (**M**);

the **design parameters**:

- compressor (stagnation) pressure ratio ($\mathbf{Pr_c}$);
- fan (stagnation) pressure ratio ($\mathbf{Pr_f}$);
- fuel-air ratios for the main burner (***f***), interturbine burner ($\boldsymbol{f_{ib}}$), and afterburner ($\boldsymbol{f_{ab}}$) with $f = \dot{m}_f / \dot{m}_a$, $f_{ib} = \dot{m}_{f\,ib}/\dot{m}_a$ 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**), defined to be the ratio of the mass of secondary air bypassing the compressor to the air flow rate entering the compressor ($\beta = \dot{m}_s / \dot{m}_a$);
- bleed ratio (**b**), defined as the flow rate of air bled from compressor over mass flow rate of air entering the compressor ($b = \dot{m}_b / \dot{m}_a$);

and the following **component/flow properties**:

- average normalized specific heats ($\mathbf{c_{p,i}/R}$), and gas molecular weight ($\overline{\mathbf{W}}_\mathbf{i}$) across each component;

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

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

- temperature limits for the main burner ($\mathbf{T_{max}}$), interturbine burner ($\mathbf{T_{max,ib}}$) and afterburner ($\mathbf{T_{max,ab}}$)
- maximum allowed bleed fraction ($\mathbf{b_{max}}$)

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 ($\mathbf{T_{oi}}$) and pressures ($\mathbf{p_{oi}}$) 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 ($\mathbf{T_i}$), velocities ($\mathbf{u_i}$) and Mach numbers ($\mathbf{M_i}$) from each of your nozzles (core and fan nozzles, or combined nozzle)
- exit stagnation pressure for the fuel pump ($\mathbf{p_{f.2}}$)
- power per unit mass **core** air flow rate ($\mathbf{w_i} = \dot{W}_i/\dot{m}_a$) for the fan, compressor, main turbine, fan turbine, and fuel pump
- effective specific thrust ($\boldsymbol{\tau/\dot{m}_a}$) accounting for additional drag (see below)
- thrust specific fuel consumption (**TSFC**), based on the *effective* specific thrust
- propulsive, thermal and overall engine efficiencies ($\boldsymbol{\eta_p, \eta_{th}, \eta_o}$), again based on *effective* thrust
- maximum allowed fuel/air ratios ($\boldsymbol{f_{max}, f_{max,ib}, f_{max,ab}}$), based on each burner's maximum temperature limits ($\mathbf{T_{max}, T_{max,ib}, T_{max,ab}}$)

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 (or γ depending on which value is given for that component), 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 u_e 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 the heat transfer to 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 P_r (outlet/inlet) and a constant polytropic efficiency η_{pi}, the **adiabatic efficiency is** given by

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

   where Pr_i is the stagnation pressure ratio across the fan or compressor.

   Similarly, **for a turbine** with a stagnation temperature ratio Tr_i (outlet/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 typically 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 $\mathbf{\Delta p_{inj}}$ than the main 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** ($\mathbf{r_d}$) in the diffuser exit (stagnation) pressure, i.e.,

   $$\mathrm{p}_{o1} = \mathbf{r_d} \times \mathrm{p}'_{o1}$$

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:

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

7. When you bleed air from the compressor (at its outlet) to cool the main turbine, assume the maximum allowable (i.e., the constraint on the) turbine inlet temperature T_{max} will increase according to the following expression:

$$\mathbf{T_{max}} = T_{max,0} + C_{b1} \cdot (b/b_{max})^n$$

   where $T_{max,0}$ is the maximum allowable temperature without bleed and 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 account for 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**. The bleed air enters with the same stagnation temperature it had when it exited the compressor, but with a lower stagnation pressure equal to the turbine exhaust 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,

$$\mathbf{\Delta d} = C_{\beta 1} \cdot M^2 \cdot (p_a/p_{atm}) \cdot \beta^{1.5}$$

   where M is the flight Mach number, p_a is the ambient pressure of the air at the cruise conditions, and p_{atm} 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 the one minus the fractional stagnation pressure loss). We will model this term (where C_{nm} is a constant) as

$$\mathbf{Pr_{nm}} = 1 - C_{nm} \cdot \beta$$

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 just 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 suggest you are simulating a ramjet.

