Project: Turbine Engine Cycle Design

In this assignment, you will explore parametric turbine engine design and approaches for improving engine performance. This is a group (3 person) project. Your group will: 1) develop the equations necessary to model the engine cycles; 2) create a computer model to simulate the engine cycles using any programming language or modeling tool (such as a spreadsheet) that you wish as long as the group develops its own software; 3) perform the engine cycle analysis; and 4) prepare a report (described later in this handout).

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

\[
R = 8.314 \text{ kJ/kmol·K} = 1545 \text{ ft·lb/lbmol·R} = 1.987 \text{ BTU/lbmol·R}
\]
\[
1 \text{ lb} = 32.2 \text{ lbm·ft/s}^2 = 1 \text{ slug·ft/s}^2
\]

Engine Cycle Equation Development

You will be designing various engines to meet several aircraft propulsion missions. You can consider all the standard engine cycles: ramjets, turbojets and turbofans. You should also consider the following additions/modifications to improve the performance of your engines: (1) bleeding compressor air to cool the turbine blades, thereby increasing the maximum allowable turbine inlet temperature; (2) adding an interturbine combustor (an extra combustor between the compressor turbine and the fan turbine); (3) adding an afterburner; and (4) using a combined nozzle for the core and bypass flows.

Engine Components and State Numbering

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

- diffuser (d)
- bypass fan (f)
- compressor (c)
- fuel pump (p)
- main combustor/burner (b)
- turbine (t): runs compressor and fuel pump
- interturbine burner (ib)
- fan turbine (ft): to run the fan
- afterburner (ab)
- core nozzle (n)
- fan nozzle (fn)
- combined nozzle (cn) [replaces core and fan nozzles]

To model the bleed air from the compressor mixing with the turbine flow, or the core and fan flows mixing in the combined nozzle, you should add “virtual” components to the engine, as you did 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. This may not be the same scheme used in the text or class, but using it will allow me to more easily check your equations.

- ambient  a  - fan turbine exhaust  5.2
- diffuser exhaust  1  - afterburner exhaust  6
- fan exhaust  2  - core nozzle exhaust  e
- compressor exhaust  3  - fan nozzle exhaust  ef
- burner exhaust  4  - nozzle mixer exhaust  7
- turbine exhaust  5.1  - combined nozzle exhaust  ec
- turbine mixer exhaust  5.m  - fuel storage  f.1
- interturbine burner exhaust  5.i  - fuel pump exhaust  f.2

**Inputs**

The equations you develop should depend on the following design parameters:

- ambient temperature \((T_a)\), ambient pressure \((p_a)\); fuel storage pressure \((p_f)\)
- flight Mach number \((M)\);
- compressor (stagnation) pressure ratio \((P_{rc})\);
- fan (stagnation) pressure ratio \((P_{rf})\);
- fuel-air ratios for the main burner \((f)\), interturbine burner \((f_{ib})\), and afterburner \((f_{ab})\) 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 entering the compressor;
- bypass ratio \((\beta)\), defined to be the ratio of the mass of secondary air bypassing the compressor to the air flow rate entering the compressor \((\beta = \dot{m}_b / \dot{m}_a)\);
- 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 \((c_{p,i}/R)\) and gas molecular weight \((\bar{M}_i)\) across each component;
- adiabatic efficiencies \((\eta_i)\) for the nozzles, diffuser, and fuel pump; 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 \((\eta_b, \eta_{ib}, \eta_{ab})\) and stagnation pressure ratios \((P_{rb}, P_{rib}, P_{rab})\) for the burner, interturbine burner and afterburner;
- fuel heating value \((\Delta h_R)\) and fuel density \((\rho_f)\) - assume the same fuel for all combustors;
- the specific drag loss \((\Delta d)\) associated with the bypass fan (see below for explanation);
• the pressure loss function for the virtual nozzle mixer (Prnm, see below).