2. Engine Cycle Simulation/Analysis and Preliminary Cycle Design

Your team has been tasked to perform a preliminary design for jet engines to power two aircraft: a commercial airliner and a high performance aircraft; you will use the cycle analysis simulations described in the previous section. For each vehicle, the manufacturer has asked you to focus on two 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 has been issued for each flight condition (also in Table 1).

Table 1. Flight conditions and required thrusts

Vehicle	**Flight Condition**	**Altitude** km//kft	**T_a** K//R	**P_a** kPa//psia	**M**	**Required Specific Thrust** kN·s/kg // lb_f·s/lb_m
Commercial Airliner	Max Takeoff (MTO) Thrust (SL Static Thrust)	0	298//519	101.3//14.7	0	3.25// 331
	Max Cruise Thrust (MCR) @High Alt Cruise	10.7//35.0	219//394	23.8//3.45	0.85	0.945// 96.3
High Performance Aircraft	MCR Thrust @High Altitude Subsonic Cruise	13.7//45.0	217//391	14.7//2.13	0.88	0.810//82.5
	MCR Thrust @High Altitude Supersonic Cruise	15.2//50.0	216//389	11.6//1.68	1.6	1.28//130

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 efficiency, i.e., *a low SFC*. In this case, a configuration means a unique set of the design parameters defined above.

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, nor can you switch from separate nozzles to a mixed nozzle to determine the maximum thrust); AND you cannot change the compressor and fan pressure ratios (Pr_c, Pr_f) and the bypass ratio (β). You <u>can</u> change the fuel/air ratios and bleed ratio.

Your **third goal** is to suggest a *single engine configuration* (not necessarily one of those chosen in the first goal) for each vehicle, 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**	$1.15 \le Pr_f \le 1.7$ if used	28.9	3.5	β_{max}=15, $C_{\beta 1}$=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	$T_{max,0}$ = 1500 K (2700 R) b_{max}=0.10, n=0.6, C_{b1}=500 K (900 R)
Turbine Mixer (tm)			28.9	Note 3	
Interturbine Burner (ib)	0.99	0.94 (if used)	28.9	Note 4	Same heating value as main, $T_{max,ib}$ = 1500K (2700 R)
Fan Turbine (ft)	0.94**		28.9	Note 5	
Afterburner (ab)	0.96	0.97 (if used)	28.9	Note 6	Same heating value as main, $T_{max,ab}$ = 2300K (4140 R)
Core Nozzle (n)	0.96*		28.9	Note 7	
Fan Nozzle (fn)	0.97* if used		28.9	3.5	
Nozzle Mixer (nm)			28.9	Note 8	C_{nm}=0.029
Combined Nozzle (cn)	0.96*		28.9	Note 9	
Fuel Pump (p)	0.45*				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}=540. kPa (78.3 lb_f/in^2)

*Adiabatic/isentropic efficiency **Polytropic efficiency

[1] $c_{p,b}/R = 3.70 + 0.66\,(T_{o3}/1000K)^2 - 0.20\,(T_{o3}/1000K)^3$

[2] $c_{p,t}/R = 3.38 + 0.70\,(T_{o4}/1000K)^2 - 0.20\,(T_{o4}/1000K)^3$

[3] $c_{p,tm}/R = 3.43 + 0.72\,(T_{o5.1}/1000K)^2 - 0.21\,(T_{o5.1}/1000K)^3$

[4] $c_{p,ib}/R = 3.82 + 0.32\,(T_{o5.1m}/1000K)^2 - 0.060\,(T_{o5.1m}/1000K)^3$

[5] $c_{p,ft}/R = 3.40 + 0.63\,(T_{o5.14}/1000K)^2 - 0.18\,(T_{o5.14}/1000K)^3$

[6] $c_{p,ab}/R = 3.50 + 0.72\,(T_{o5.2}/1000K)^2 - 0.21\,(T_{o5.2}/1000K)^3$

[7] $c_{p,n}/R = 3.45 + 0.55\,(T_{o6}/1000K)^2 - 0.15\,(T_{o6}/1000K)^3$

[8] $\gamma_{nm} = 1.44 - 0.139\,(T_{o7}/1000K) + 0.0357\,(T_{o7}/1000K)^2 - 0.004\,(T_{o7}/1000K)^3$

[9] $c_{p,cn}/R = 3.45 + 0.55\,(T_{o7}/1000K)^2 - 0.15\,(T_{o7}/1000K)^3$

3. Written Report

Your team will prepare a *professional looking* report of 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*.)