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

• temperature limits for the main burner (Tmax), interturbine burner (Tmax,ib) and afterburner (Tmax,ab)
• maximum allowed bleed fraction (bmax)

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 temperatures (Ti) and pressures (pi) for each component except the fuel pump (these should be the stagnation temperature and pressures for all components except nozzles)
• for the fan, compressor and turbines, write your outputs (where appropriate) in terms of the polytropic efficiencies, see below (DO NOT REPORT EQUATIONS THAT USE THE ADIABATIC efficiencies)
• exit pressure for the fuel pump
• 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
• exhaust velocity from the core nozzle, fan nozzle, combined nozzle (u_e, uef, uec)
• 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 (\eta_p, \eta_th, \eta_o), again based on effective thrust
• maximum allowed fuel/air ratios (fmax, fmax,ib, fmax,ab), based on each burner’s maximum temperature limits (Tmax, Tmax,ib, Tmax,ab)

Note, you should write the simplest forms for the equations that you can. The output conditions for a component should only be a function of inputs to that component, the component/flow properties (e.g., component efficiencies, cp/R or \gamma values depending on which is given to you, etc.), and other outputs from that component if you already have an equation for them. For example, do not try and write an expression for ue 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., cp/R, \overline{R}, \mu and molecular weight). 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 $\eta_p$. It turns out that well-designed, modern compression machines have similar polytropic efficiencies, independent of the overall pressure ratio (this is also true for turbines). 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 $\eta_p$, the adiabatic efficiency is given by

$$\eta_i = \frac{\frac{\gamma-1}{P_r} - 1}{\frac{\gamma-1}{P_r} - 1}$$

where $P_r$ is the stagnation pressure ratio across the fan or compressor.

Similarly for a turbine with a stagnation temperature ratio $T_r$ (outlet/inlet), the adiabatic efficiency can be shown to be given by

$$\eta_i = \frac{\frac{T_r - 1}{1}}{\frac{T_r \eta_p - 1}{1}}$$

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 pressurized by to be $\Delta p_{f,1}$ above the local ambient pressure and that the pump must produce an exit pressure that is higher by an amount $\Delta p_{inj}$ than the main combustor/burner inlet pressure.

5. 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 assuming that the diffuser exit (stagnation) pressure is given by

$$p_{oi} = p_{oi}(\text{adiabatic efficiency}) \times r_d,$$

where $p_{oi}(\text{adiabatic efficiency})$ is the diffuser exhaust stagnation pressure that would be calculated from the adiabatic efficiency $\eta_d$ (and which accounts for boundary layers in the inlet), and $r_d$ is the loss in (ram) stagnation pressure due to shocks outside the inlet. A standard model for $r_d$ for a well-designed inlet is based on a “milspec” (military specification), MIL-E-5008B:
6. 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 $T_{\text{max}}$ will increase according to the following expression:

$$
T_{\text{max}} = T_{\text{max},0} + C_{b1} \cdot (b/b_{\text{max}})^{1/2}
$$

where $T_{\text{max},0}$ is the maximum allowable temperature without bleed and $b_{\text{max}}$ 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, assume the bleed air returns to the main air flow at 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.

7. For the bypass fan, assume a forward mounted fan, i.e., all the air entering the engine passes through the diffuser and then the fan. After the fan, the air is diverted, some passing through the core and the rest (the bypass), exiting through the fan nozzle. The maximum bypass ratio allowed is $\beta_{\text{max}}$.

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 ($\Delta d = \text{Drag Increase}/m_a$, e.g., N/(kg/s)) is given by a modified form of a drag coefficient,

$$
\Delta d = C_{\beta} \cdot M^2 \cdot \left(\frac{p_a}{p_{\text{atm}}}\right) \cdot \beta^{1.5}
$$

where $M$ is the flight Mach number, $p_a$ is the ambient pressure of the air at the cruise conditions, and $p_{\text{atm}}$ is the standard atmospheric pressure at sea-level. Thus, the effective thrust of the engine is $\tau/m_a|_{\text{effective}} = \tau/m_a - \Delta d$.

8. When you use the combined nozzle (instead of separate core nozzle and fan nozzle for the bypass), 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

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

where $p_{o7,\text{rev}}$ is the stagnation pressure that would be calculated if the mixing was reversible and $Pr_{\text{nm}}$ describes the effective pressure loss.
Engine Cycle Simulation/Analysis

You have been tasked to design jet engines to power two new aircraft. For each aircraft, the manufacturer is interested in 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).

Your **first goal** is to identify four engine cycles, a separate cycle for each vehicle/flight condition, that provides (exactly) the desired specific thrust levels, while providing high efficiency, i.e., a low specific fuel consumption. The cycles should be based on the components defined in the Equation Development section.

Your **second goal** is to determine the maximum thrust (and the corresponding SFC) of each of these four engine cycles at all four flight conditions. For the purposes of this project, an individual engine cycle means a fixed set of components (e.g., you can’t go from a ramjet to a turbofan, and if your original cycle had no afterburner or interturbine burner, you can’t add one to find the maximum thrust), AND you can not change the compressor and fan pressure ratios, or the bypass ratio. You can change things like fuel/air ratios and bleed ratio.

Your **third goal** is to suggest a single engine cycle (not necessarily one of those chosen in the first part) for each vehicle, taking into account both flight conditions (i.e., your engine should be able to provide the necessary thrust for both) and any other considerations you feel might make the engine more “desirable” to the aircraft manufacturer. Please note; there is **no “single right answer”** in this assignment. Come up with the best designs based on your analysis and engineering insight.

### Table 1. Standard flight conditions

<table>
<thead>
<tr>
<th>Vehicle</th>
<th>Flight Condition</th>
<th>Altitude km/kft</th>
<th>$T_a$ K/R</th>
<th>$P_a$ kPa/psia</th>
<th>$M$</th>
<th>Required Specific Thrust kN/s/kg // lb$_s$/lb$_m$</th>
</tr>
</thead>
<tbody>
<tr>
<td><strong>Commercial Airliner</strong></td>
<td>Max Takeoff (MTO) Thrust (SL Static Thrust)</td>
<td>0</td>
<td>298//519</td>
<td>101.3//14.7</td>
<td>0</td>
<td>2.85// 290</td>
</tr>
<tr>
<td></td>
<td>Max Cruise Thrust (MCR) @ High Alt Cruise</td>
<td>10.7//35.0</td>
<td>219//394</td>
<td>23.8//3.45</td>
<td>0.85</td>
<td>0.875// 89.2</td>
</tr>
<tr>
<td><strong>High Performance Aircraft</strong></td>
<td>MTO Thrust @ USAFA</td>
<td>2.21//7.26</td>
<td>274//493</td>
<td>77.0//11.2</td>
<td>0</td>
<td>3.00// 306</td>
</tr>
<tr>
<td></td>
<td>MCR Thrust @ High Altitude Supersonic Cruise</td>
<td>15.2//50.0</td>
<td>216//389</td>
<td>11.6//1.68</td>
<td>1.6</td>
<td>1.06//108</td>
</tr>
</tbody>
</table>

**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.

<table>
<thead>
<tr>
<th>Component</th>
<th>$\eta_i$</th>
<th>$P_{ri}$</th>
<th>MolecWt</th>
<th>$c_p/R$</th>
<th>Other</th>
</tr>
</thead>
<tbody>
<tr>
<td>Diffuser (d)</td>
<td>0.92*</td>
<td></td>
<td>28.8</td>
<td>3.5</td>
<td></td>
</tr>
<tr>
<td>Fan (f)</td>
<td>0.90**</td>
<td>1.1≤$P_{ri}$≤1.5 if used</td>
<td>28.8</td>
<td>3.5</td>
<td>$\beta_{max}=14$, $C_{f1}=0.245$ kN·s/kg (25.0 lb·s/lbₘ)</td>
</tr>
<tr>
<td>Compressor (c)</td>
<td>0.90**</td>
<td>$P_{rc}&lt;60/P_{ri}$</td>
<td>28.8</td>
<td>3.64</td>
<td></td>
</tr>
<tr>
<td>Main Burner (b)</td>
<td>0.99</td>
<td>0.98</td>
<td>28.8</td>
<td>Note 1</td>
<td>Heating Value of fuel= 43.18 MJ/kg or 18,560 BTU/lbm</td>
</tr>
<tr>
<td>Turbine (t)</td>
<td>0.92**</td>
<td></td>
<td>28.8</td>
<td>Note 2</td>
<td>$T_{max,0}=1500$ K (2700 R) $b_{max}=0.10$, $C_{b1}=350$ K (630 R)</td>
</tr>
<tr>
<td>Turbine Mixer (tm)</td>
<td></td>
<td></td>
<td>28.8</td>
<td>Note 3</td>
<td></td>
</tr>
<tr>
<td>Interturbine Burner (ib)</td>
<td>0.98</td>
<td>0.97</td>
<td>28.8</td>
<td>Note 4</td>
<td>Same heating value as main $T_{max,ib}=1500$K (2700 R)</td>
</tr>
<tr>
<td>Fan Turbine (ft)</td>
<td>0.92**</td>
<td></td>
<td>28.8</td>
<td>Note 5</td>
<td></td>
</tr>
<tr>
<td>Afterburner (ab)</td>
<td>0.96</td>
<td>0.97 if used (1.0 if not)</td>
<td>28.8</td>
<td>Note 6</td>
<td>Same heating value as main $T_{max,ab}=2200K$ (3780 R)</td>
</tr>
<tr>
<td>Core Nozzle (n)</td>
<td>0.95*</td>
<td></td>
<td>28.8</td>
<td>Note 7</td>
<td></td>
</tr>
<tr>
<td>Fan Nozzle (fn)</td>
<td>0.97* if used</td>
<td></td>
<td>28.8</td>
<td>3.5</td>
<td></td>
</tr>
<tr>
<td>Nozzle Mixer (nm)</td>
<td></td>
<td></td>
<td>28.8</td>
<td>Note 8</td>
<td>$Pr_{in}=0.80$</td>
</tr>
<tr>
<td>Combined Nozzle (cn)</td>
<td>0.95*</td>
<td></td>
<td>28.8</td>
<td>Note 9</td>
<td></td>
</tr>
<tr>
<td>Fuel Pump (p)</td>
<td>0.35*</td>
<td></td>
<td></td>
<td></td>
<td>$\rho_{f}=780$ kg/m³ (48.7 lbm/ft³) $\Delta p_{f1}=20.7$ kPa (3.00 lb/in²) $\Delta p_{g}=550$. kPa (80.0 lb/in²)</td>
</tr>
</tbody>
</table>