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. 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 reports readability, and therefore you 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 two final engines). The Executive Summary has to *sell* your ideas to a high level executive so they know what you accomplished and what you recommend without having to read the details.

The main body of your report (**Introduction**, **Approach** and **Results and Discussion** sections) should provide all the necessary information for the reader, and be written as if there was no Executive Summary section.

The **Introduction** section should completely describe the problem you are trying to solve and clearly explain the specific and detailed design goals. 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 engine cycles, and how to determine the maximum specific thrust for each cycle. It should NOT include any results or findings/conclusions.

The **Results and Discussion** section should contain a clear presentation of your team's numerical results and a meaningful discussion those results (in the context of presenting your preliminary designs). You need to present results (as explained below) for the four individual "optimized" engine cycle designs and for your overall ("best") two engine designs. For 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], Pr_f [if used], Pr_c [report 1 if not using a compressor], b, f, f_{ib} [if used], and f_{ab} [if used] and whether you are using separate or combined nozzles
- each of the output results described in Table 3 for that design's flight condition

- the maximum possible ST and corresponding SFC values for each design at all four flight conditions.

For the two overall designs (one for each vehicle), you must provide

- your fixed design choices for: β [report 0 if not a turbofan], Pr_f [if used], Pr_c [report 1 if not using a compressor], and whether you are using separate nozzles or a combined nozzle (these can't change between the vehicle's two flight conditions)
- at each of the vehicle's two flight conditions, your choices for the <u>variable design parameters</u>: b, f, f_{ib} [if used], and f_{ab} [if used] AND the <u>performance results</u>: ST and the corresponding SFC

Remember to use **effective thrust** in all your results, and use 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 you could 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, interturbine burner and afterburner [if used]
- maximum allowed fuel/air ratios in main burner, interturbine 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 turbine [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; as noted previously, **please write, simple, short versions of the equations.**

Test Case

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; so do not use them as guideposts to your design.

Inputs

T_a (K)	P_a(kPa)	M	Pr_c	Pr_f	β	b	f	f_{ib}	f_{ab}
220	11.0	1.10	15	1.2	2.0	0.06	0.021	0.0040	0.0050

Outputs

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)	$T_{o5.1m}$ (K)	$p_{o5.1m}$ (kPa)
273.2	22.46	289.2	26.96	657.9	404.4	1477	384.1	1151	124.0	1122.	127.7

$T_{o5.14}$ (K)	$p_{o5.14}$ (kPa)	$T_{o5.2}$ (K)	$p_{o5.2}$ (kPa)	T_{o6} (K)	p_{o6} (kPa)	T_e (K)	T_{ef} (K)	T_{o7} (K)	p_{o7} (kPa)	T_{ec} (K)	
1258	120.1	1218	104.4	1380.	101.2	826.8	225.8	660.1	120.2	355.3	

$p_{f.1}$ (kPa)	$p_{f.2}$ (kPa)
31.70	944.4

Additional Values

w_c (kJ/kg)	w_p (kJ/kg)	w_t (kJ/kg)	w_{ft} (kJ/kg)	f_{max}	$f_{max,ib}$	$f_{max,ab}$
384.0	0.078	384.1	48.08	0.0314	0.0111	0.0343

Performance: Selected Values for Combined Nozzle

u_{ec} (m/s)	M_{ec}	$\tau/\dot{m}_a$ (kN s/kg)	TSFC (kg /kN·hr)	η_p (%)	η_o (%)
799.7	2.13	1.343	80.41	54.4	33.7

Performance: Selected Values for Separate Nozzles

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 (%)
1143	2.04	357.2	1.18	0.8121	133.0	49.0	20.4