*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 $c_{p,b}/R = 3.30 + 0.70 (T_{o4}/1000K)^2 - 0.20 (T_{o4}/1000K)^3$
3 $c_{p,tm}/R = 3.43 + 0.75 (T_{o5,m}/1000K)^2 - 0.21 (T_{o5,m}/1000K)^3$
4 $c_{p,ab}/R = 3.90 + 0.32 (T_{o5,ab}/1000K)^2 - 0.060 (T_{o5,ab}/1000K)^3$
5 $c_{p,ft}/R = 3.40 + 0.60 (T_{o5,ft}/1000K)^2 - 0.18 (T_{o5,ft}/1000K)^3$
6 $c_{p,ab}/R = 3.70 + 0.65 (T_{o5,ab}/1000K)^2 - 0.20 (T_{o5,ab}/1000K)^3$
7 $c_{p,ab}/R = 3.45 + 0.55 (T_{o6}/1000K)^2 - 0.18 (T_{o6}/1000K)^3$
8 $\gamma_{in} = 1.44 - 0.139 (T_{o7}/1000K) + 0.0357 (T_{o7}/1000K)^2 - 0.004 (T_{o7}/1000K)^3$
9 $c_{p,cs}/R = 3.45 + 0.55 (T_{o7}/1000K)^2 - 0.18 (T_{o7}/1000K)^3$
Written Report

You will prepare a short *professional looking* report of your designs and analysis. The *written* report (to be turned in online) should have a **Cover Page** and the following four sections: I) Executive Summary, II) Introduction and Approach, III) Results, and IV) Appendices. *(You must incorporate the sections in this order and with the above titles.)* The **Executive Summary** should not exceed one single-sided page (and should appear on its own, separate page). Combined, Sections II and III of the report should be no longer than 12 single-sided pages. This page limit *includes* all tables and figures in those sections. Note: there is no advantage to handing in a long report; covering the material described below *clearly* and succinctly will improve your grade.

The **Cover Page** should at least include (in any order): the names of your team, a title for your report, and the date. The **Executive Summary** is a *brief* statement of the problem (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 final engine designs, specific overall performance results, etc.). 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.

In the **Introduction and Approach** section, you should: 1) describe your design goals and 2) discuss your approach to solving the problem. The description of your approach should include, *at a minimum*, a reference to the equations listed in the Appendices, the computer tool(s) used to solve the equations, and how you attacked the problem - what was your “game plan.” It should NOT include any results or findings.

The **Results** section should contain a clear presentation of you numerical results and a meaningful discussion. You need to present results for the four individual engine cycle designs (including the maximum thrust values), and the results for your overall (“best”) two engine designs. In all of these cases, you must present each of the output results described above (also listed in Table 3). Remember to use effective thrust in all your results, and use the units listed in the table 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).

The **Appendices** should at least incorporate a list of equations (e.g., *Appendix A - Equations*). This appendix should list the “final” equations for the variables listed in the output section above; as noted previously, please write, **simple, short versions of the equations**. In addition, include a *flow diagram of the general engine*. The diagram should have all the components described above, and the state numbering scheme should be indicated on the diagram. You should have equations for at least all the parameters listed under Outputs above.
Table 3. List of engine 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⋅s or lbm/lbf⋅hr), based on effective specific thrust
- exhaust velocities (m/s or ft/s) from core, fan or combined nozzles [whichever used]
- fuel/air ratios in main burner, interturbine 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 turbines [if used]
- exit temperatures (K or R) and pressures (kPa or psia) for each component (use stagnation values for all components except nozzles), except the fuel pump, for which you only need to report the exit pressure
# 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; do not use them as guideposts to your design.

## Inputs

<table>
<thead>
<tr>
<th>$T_a$ (K)</th>
<th>$P_a$ (kPa)</th>
<th>M</th>
<th>$P_{rc}$</th>
<th>$P_f$</th>
<th>$\beta$</th>
<th>b</th>
<th>$f_{ib}$</th>
<th>$f_{ab}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>220</td>
<td>10.0</td>
<td>1.10</td>
<td>32</td>
<td>1.2</td>
<td>2.0</td>
<td>0.080</td>
<td>0.018</td>
<td>0.010</td>
</tr>
</tbody>
</table>

## Outputs

### $T$ and $p$ values

<table>
<thead>
<tr>
<th>$T_{o1}$ (K)</th>
<th>$p_{o1}$ (kPa)</th>
<th>$T_{o2}$ (K)</th>
<th>$p_{o2}$ (kPa)</th>
<th>$T_{o3}$ (K)</th>
<th>$p_{o3}$ (kPa)</th>
<th>$T_{o4}$ (K)</th>
<th>$p_{o4}$ (kPa)</th>
<th>$T_{o5.1}$ (K)</th>
<th>$p_{o5.1}$ (kPa)</th>
<th>$T_{o5.2}$ (K)</th>
<th>$p_{o5.2}$ (kPa)</th>
<th>$T_{o6}$ (K)</th>
<th>$p_{o6}$ (kPa)</th>
<th>$T_e$ (K)</th>
<th>$T_{ef}$ (K)</th>
<th>$T_{o7}$ (K)</th>
<th>$\gamma_{nm}$</th>
<th>$p_{o7}$ (kPa)</th>
<th>$T_{ec}$ (K)</th>
</tr>
</thead>
<tbody>
<tr>
<td>273.2</td>
<td>20.14</td>
<td>289.5</td>
<td>24.17</td>
<td>833.9</td>
<td>773.4</td>
<td>1521.0</td>
<td>758.0</td>
<td>1019.0</td>
<td>121.3</td>
<td>1005.1</td>
<td>122.0</td>
<td></td>
<td></td>
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<td></td>
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</tr>
<tr>
<td>1338</td>
<td>118.3</td>
<td>1297.0</td>
<td>103.1</td>
<td>1450.0</td>
<td>100.0</td>
<td>853.3</td>
<td>226.9</td>
<td>684.7</td>
<td>1.36</td>
<td>101.9</td>
<td>378.6</td>
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</tbody>
</table>

## Additional Values

<table>
<thead>
<tr>
<th>$w_c$ (kJ/kg)</th>
<th>$w_p$ (kJ/kg)</th>
<th>$w_{ft}$ (kJ/kg)</th>
<th>$f_{\text{max}}$</th>
<th>$f_{\text{max},ib}$</th>
<th>$f_{\text{max},ab}$</th>
</tr>
</thead>
<tbody>
<tr>
<td>572.1</td>
<td>0.16</td>
<td>49.35</td>
<td>0.0259</td>
<td>0.0149</td>
<td>0.0302</td>
</tr>
</tbody>
</table>

## Performance – Selected Values for Separate Nozzles

<table>
<thead>
<tr>
<th>$u_e$ (m/s)</th>
<th>$u_{ef}$ (m/s)</th>
<th>$\tau/\dot{m}_a$ (kN s/kg)</th>
<th>TSFC (kg/kN s)</th>
<th>$\eta_{th}$ (%)</th>
<th>$\eta_o$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1182</td>
<td>355.6</td>
<td>0.866</td>
<td>0.0381</td>
<td>48.2</td>
<td>19.9</td>
</tr>
</tbody>
</table>

## Performance – Selected Values for Combined Nozzle

<table>
<thead>
<tr>
<th>$u_{ec}$ (m/s)</th>
<th>$\tau/\dot{m}_a$ (kN s/kg)</th>
<th>TSFC (kg/kN s)</th>
<th>$\eta_p$ (%)</th>
<th>$\eta_o$ (%)</th>
</tr>
</thead>
<tbody>
<tr>
<td>803.2</td>
<td>1.37</td>
<td>0.0241</td>
<td>55.0</td>
<td>31.5</td>
</tr>
</tbody>
</table>